A09778

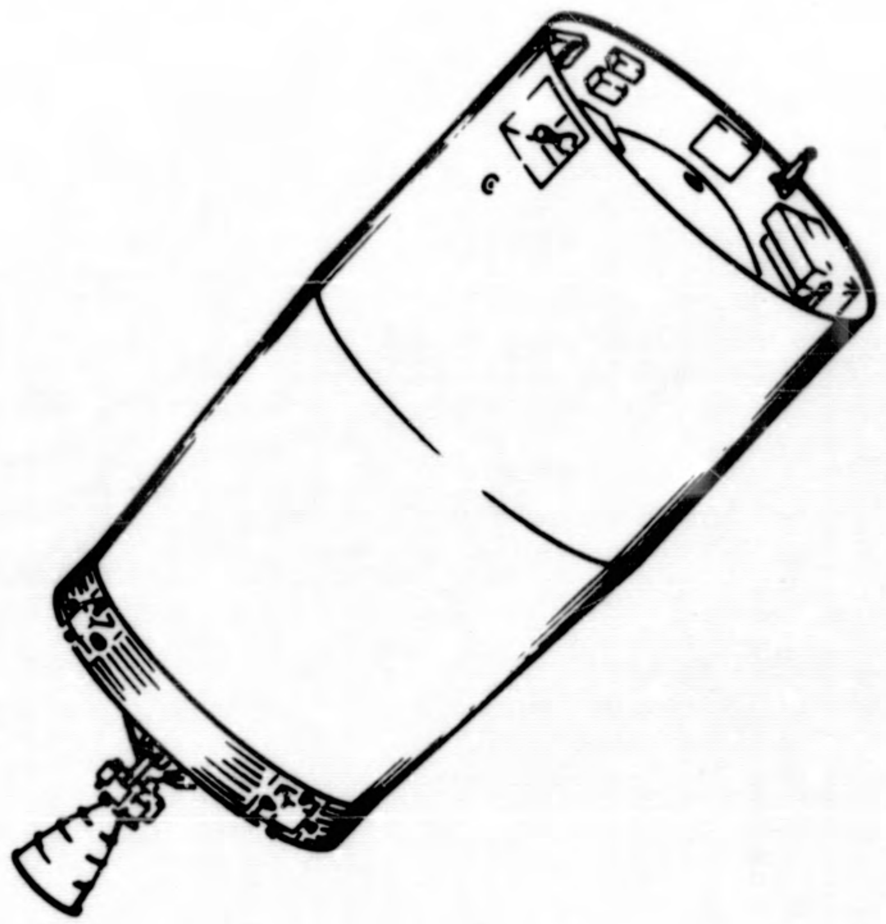
SATURN SUSTAINING ENGINEERING REPORT

SPACE TUG
POINT DESIGN STUDY
FINAL REPORT

VOLUME III
DESIGN DEFINITION
PART - I

PREPARED FOR
GEORGE C. MARSHALL
SPACE FLIGHT CENTER

SD72-SA-0032



FEBRUARY 11, 1972

Space Division
North American Rockwell

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Space Division
North American Rockwell

SD72-SA-0032

SPACE TUG POINT DESIGN STUDY FINAL REPORT

VOLUME III

DESIGN DEFINITION

PART - I

PROPULSION AND MECHANICAL, AVIONICS,
THERMAL CONTROL AND
ELECTRICAL POWER SUBSYSTEMS

FEBRUARY 11, 1972

Prepared for

George C. Marshall Space Flight Center

Approved by

R. Schwartz

R. Schwartz

Chief Program Engineer

FOREWORD

The final report on the Tug Point Design Study was prepared by the North American Rockwell Corporation through its Space Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with SA 2190 and Contract No. NAS 7-200.

The study effort described herein was conducted under the direction of NASA MSFC Study Leader, Mr. C. Gregg. The report was prepared by NR-SD, Seal Beach, California under the direction of Mr. T. M. Littman, Study Manager. The study results were developed during the period from 4 November 1971 through 11 February 1972 and the final report was submitted in February of 1972.

Valuable guidance and assistance was provided throughout the study by the following NASA/MSFC personnel:

C. Gregg - Study Leader
S. Denton - Structures
A. Willis - Avionics
J. Sanders - Propulsion
R. Nixon - Thermal Protection
A. Young - Flight Performance
R. L. Klan - Cost

The complete set of volumes comprising the report includes:

I Summary
II Operations, Performance, and Requirements
III Design Definition

Part 1 - Propulsion and Mechanical Subsystems, Avionic Subsystems, Thermal Control, and Electrical Power Subsystem

Part 2 - Insulation Subsystems, Meteoroid Protection, Structures, Mass Properties, Ground Support Equipment, Reliability, and Safety

IV Program Requirements
V Cost Analysis

This part of Volume III presents the details of the Point Design Tug in the areas of propulsion and mechanical, avionics, thermal control, and electrical power subsystems. The data generated include engineering drawings, schematics, subsystems operation, and component description. Also presented are the various options investigated and the rationale for the point design selection.

ABSTRACT

The primary objective of the Tug Point Design Study was to verify through detail design and analysis the performance capability of a baseline design to deliver and retrieve payloads between 100 nautical miles/28.5 degrees inclination and geosynchronous. The Tug as groundruled for the study, is ground-based, reusable for 20 mission cycles, and is shuttled to and from low earth orbit by an Earth Orbital Shuttle (EOS) with a 65,000 pound payload capability. A 1976 state-of-the-art also was groundruled for the investigations.

The results of the effort show that the baseline concept can be designed to meet the target performance goals. Round trip payload capability to geosynchronous orbit is 3720 pounds; 720 pound margin over the established goal.

The design analysis performed to ascertain the Tug propellant mass fraction encompassed definition of the vehicle primary structure, thermal control, meteoroid protection, propulsion and mechanical subsystems, and avionics including power generation and distribution.

Graphite-epoxy composite material was determined to be feasible for Tug use and resulted in considerable weight savings. The concept of employing the primary load-carrying outer shell as a multi-function element integrating the meteoroid shield and insulation purge bag requirements is also feasible and enhances design simplicity. In addition, the use of a dual-mode pressure schedule during boost to orbit when applied loads are highest resulted in minimum tank weight. This, combined with an integrated gaseous O_2/H_2 auxiliary propulsion for stability and control, main tanks prepressurization, and fuel cell usage yield a minimum weight and operationally simple system.

Reliability and Safety analyses verified that no single failure of a component would result in a critical or unsafe condition. This was accomplished employing redundancy as required, notably in propulsion subsystems valving and attitude control components.

Program requirements were developed to verify the feasibility, producibility and operational capability of the point design. The results indicate that an "on-condition" maintenance approach similar to that used by commercial airlines and military operations would effectively serve Tug requirements.

Technology development study effort was concentrated on identifying the technologies needed for the baseline design. The more critical technologies requiring development include high performance engines, high performance insulation, large composite structures, and avionics.

A preliminary program development schedule was structured summarizing the integrated activities necessary to support the Tug through design development, production, and ground and flight testing.

The cost analysis performed covered the five major cost categories of DDT&E, first unit production, SR&T, average flight maintenance and refurbishment, and flight test vehicle refurbishment.

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ILLUSTRATIONS

Space Division
North American Rockwell

ILLUSTRATIONS

ILLUSTRATIONS

ILLUSTRATIONS

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TABLES

1.0 INTRODUCTION

The Space Tug is a high performance propulsion stage designed to operate as an orbital maneuvering stage launched by the two-stage Space Shuttle. Because of the nature of the Tug mission, performance capability is very sensitive to Tug mass fraction. This study was conducted to answer the questions "What Tug mass fractions are really achievable by 1980?", and "What level of technology effort is required in order to build a Tug having the high performance defined in NASA/MSFC's Study Plan (Reference 1)?". Both questions are discussed below.

1.1 BACKGROUND

Several pre-Phase A Tug/OOS (Orbit-to-Orbit Shuttle) studies have been conducted for NASA and USAF agencies with a wide variation in the mass fractions quoted. NR performed a reusable Space Tug study for NASA-MSC in 1970-71 (Reference 2) and both NR and MDAC evaluated OOS feasibility for SAMSO/Aerospace Corporation in 1971 (References 3, 4). Additionally, two European teams conducted Tug system studies for the European Space Agency (ELDO) during 1970-71 (References 5, 6). Investigations also have been accomplished by MSFC and Aerospace Corporation. These studies considered a wide variety of design concepts and autonomy limits, ground and space-based operational requirements, degree of reusability, unmanned and manned payload implications, single and multi-stages, and different technology bases.

Projected NASA and DOD missions for the 1980's and beyond demand a Tug designed for a high degree of reusability and operational flexibility to assure significant improvement in space flight economy. Furthermore, Tug design must be compatible with Shuttle orbiter cargo bay size, weight limitations, and environment. For a ground-based system, consideration also must be given to Shuttle transport of a mated Tug/Payload.

1.2 OBJECTIVES

This point design study had one primary aim which was to be verified by design detail and analysis; namely, that a reusable, ground-based Space Tug with an IOC target by about the end of 1979 (1976 state-of-the-art) can carry a 3000-pound round trip payload between orbits at 100 nautical miles/28.5 degrees inclination and geosynchronous. The key constraint was use of a Space Shuttle having a 65,000 pound orbital delivery capability. A minimum usable propellant mass fraction of 0.895 also was desired. Additional study objectives were to (1) define the necessary supporting research and technology (SR&T) activities and their associated funding, and (2) determine Tug development, first production, and maintenance/repair costs.

1.3 STUDY SCOPE

The detail design of an integrated system was performed for a baseline concept. The concept was derived from MSFC's Study Plan and NR-selected materials, fabrication techniques, and subsystems resulting from currently available data and new trade studies.

Concurrent with the baseline study, options were evaluated having the potential for improving Tug mass fraction and mission performance. Emphasis was placed on the areas of alternate materials and subsystems, flight mode and operational variations, and use of advanced technology.

The study logic of Figure 1.3-1 depicts the major functional activities and outputs of these activities. The analyses performed to satisfy study objectives can be subdivided into three inter-related major efforts which started at study outset and ran concurrently to completion. Initiation of these efforts at the same time was made possible by the large amount of technical data available from the data bank. System requirements and criteria definition and program support gave the design definition effort the input data necessary for realistic structural, mechanical, thermal, and avionics subsystems design taking into account reliability and safety requirements. The three major tasks formed an iterative loop to the extent that the study schedule permitted. As the design of each component and subsystems evolved, the results were fed to the supporting activities which served to increase the depth of analysis and visibility of the overall system characteristics with each succeeding step. This approach also adapted itself to the timely establishment of performance sensitivities and development of potentially attractive subsystem concepts.

1.4 STUDY GUIDELINES

This section highlights those elements of the NASA Study Plan (Reference 1) which were most influential in directing the NR effort toward the achievement of the aforementioned objectives.

Key Assumptions and Guidelines

The items listed below provided the key design and operational drivers for the Tug:

1. 1976 Materials & Concepts Technology

2. Unmanned Design, Fail-Safe Operation

3. Reusable - Lifetime of 20 Missions

4. Ground Based - Refurbishment After Each Mission

5. 6-Day on Orbit Stay Time Unattached to Shuttle

6. Flight Between 100 n mi Circular, 28.5 degrees Inclination & Geosynchronous Orbit

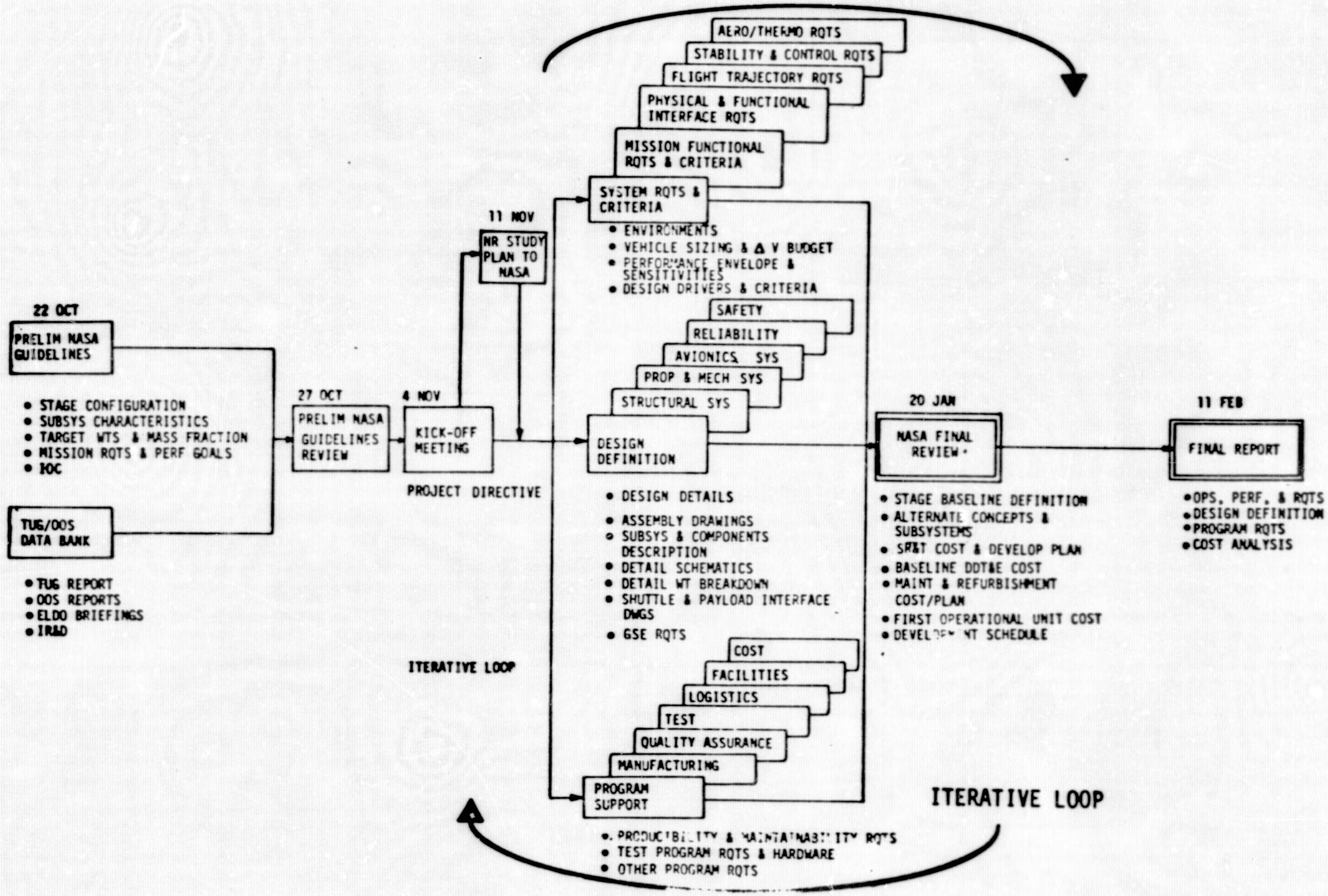

Figure 1.3-1 Top Level Study Logic

7. Payload Deliver/Retrieve Mixes in pounds

Baseline	3K/3K	
Alternates	0/4.16K 8.06K/0	Sizes Outer Shell Structure

8. Abort From Orbit Only & Propellant Dump/Inerting From Cargo Bay

9. Integrated MPS & APS Subsystems

Low vehicle weight was a key design criterion due to the aforementioned performance objectives. Therefore, strong emphasis was given to the use of advanced materials and concepts deemed part of the 1976 technology base, but achievable without incurring severe cost penalties or high development risks. Fail-safe (FS) operations also provide for lower weight due to redundancy limitations (compared to the more demanding FO/FS requirements as employed in the OSS studies). However, FS does necessitate the highest practical component reliability to achieve an acceptable (over 0.9) mission success probability. Fail-safe is defined here as no failure modes which would cause an unsafe situation for the Shuttle or its crew, or destruction of the Tug payload. In the event of mission abort (limited to abort from orbit) while the Tug is still in the cargo bay, propellant dumping, tank inerting, and subsystems safing are required. These capabilities also are specified for normal reentry and landing conditions to minimize hazards.

Unmanned design necessitates a high degree of subsystem/operational autonomy with ground support provided as emergency backup or when it yields weight and design simplicity advantages.

Reusability for 20 mission cycles (which may cover a period in excess of 3 years) can only be achieved in a practical cost-effective sense if airline-type servicing techniques are developed for Tug (as is planned for Shuttle). Strong attention must be given to assure a design compatible with this approach (accessibility, ease of inspection, and checkout).

The six-day orbital stay time affects cryogenic tankage protection and the total space exposure (for 20 missions) specifies meteoroid shielding requirements.

The baseline (3000 pound round trip) payload capability represents the most demanding from a performance (mass fraction) viewpoint. However, normal Shuttle ascent and descent carrying the Tug and the alternate payloads were employed to size the Tug outer shell structure, based on the flight load factors provided by MSFC for the study.

One additional assumption agreed to between MSFC and NR, use of an integrated LOX/LH_2 propellant system for both main and auxiliary propulsion, provides design simplicity as well as weight and performance advantages.

Tug Baseline Concept

The NASA baseline configuration (Figure 1.4-1) which served as the starting point for this study is a single stage orbital propulsion system. It is limited to a maximum overall diameter of 15 feet and a maximum length of 35 feet, including Shuttle/Tug and Payload/Tug docking mechanisms. This vehicle is intended to separate from the Shuttle in orbit at 100 n mi/ 28.5 degrees inclination with a 3000-pound payload (15 ft x 25 ft) attached, ascent to geosynchornous orbit, deploy the up payload, retrieve a 3000-pound payload within 6000 n mi of the deployed payload, return to the near-vicinity of the Shuttle, redock, and return to earth. Payload center-of-gravity was defined as being at the geometric center of the 15 x 25 feet payload envelope.

The Tug has a non-integral tankage arrangement and is sized for a total propellant capacity of 56, 394 pounds including 350 pounds of reserve plus allocations for reaction control/auxiliary propulsion (APS), fuel cell, residuals, and losses. The LH_2 tank has hemispherical bulkheads and a cylindrical section, whereas the LOX tank consists of two ellipsoidal bulkheads.

The docking systems are designed such that the active portion is left with the Tug in the Tug/payload interface and with the Shuttle in the Tug/ Shuttle interface.

Other pertinent features are indicated on the profile. It should be noted that the Tug is attached at its aft end to the forward part of the orbiter cargo bay and thus is transported between Earth and orbit in an inverted attitude.

Tug Weight Targets

Table 1.4-1 lists the "bogey" weights provided by MSFC as design goals to assure meeting mass fraction requirements with the constraints of a 65,000 pound Shuttle capability and a 3000-pound Tug payload. No specific allocation was made for Tug-supportive hardware and fluids which remain in the EOS cargo bay. Instead, these were assumed to be contained within structure and other subsystems.

Structure includes all dry structure (docking mechanisms, meteoroid shield, outer shell, supports, thrust structure) and tankage subsystems. Thermal control includes cryogenic insulation, avionics cooling/heating hardware, and purge systems. Avionics contains GN&C, communications, data management, power generation and distribution, rendezvous and docking, and Tug electrical interfaces for ground and Shuttle and provisions for on-board checkout. Propulsion includes dry main engine, propellant feed, pressurization, fill/drain and vent/purge umbilicals, propellant feed, pressurization, fill/drain and vent/purge umbilicals, propellant dump, tank baffles/screens, APS thrusters/feed system/tanks, main engine actuators, and ullage venting control.

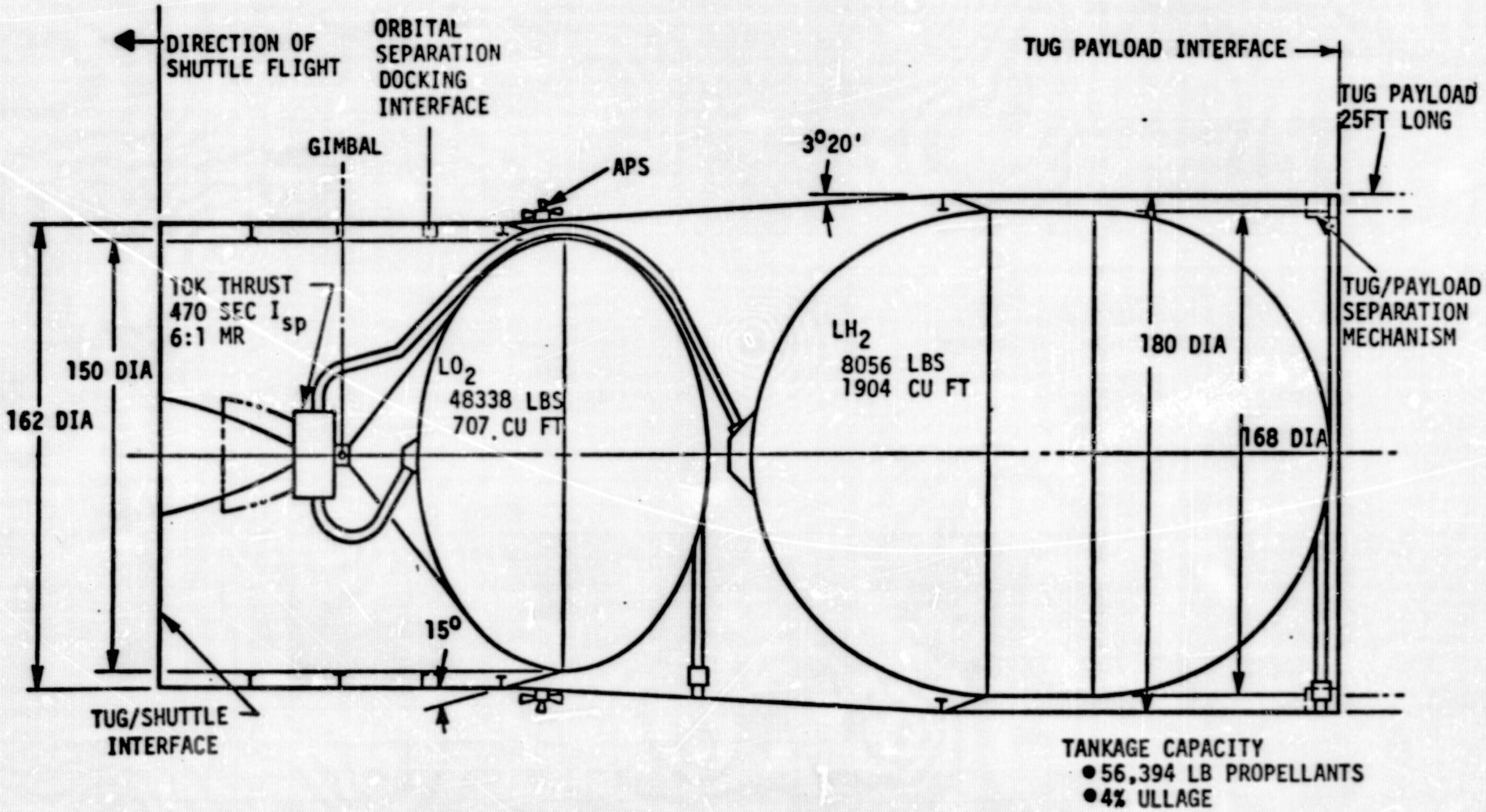

Figure 1.4-1 NASA Tug Baseline Concept & Sizing

Non-usable fluids include propellant reserves, pressurant, thermal control fluids, and residuals. The main engine propellant bogey weight contains all propellant burned by the main engine during a nominal mission. APS propellant includes all burned attitude control and small delta-V translational maneuver requirements during a nominal mission. The miscellaneous fluids category contains all other unburned fluids (fuel cell, reactants, and vent/childown/start-stop losses). These have been numerically lumped together with non-usable fluids in the bogey weight table.

Table 1.4-1. Tug Bogey Weights

	Weight (lb)
Structure	2,552
Thermal control	476
Avionics	1,011
Propulsion	1,057
Dry Weight	5,096
10% contingency	510
Non-usable fluids	842*
Burnout Weight	6,448
Usable main engine propellant	55,148
Usable APS propellant	404
Misc fluids & losses	--
Tug flight wt at Tug/EOS separation	62,000
EOS payload - chargeable interface prov	--
Tug gross wt at EOS liftoff	62,000
Gross EOS payload at liftoff	65,000
Mass fraction, λ	0.895
*Incl 350 lb prop reserve	

1.5 INTEGRATED SYSTEM

The Space Tug may be structurally divided into four (4) basic sections:

1. The outer shell or unpressurized structure.

2. The propellant tanks and supports with their associated equipment or the pressurized structure.

3. The thrust structure and engine system.

4. Miscellaneous mounting provisions for various subsystem equipment (Avionics, APS, EPS, etc.)

As shown on the inboard profile (Figure 1.5-1) the Tug is comprised of a shell structure within which the propellant tanks and engine system (thrust structure) are suspended. Wherever feasible the structure has been designed with a multi-purpose function in order to minimize the structural weight. This is most evident in the cylindrical outer shell where in addition to providing the primary structural load path, the shell functions as a purge bag and meteoroid shield. In every instance concerning the structure, the lightest weight system has been of primary importance.

The outer shell structure can be divided into the forward skirt, the intertank structure, the aft skirt and the EOS adapter. Each of these shells is a graphite epoxy skin over aluminum honeycomb core structure of varying thicknesses and shapes.

The forward skirt is cylindrical, 180 inches in diameter and 147.5 inches long. The shell is 3/8 of an inch thick. The aft end of the cylinder attaches to the forward end of the intertank structure at the LH_2 tank attach frame. The forward skirt is mechanically fastened at this joint.

At the forward end the shell provides the interface with the payload and consequently there is a deep channel ring frame at this location. This frame is a graphite epoxy composite with gussets and web stiffeners, of the same material, as required. It is 16 inches deep at the interface plane. Payload docking and latching provisions are accommodated on this frame. Four additional stability ring frames are equally spaced on the forward skirt. The first frame aft of the forward interface is a larger frame, 6 inches deep, since it provides support for the docking probes. The remaining three frames are small stability frames, 1.5 inches deep. All of the frames are layed up from piles of graphite epoxy composite.

This shell is composed of 4 quarter panel subassemblies which are spliced with longitudinal strips of graphite epoxy composite (same as facing sheets). Several cutouts are provided in the shell to accommodate the antennas, umbilicals, and star tracker and horizon sensor field of views. Reinforcement is provided around the smaller cutouts by added plies of the facing sheet material while around the larger cutouts, channel-shaped intercostals between frames on each side of the opening provide the necessary stiffening. The heavy frame at the forward interface distributes the orbiter cargo bay attachment loads, and supports the payload attachment. These external fittings, which terminate in hemispherical ends, engage the orbiter cargo bay supports. Each fitting is loaded only in the tangential direction to the frame. Two fittings (Z-Z in orbiter) to carry vertical loads only are located on the side of the ring and one lower fitting reacts the lateral load (Y-Y in orbiter). The ring frame then distributes these loads to the shell structure.

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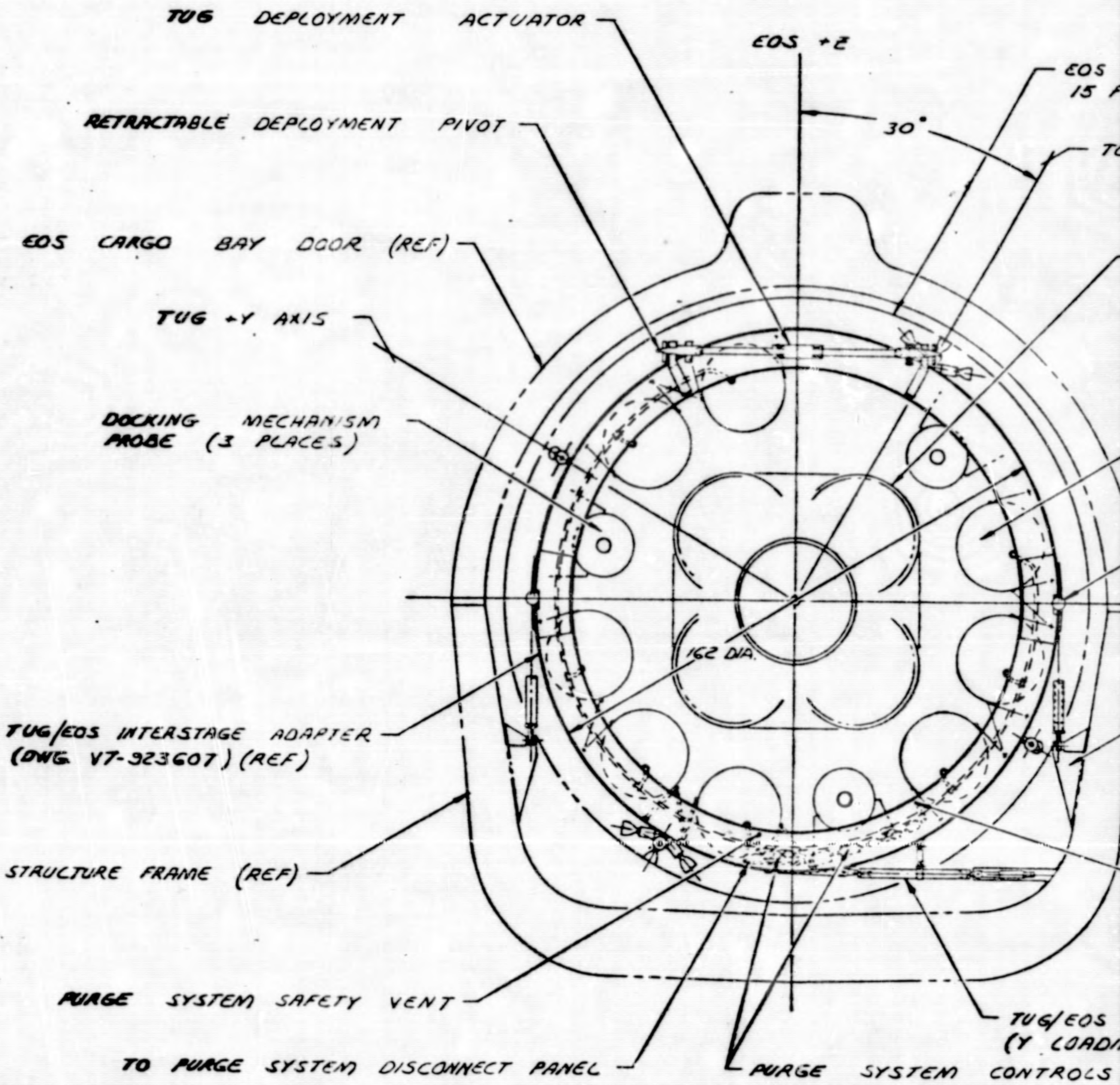
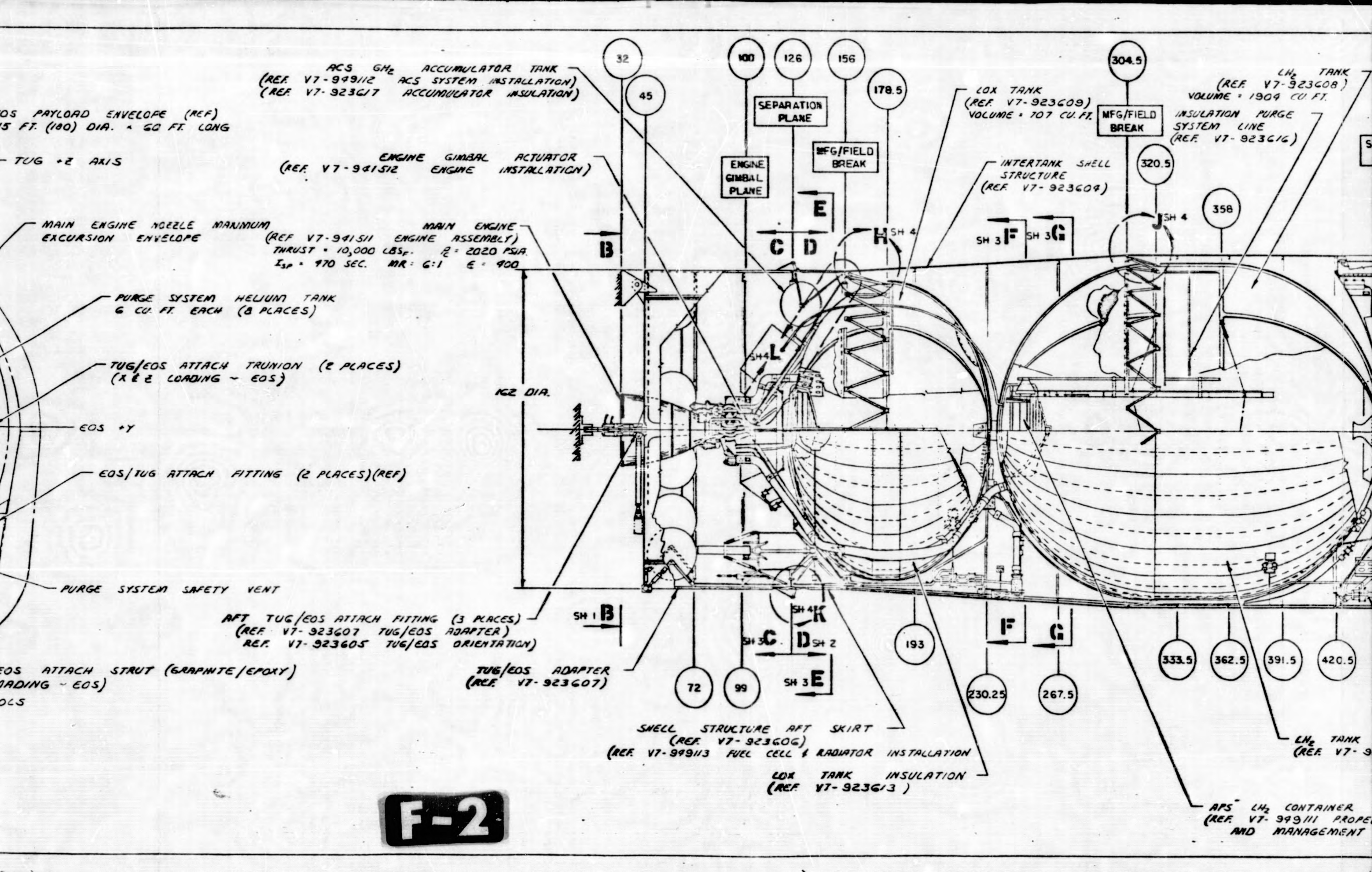
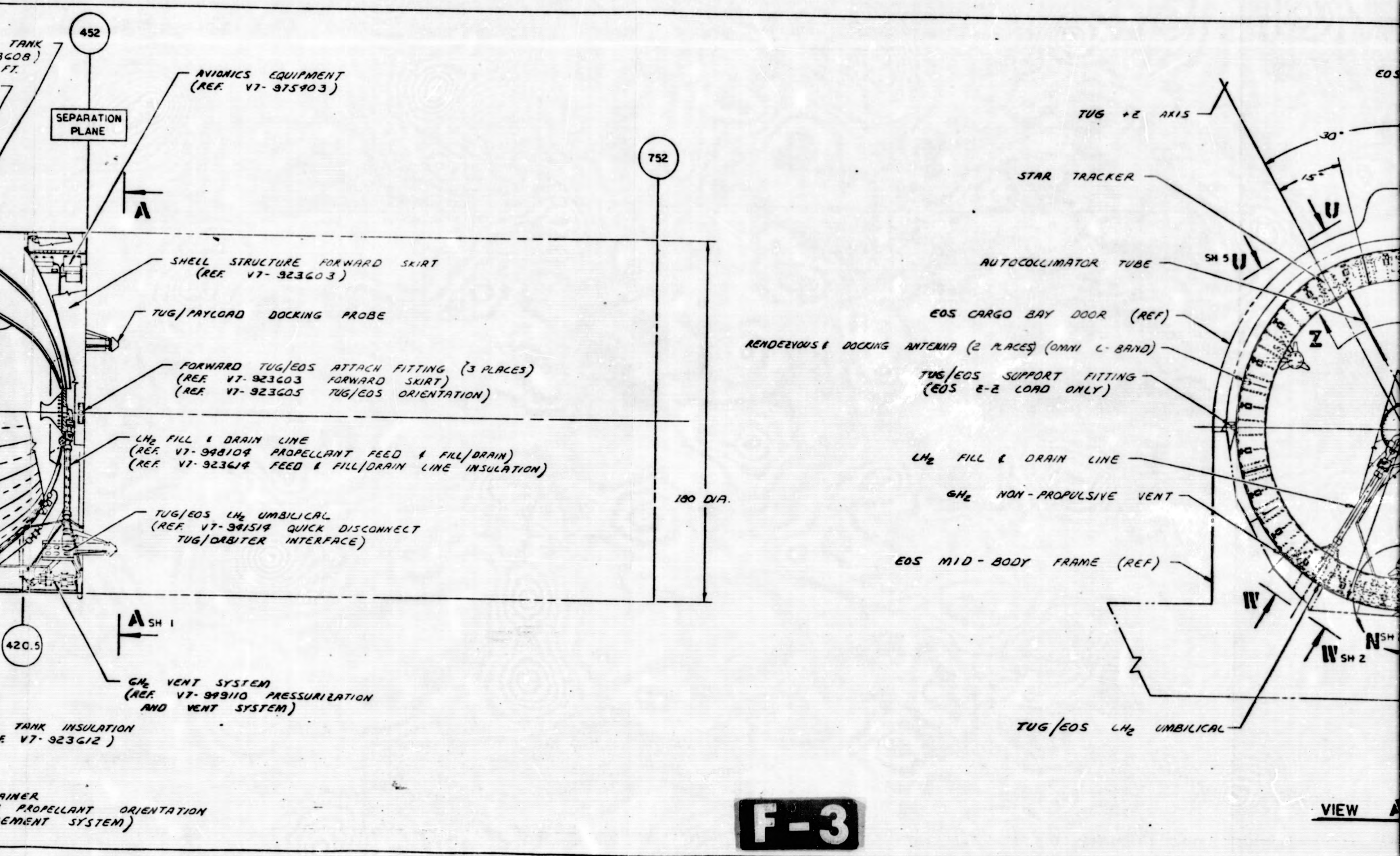


VIEW B - B SH 1

F-1



ACS GH2 ACCUMULATOR TANK
(REF V7-949112 ACS SYSTEM INSTALLATION)
(REF V7-923617 ACCUMULATOR INSULATION)
OS PAYLOAD ENVELOPE (REF)
15 FT. (180) DIA. × 60 FT. LONG
TUG +Z AXIS
ENGINE GIMBAL ACTUATOR
(REF V7-941512 ENGINE INSTALLATION)
MAIN ENGINE NOZZLE MAXIMUM EXCURSION ENVELOPE
MAIN ENGINE
(REF V7-941511 ENGINE ASSEMBLY)
THRUST = 10,000 LBSF. Pc = 2020 PSIA
Isp = 470 SEC. MR = 6:1 ε = 400
PURGE SYSTEM HELIUM TANK
6 CU. FT. EACH (8 PLACES)
TUG/EOS ATTACH TRUNION (2 PLACES)
(X & Z LOADING ~ EOS)
162 DIA.
EOS +Y
EOS/TUG ATTACH FITTING (2 PLACES)(REF)
PURGE SYSTEM SAFETY VENT
AFT TUG/EOS ATTACH FITTING (3 PLACES)
(REF V7-923607 TUG/EOS ADAPTER)
REF V7-923605 TUG/EOS ORIENTATION)
EOS ATTACH STRUT (GRAPHITE/EPOXY)
LOADING ~ EOS)
TUG/EOS ADAPTER
(REF V7-923607)
32
45
100
126
156
178.5
304.5
320.5
358
SEPARATION PLANE
ENGINE GIMBAL PLANE
MFG/FIELD BREAK
LOX TANK
(REF V7-923609)
VOLUME = 707 CU. FT.
INTERTANK SHELL STRUCTURE
(REF V7-923604)
LH2 TANK
(REF V7-923608)
VOLUME = 1909 CU. FT.
INSULATION PURGE SYSTEM LINE
(REF V7-923616)
B
C
D
E
F
G
H SH 4
J SH 4
SH 3 F
SH 3 G
SH 4 L
SH 1 B
SH 4 K
SH 3 C
D SH 2
SH 3 E
72
99
193
230.25
267.5
333.5
362.5
391.5
420.5
SHELL STRUCTURE AFT SKIRT
(REF V7-923606)
(REF V7-949113 FUEL CELL & RADIATOR INSTALLATION
LOX TANK INSULATION
(REF V7-923613)
LH2 TANK
(REF V7-9
APS LH2 CONTAINER
(REF V7-949111 PROPE
AND MANAGEMENT
F-2


TANK
3608)
FT.
452
SEPARATION PLANE
AVIONICS EQUIPMENT
(REF. V7-975403)
752
A
SHELL STRUCTURE FORWARD SKIRT
(REF. V7-923603)
TUG/PAYLOAD DOCKING PROBE
FORWARD TUG/EOS ATTACH FITTING (3 PLACES)
(REF. V7-923603 FORWARD SKIRT)
(REF. V7-923605 TUG/EOS ORIENTATION)
LH2 FILL & DRAIN LINE
(REF. V7-948104 PROPELLANT FEED & FILL/DRAIN)
(REF. V7-923614 FEED & FILL/DRAIN LINE INSULATION)
180 DIA.
TUG/EOS LH2 UMBILICAL
(REF. V7-941514 QUICK DISCONNECT
TUG/ORBITER INTERFACE)
A SH 1
420.5
GH2 VENT SYSTEM
(REF. V7-949110 PRESSURIZATION
AND VENT SYSTEM)
TANK INSULATION
E V7-923612)
AINER
PROPELLANT ORIENTATION
EMENT SYSTEM)
EOS
TUG +Z AXIS
30°
15°
STAR TRACKER
U
SH 5 U
AUTOCOLLIMATOR TUBE
EOS CARGO BAY DOOR (REF)
Z
RENDEZVOUS & DOCKING ANTENNA (2 PLACES) (OMNI C-BAND)
TUG/EOS SUPPORT FITTING
(EOS Z-Z LOAD ONLY)
LH2 FILL & DRAIN LINE
GH2 NON-PROPULSIVE VENT
EOS MID-BODY FRAME (REF)
W
W SH 2
N SH
TUG/EOS LH2 UMBILICAL
VIEW A

F-3

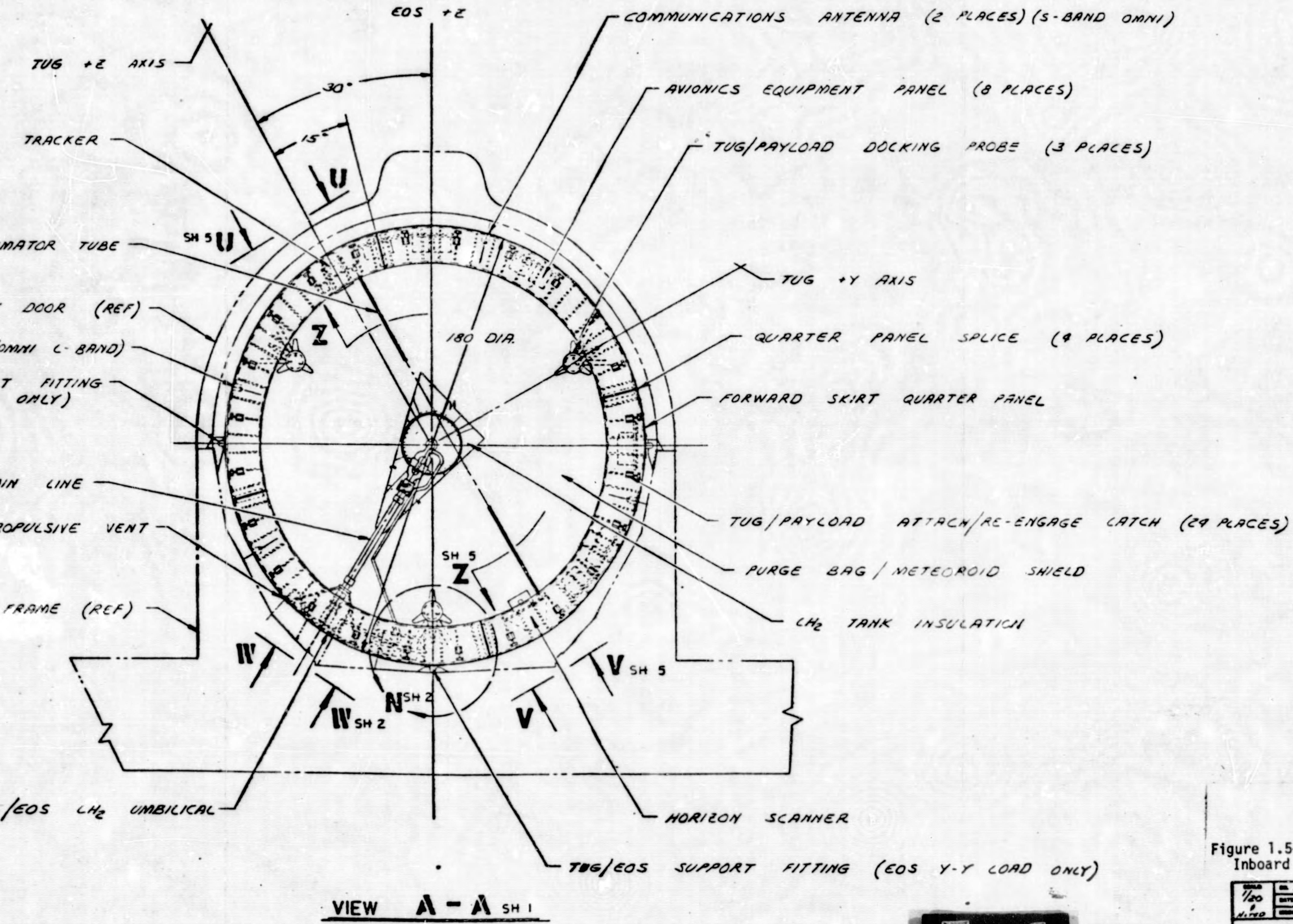
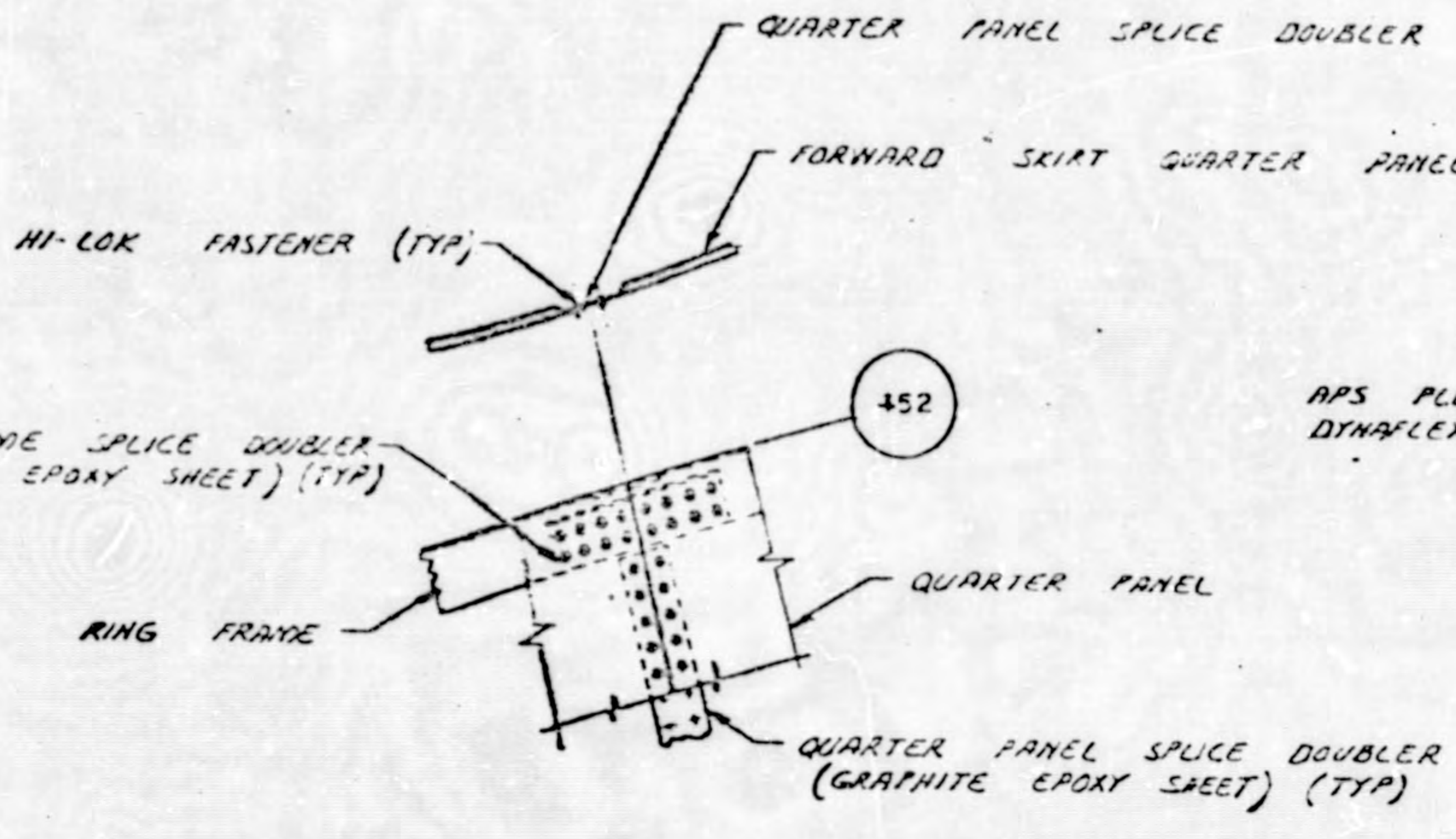


Figure 1.5-1
Inboard Profile Tug (Sheet 1 of 5)

SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION	INBOARD PROFILE TUG	V7-923602 SH 1 OF 5

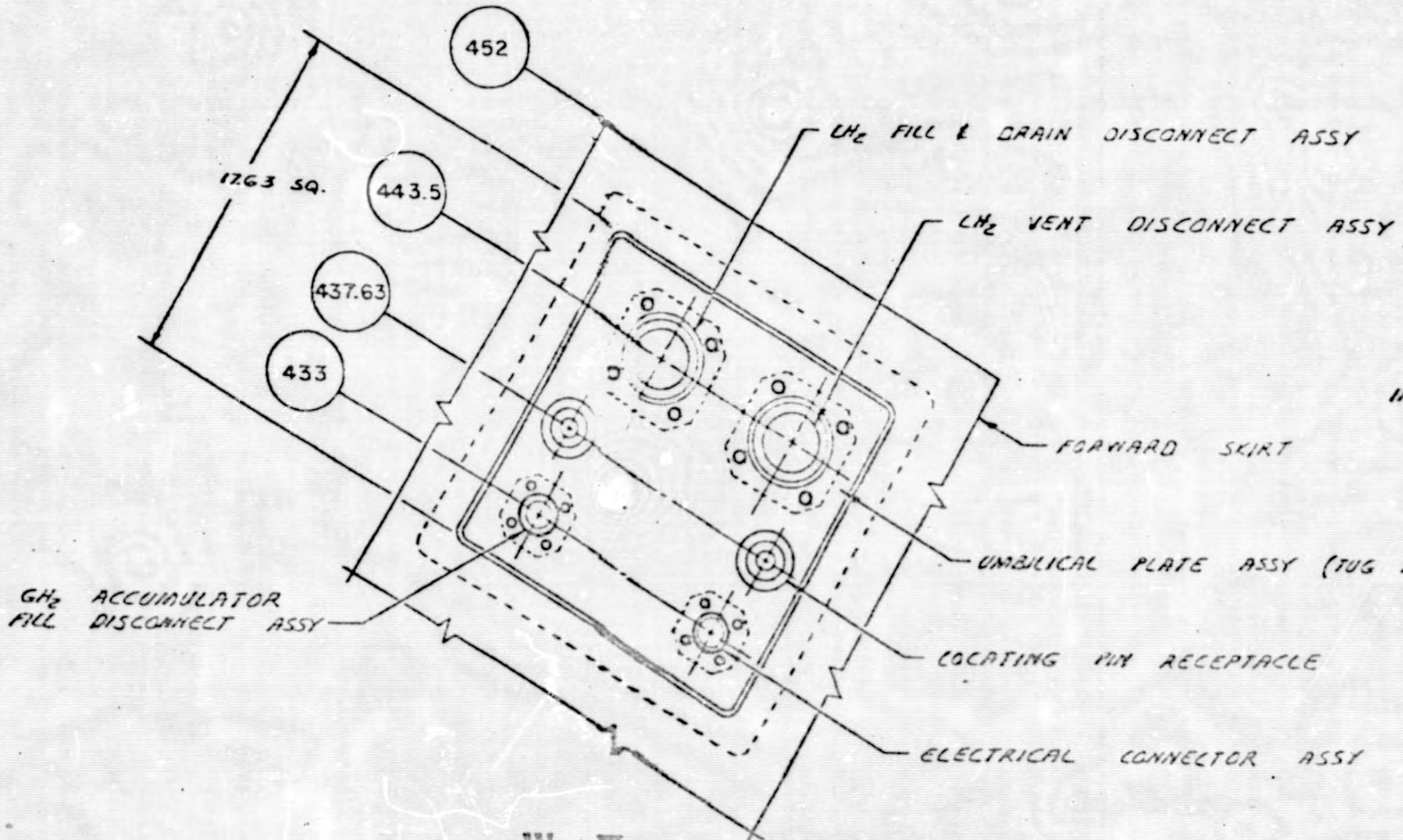
F-4

1-9, 1-10

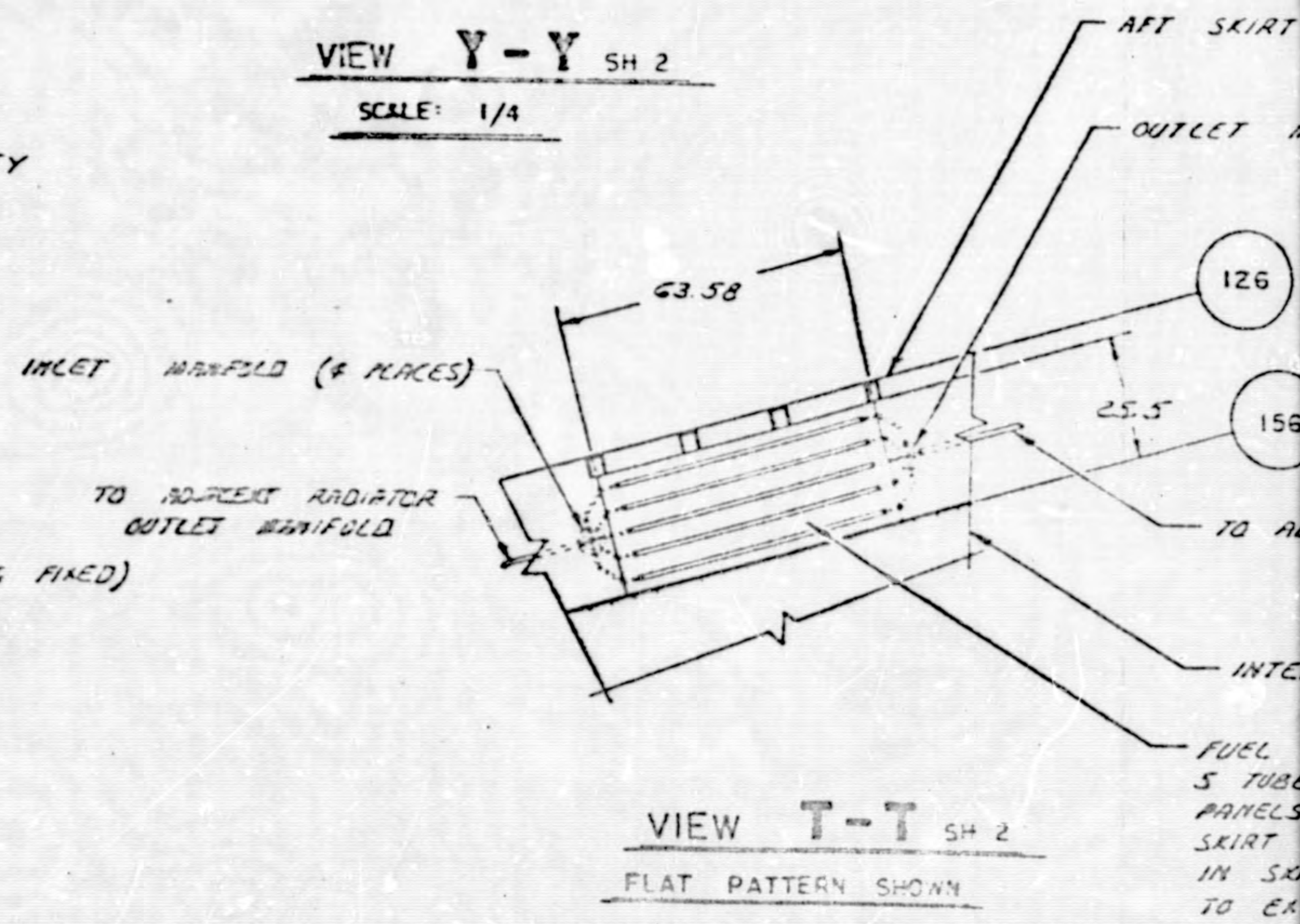


VIEW Y-Y SH 2
SCALE: 1/4

APS PLUME IMPINGE
DYNAFLEX R.F. 2700

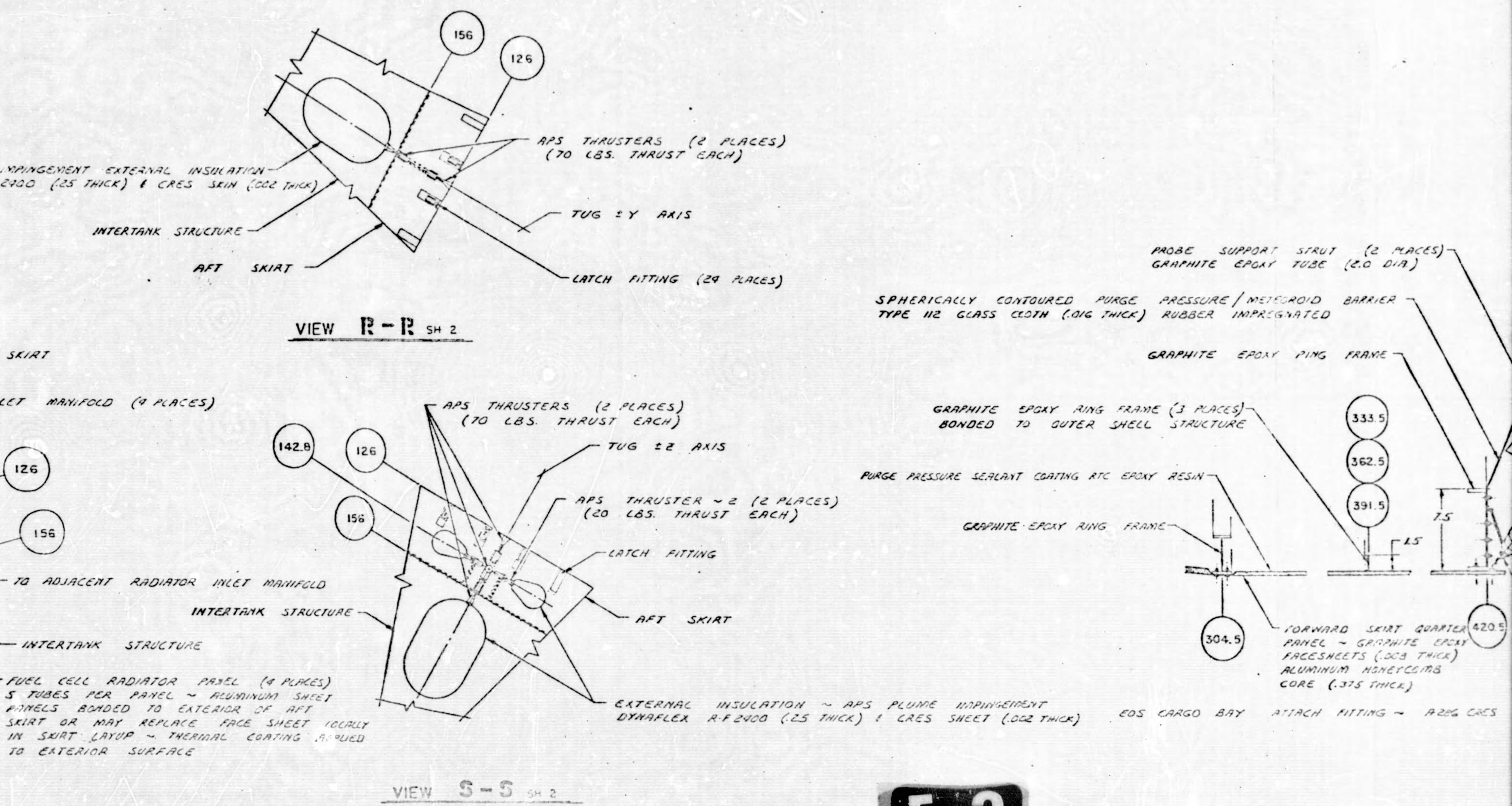


VIEW W-W SH 1
SCALE: 1/4



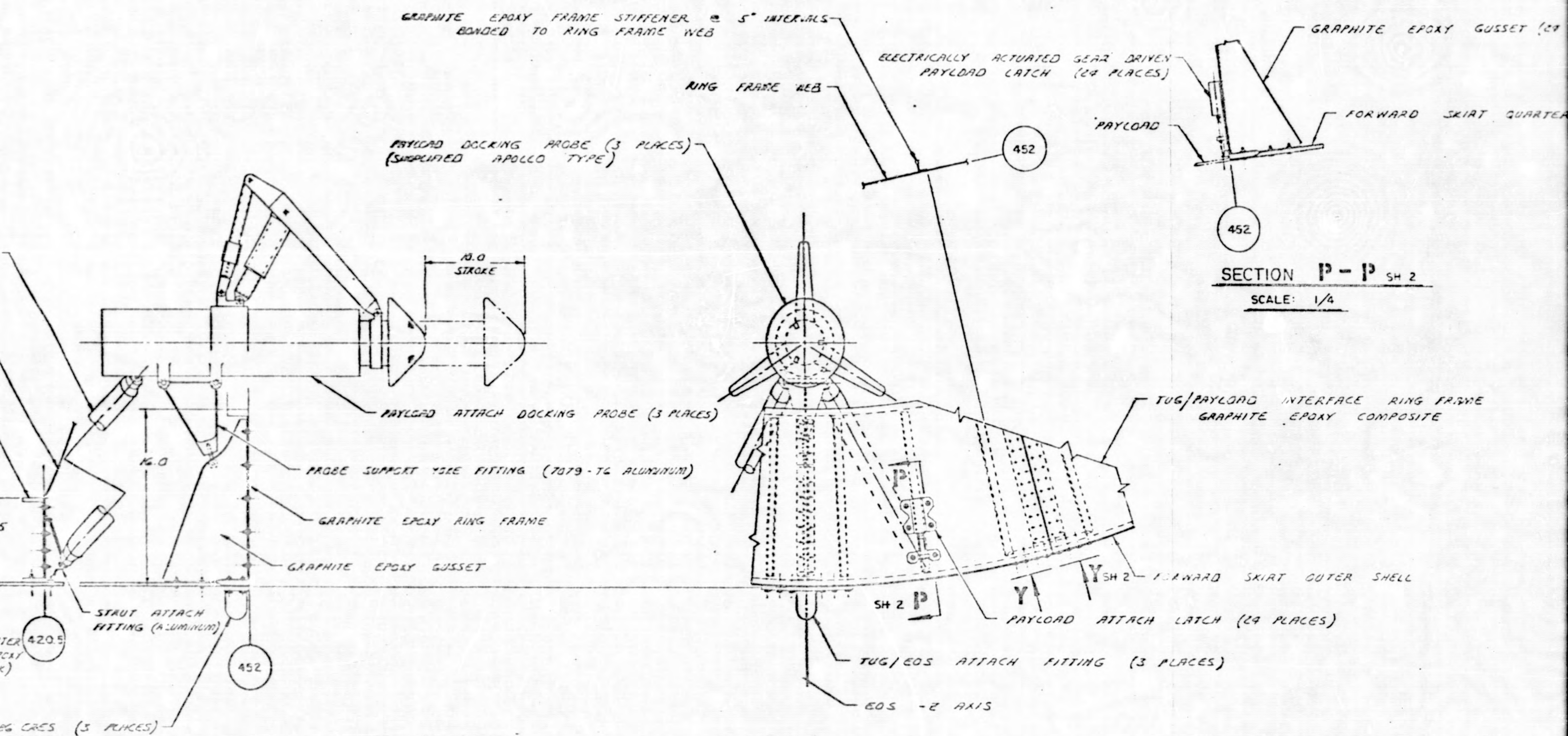
VIEW T-T SH 2
FLAT PATTERN SHOWN

F-1



156
126
APS THRUSTERS (2 PLACES)
(70 LBS. THRUST EACH)
IMPINGEMENT EXTERNAL INSULATION
2400 (.25 THICK) & CRES SKIN (.002 THICK)
TUG ±Y AXIS
INTERTANK STRUCTURE
AFT SKIRT
LATCH FITTING (24 PLACES)
VIEW R - R SH 2
SKIRT
LET MANIFOLD (4 PLACES)
126
156
TO ADJACENT RADIATOR INLET MANIFOLD
INTERTANK STRUCTURE
FUEL CELL RADIATOR PANEL (4 PLACES)
5 TUBES PER PANEL ~ ALUMINUM SHEET
PANELS BONDED TO EXTERIOR OF AFT
SKIRT OR MAY REPLACE FACE SHEET LOCALLY
IN SKIRT LAYUP ~ THERMAL COATING APPLIED
TO EXTERIOR SURFACE
APS THRUSTERS (2 PLACES)
(70 LBS. THRUST EACH)
142.8
126
156
TUG ±Z AXIS
APS THRUSTER ~ 2 (2 PLACES)
(20 LBS. THRUST EACH)
LATCH FITTING
INTERTANK STRUCTURE
AFT SKIRT
EXTERNAL INSULATION ~ APS PLUME IMPINGEMENT
DYNAFLEX R-F 2400 (.25 THICK) & CRES SHEET (.002 THICK)
VIEW S - S SH 2
PROBE SUPPORT STRUT (2 PLACES)
GRAPHITE EPOXY TUBE (2.0 DIA)
SPHERICALLY CONTOURED PURGE PRESSURE / METEOROID BARRIER
TYPE 112 GLASS CLOTH (.016 THICK) RUBBER IMPREGNATED
GRAPHITE EPOXY RING FRAME
GRAPHITE EPOXY RING FRAME (3 PLACES)
BONDED TO OUTER SHELL STRUCTURE
333.5
362.5
391.5
PURGE PRESSURE SEALANT COATING RTC EPOXY RESIN
7.5
GRAPHITE EPOXY RING FRAME
1.5
304.5
FORWARD SKIRT QUARTER
PANEL ~ GRAPHITE EPOXY
FACESHEETS (.008 THICK)
ALUMINUM HONEYCOMB
CORE (.375 THICK)
420.5
EOS CARGO BAY ATTACH FITTING ~ A286 CRES

F-2



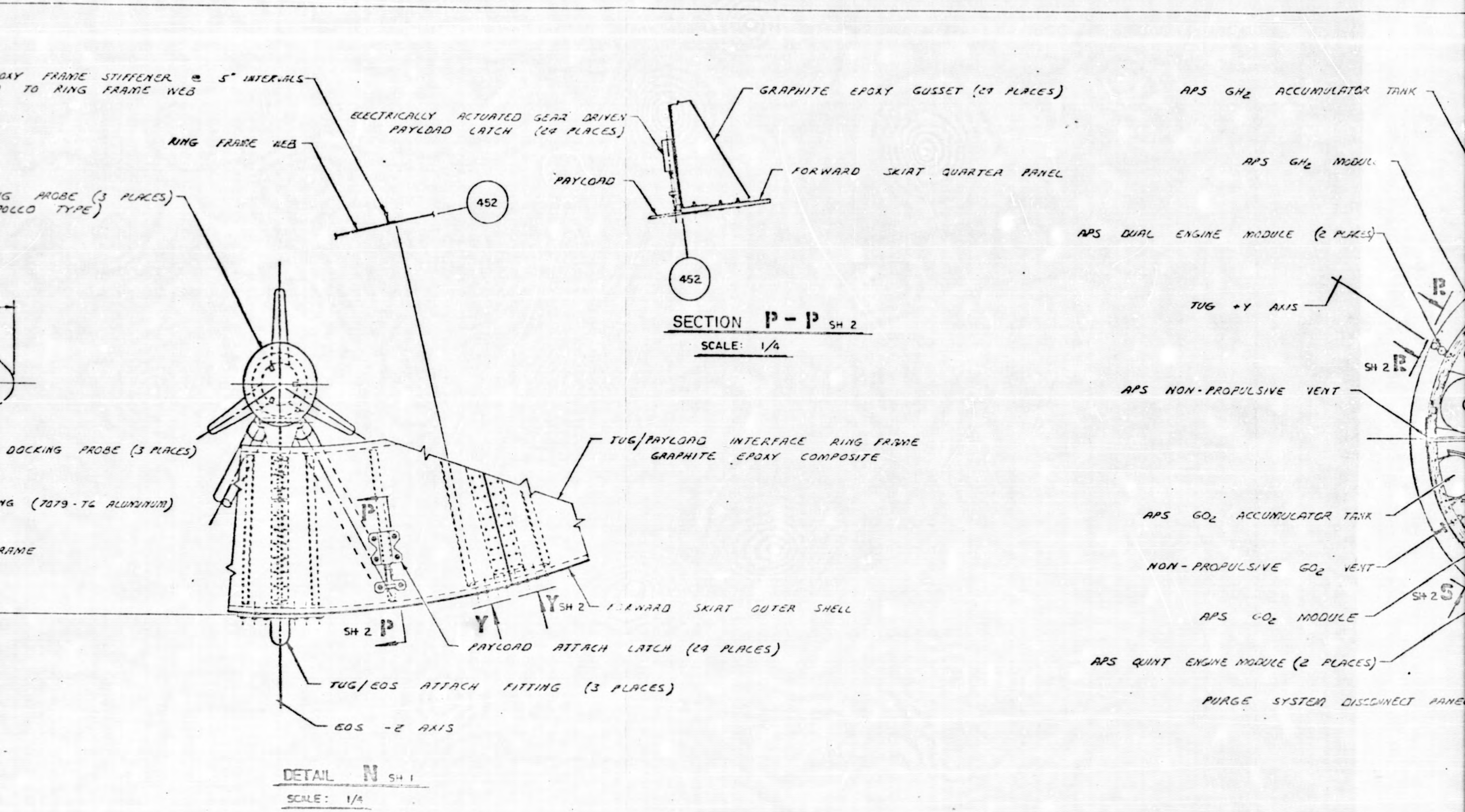
SECTION P - P SH 2

SCALE: 1/4

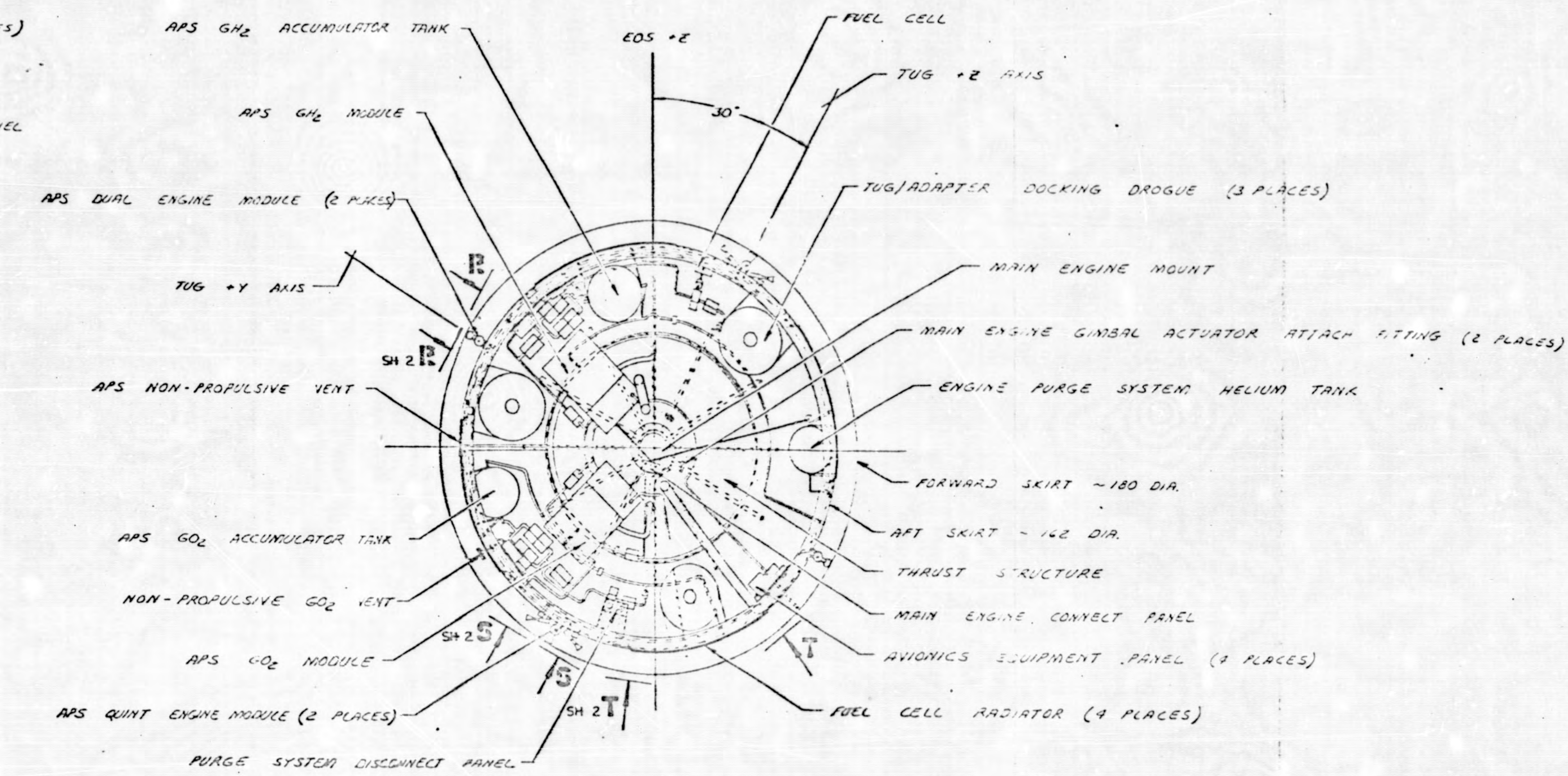
DETAIL N SH 1

SCALE: 1/4

F-3


OXY FRAME STIFFENER @ 5" INTERVALS
TO RING FRAME WEB
ELECTRICALLY ACTUATED GEAR DRIVEN PAYLOAD LATCH (24 PLACES)
RING FRAME WEB
452
G PROBE (3 PLACES)
OLLO TYPE)
DOCKING PROBE (3 PLACES)
NG (7079-T6 ALUMINUM)
RAME
TUG/PAYLOAD INTERFACE RING FRAME
GRAPHITE EPOXY COMPOSITE
P
SH 2 P
Y
Y SH 2
FORWARD SKIRT OUTER SHELL
PAYLOAD ATTACH LATCH (24 PLACES)
TUG/EOS ATTACH FITTING (3 PLACES)
EOS -Z AXIS
DETAIL N SH 1
SCALE: 1/4
GRAPHITE EPOXY GUSSET (24 PLACES)
PAYLOAD
FORWARD SKIRT QUARTER PANEL
452
SECTION P - P SH 2
SCALE: 1/4
APS GH_2 ACCUMULATOR TANK
APS GH_2 MODUL
APS DUAL ENGINE MODULE (2 PLACES)
R
TUG +Y AXIS
SH 2 R
APS NON-PROPULSIVE VENT
APS GO_2 ACCUMULATOR TANK
NON-PROPULSIVE GO_2 VENT
SH 2 S
APS GO_2 MODULE
APS QUINT ENGINE MODULE (2 PLACES)
PURGE SYSTEM DISCONNECT PANEL

F-4



SECTION D - D SH 1

Figure 1.5-1
Inboard Profile Tug (Sheet 2 of 5)

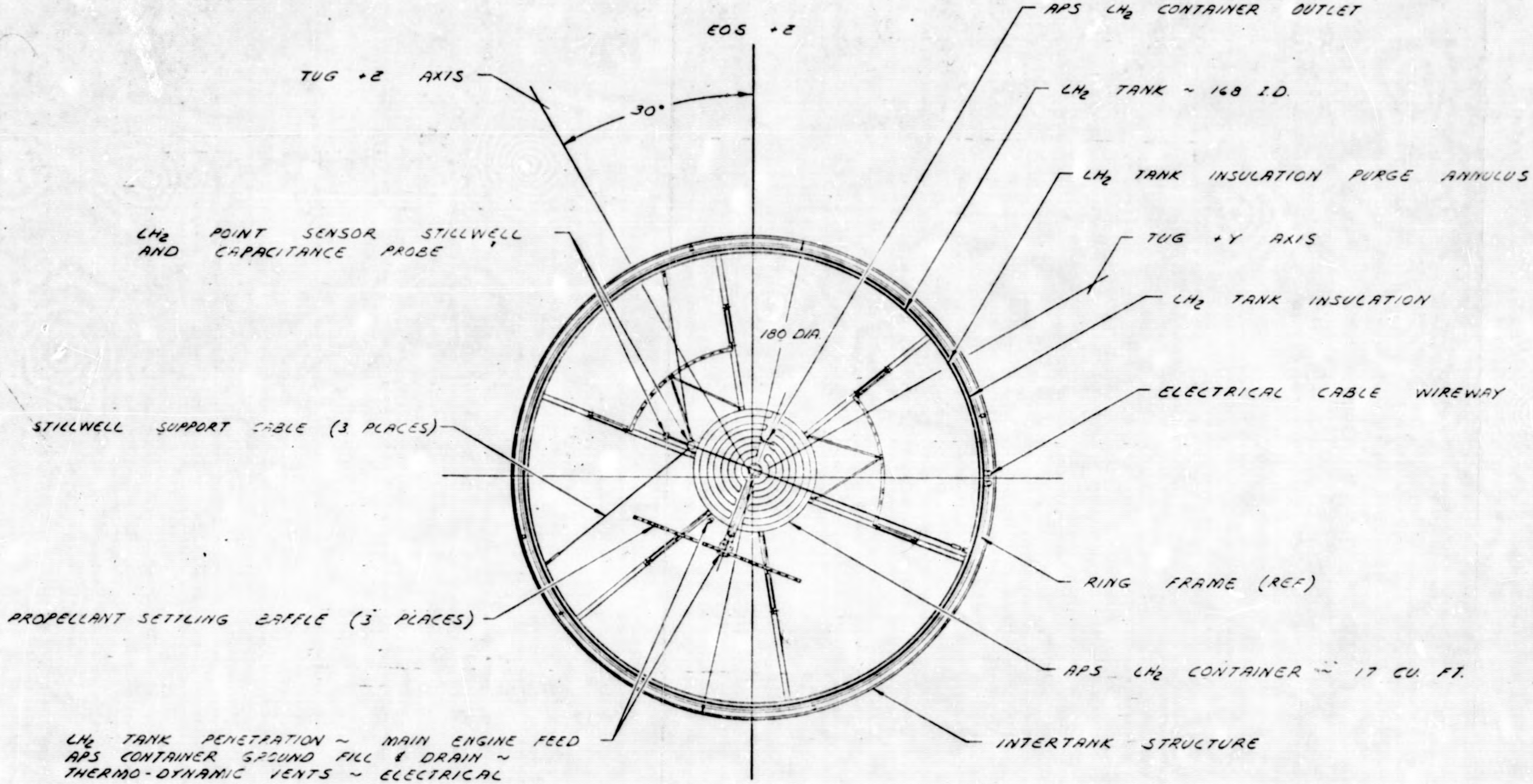
F-5

1 - 11, 1 - 12

NORTH AMERICAN ROCKWELL CORPORATION

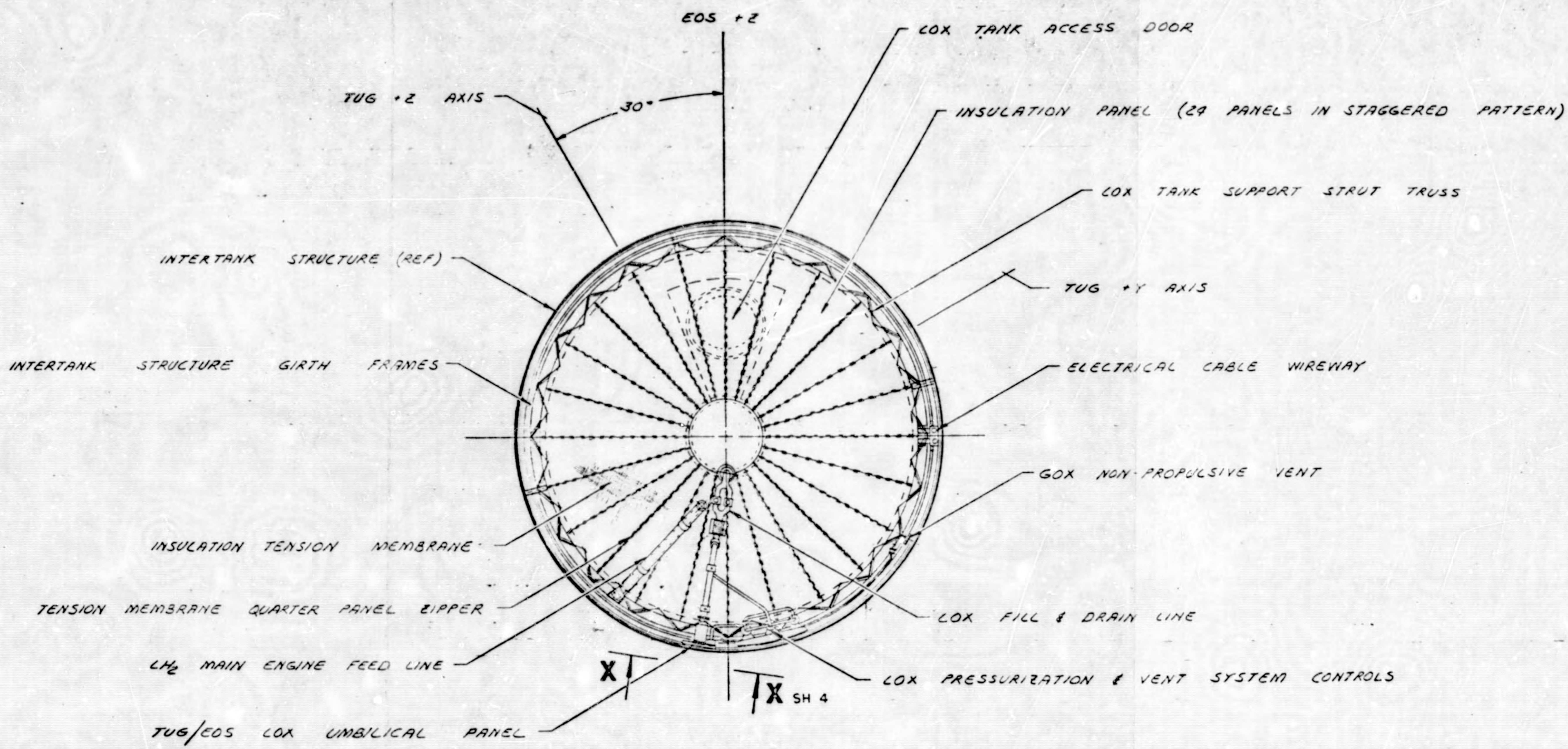
INBOARD PROFILE
TUG

V7-923602
SH 2 OF 5



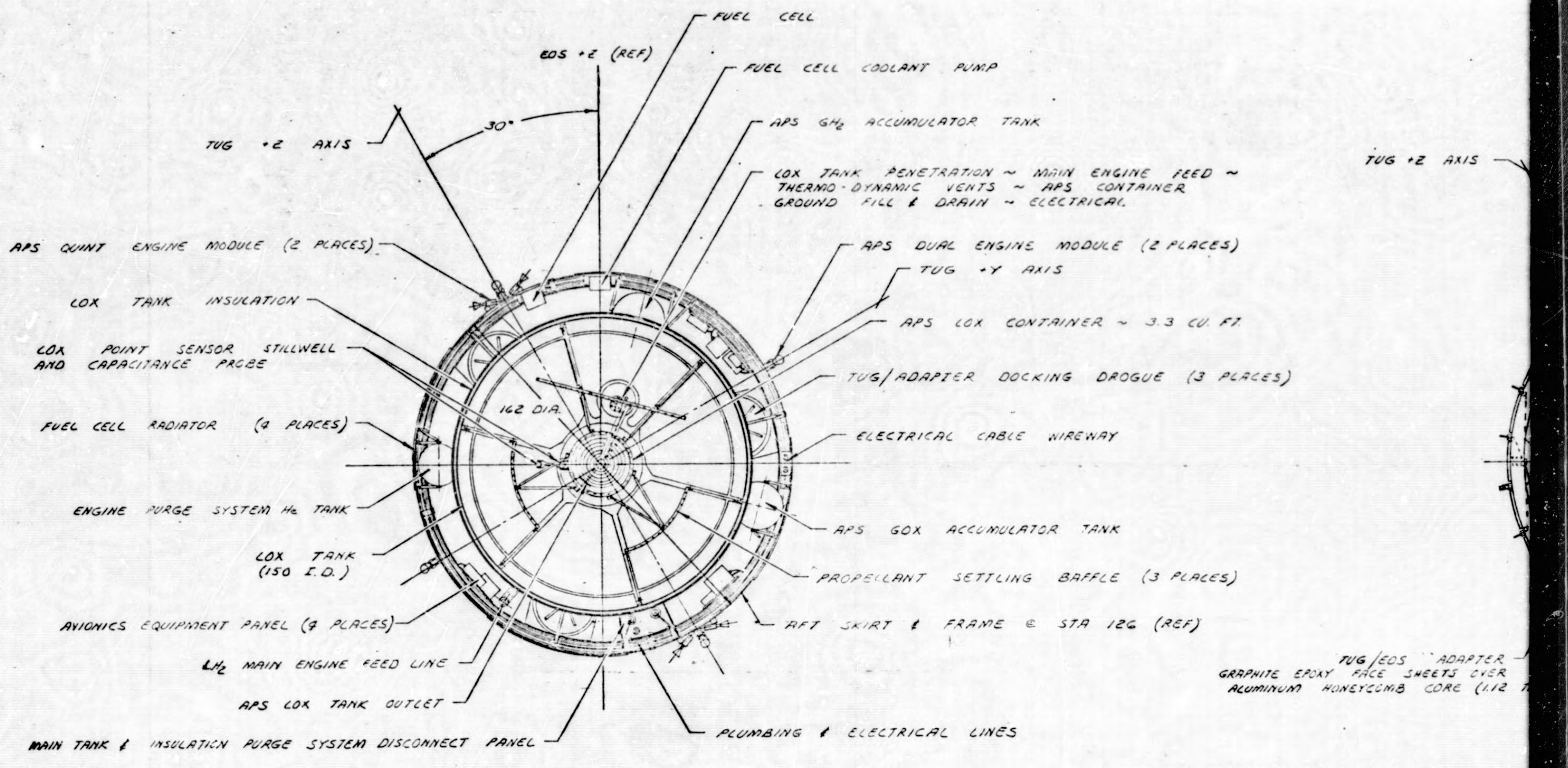
SECTION G – G SH 1

F-1



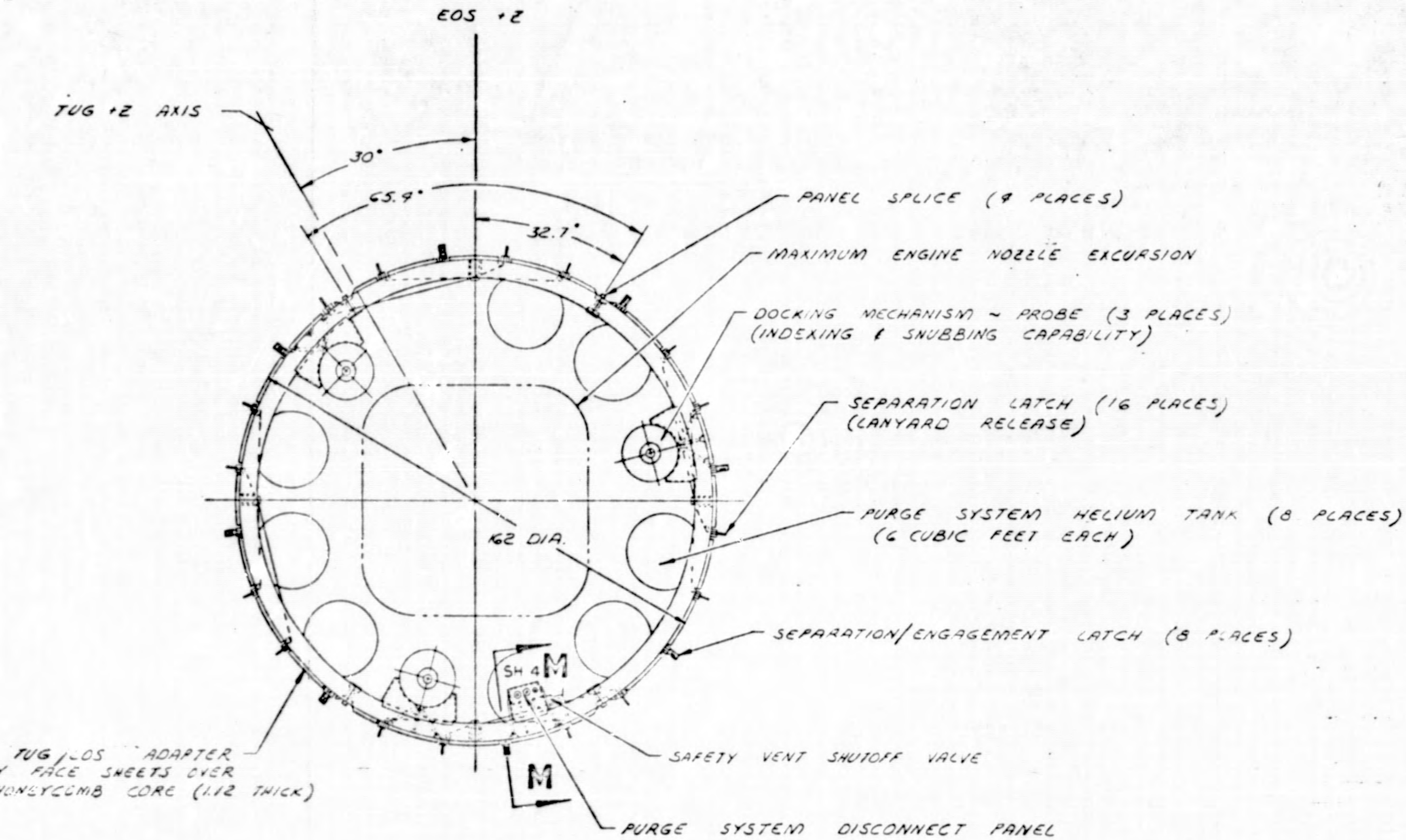
SECTION F - F SH 1

F-2



SECTION E - E SH 1

F-3

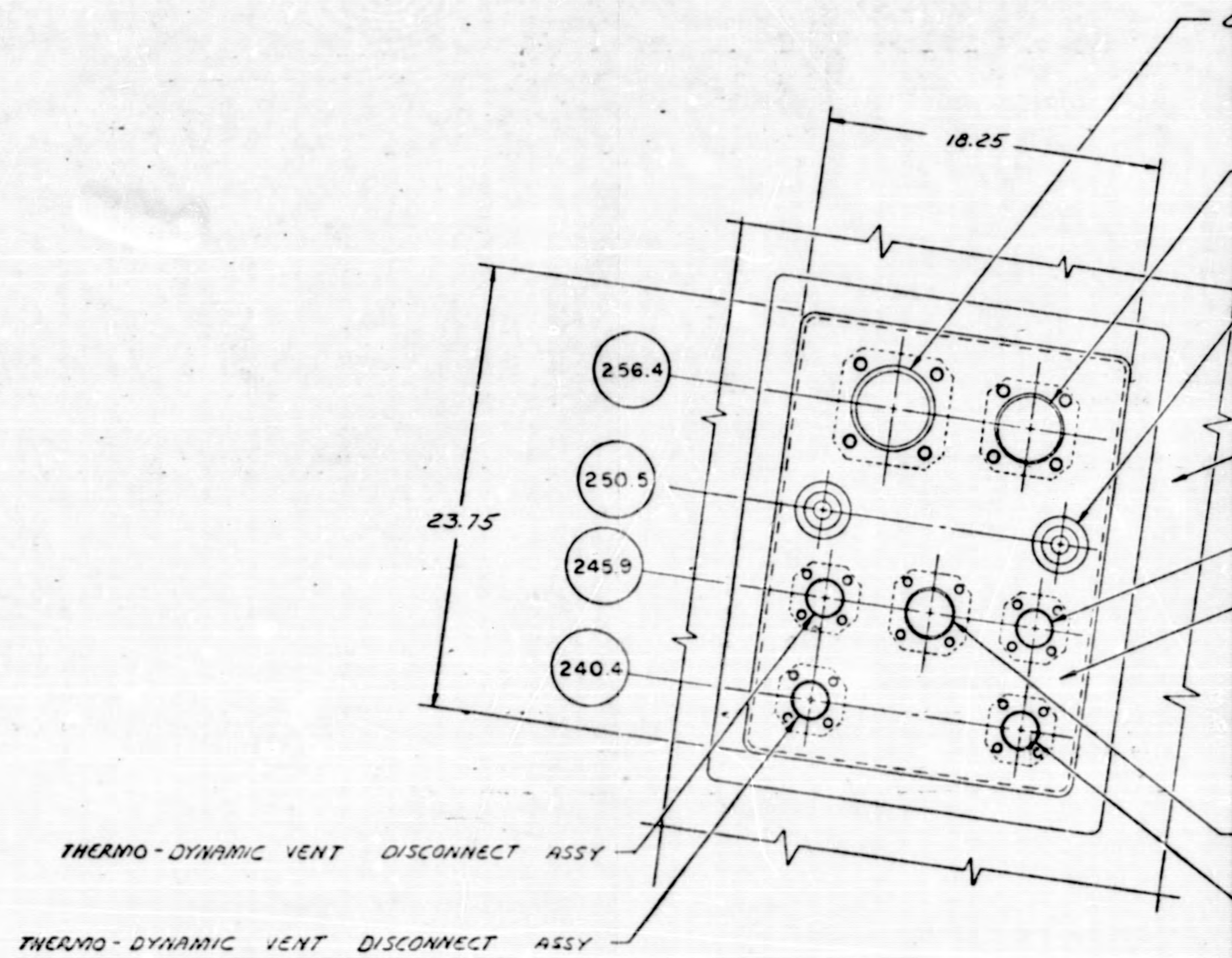


SECTION C - C SH 1

Figure 1.5-1
Inboard Profile Tug (Sheet 3 of 5)

SCALE 1/20 NOTED	DATE 1-28-72 MODEL TUG	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY CALIFORNIA	
	INBOARD PROFILE TUG		V7-923602 SH 3 OF 5

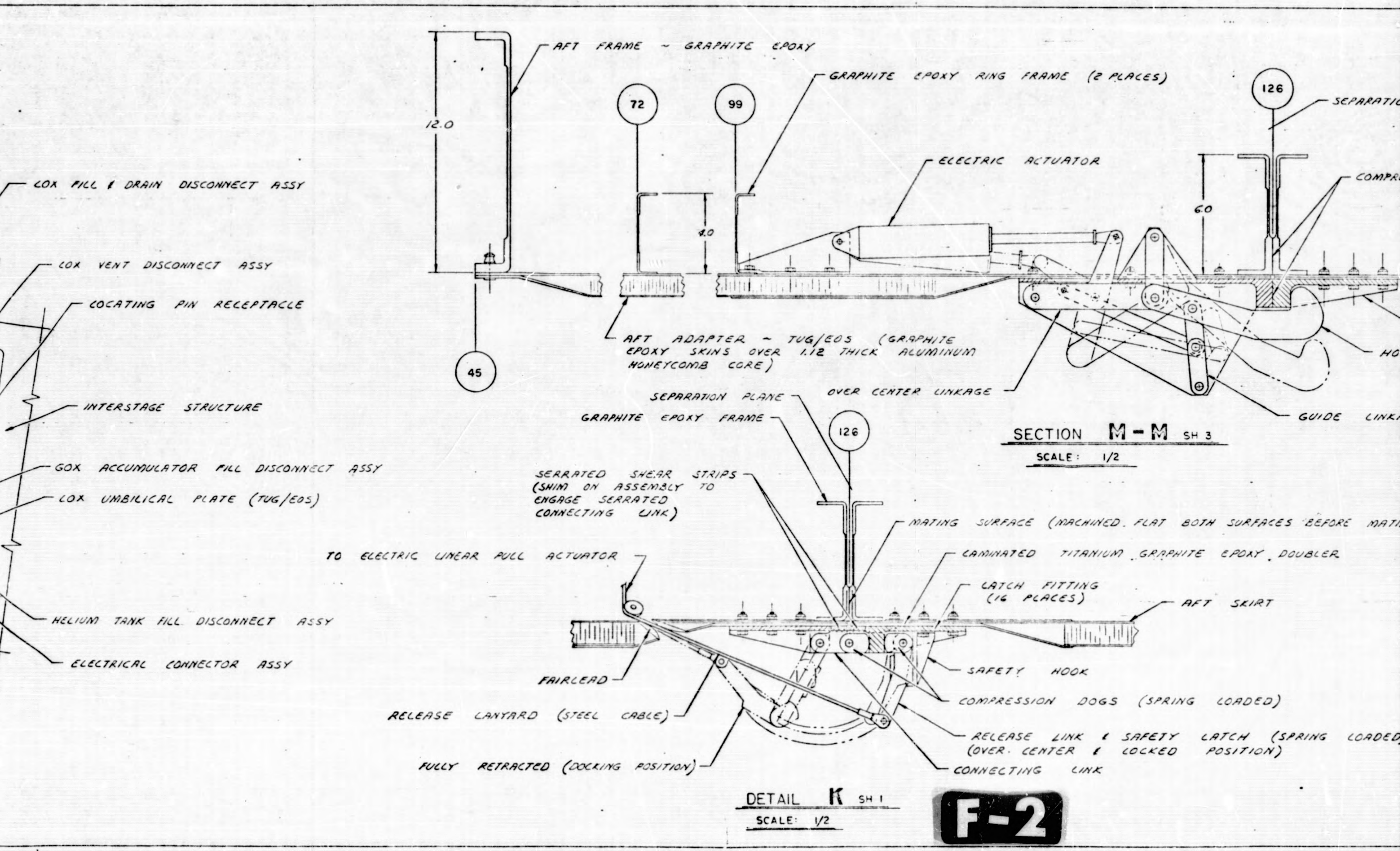
F-4

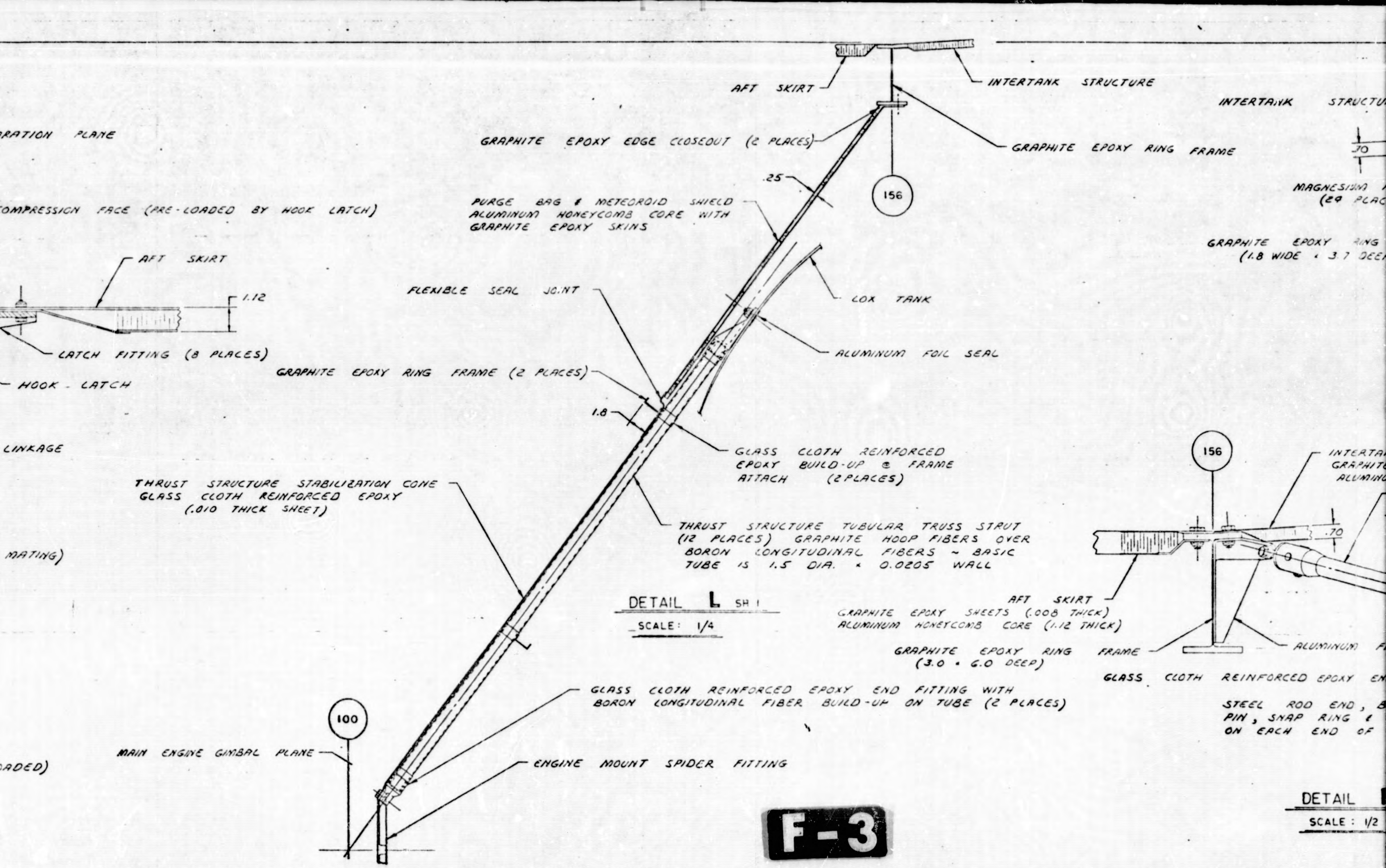


VIEW X - X SH 3

SCALE: 1/4

F-1


AFT FRAME ~ GRAPHITE EPOXY
GRAPHITE EPOXY RING FRAME (2 PLACES)
126
SEPARATIO
72
99
12.0
ELECTRIC ACTUATOR
COMPRE
6.0
LOX FILL & DRAIN DISCONNECT ASSY
4.0
LOX VENT DISCONNECT ASSY
LOCATING PIN RECEPTACLE
AFT ADAPTER ~ TUG/EOS (GRAPHITE EPOXY SKINS OVER 1.12 THICK ALUMINUM HONEYCOMB CORE)
HOO
45
OVER CENTER LINKAGE
INTERSTAGE STRUCTURE
SEPARATION PLANE
GRAPHITE EPOXY FRAME
GUIDE LINKA
126
SECTION M-M SH 3
SCALE: 1/2
GOX ACCUMULATOR FILL DISCONNECT ASSY
SERRATED SHEAR STRIPS (SHIM ON ASSEMBLY TO ENGAGE SERRATED CONNECTING LINK)
LOX UMBILICAL PLATE (TUG/EOS)
MATING SURFACE (MACHINED FLAT BOTH SURFACES BEFORE MATIN
TO ELECTRIC LINEAR PULL ACTUATOR
LAMINATED TITANIUM GRAPHITE EPOXY DOUBLER
LATCH FITTING (16 PLACES)
AFT SKIRT
HELIUM TANK FILL DISCONNECT ASSY
ELECTRICAL CONNECTOR ASSY
SAFETY HOOK
FAIRLEAD
COMPRESSION DOGS (SPRING LOADED)
RELEASE LANYARD (STEEL CABLE)
RELEASE LINK & SAFETY LATCH (SPRING LOADED) (OVER CENTER & LOCKED POSITION)
FULLY RETRACTED (DOCKING POSITION)
CONNECTING LINK
DETAIL K SH 1
SCALE: 1/2
F-2


AFT SKIRT
INTERTANK STRUCTURE
INTERTANK STRUCTU
RATION PLANE
GRAPHITE EPOXY EDGE CLOSEOUT (2 PLACES)
GRAPHITE EPOXY RING FRAME
.70
.25
156
MAGNESIUM
(29 PLAC
OMPRESSION FACE (PRE-LOADED BY HOOK LATCH)
PURGE BAG & METEOROID SHIELD
ALUMINUM HONEYCOMB CORE WITH
GRAPHITE EPOXY SKINS
GRAPHITE EPOXY RING
(1.8 WIDE × 3.7 DEE
AFT SKIRT
1.12
FLEXIBLE SEAL JOINT
LOX TANK
LATCH FITTING (8 PLACES)
ALUMINUM FOIL SEAL
GRAPHITE EPOXY RING FRAME (2 PLACES)
HOOK - LATCH
1.8
LINKAGE
GLASS CLOTH REINFORCED
EPOXY BUILD-UP @ FRAME
ATTACH (2 PLACES)
156
INTERTA
GRAPHIT
ALUMIN
THRUST STRUCTURE STABILIZATION CONE
GLASS CLOTH REINFORCED EPOXY
(.010 THICK SHEET)
THRUST STRUCTURE TUBULAR TRUSS STRUT
(12 PLACES) GRAPHITE HOOP FIBERS OVER
BORON LONGITUDINAL FIBERS ~ BASIC
TUBE IS 1.5 DIA. × 0.0205 WALL
.70
MATING)
DETAIL L SH 1
SCALE: 1/4
AFT SKIRT
GRAPHITE EPOXY SHEETS (.008 THICK)
ALUMINUM HONEYCOMB CORE (1.12 THICK)
GRAPHITE EPOXY RING FRAME
(3.0 × 6.0 DEEP)
ALUMINUM F
GLASS CLOTH REINFORCED EPOXY EN
GLASS CLOTH REINFORCED EPOXY END FITTING WITH
BORON LONGITUDINAL FIBER BUILD-UP ON TUBE (2 PLACES)
STEEL ROD END, B
PIN, SNAP RING &
ON EACH END OF
100
MAIN ENGINE GIMBAL PLANE
ENGINE MOUNT SPIDER FITTING
OADED)
DETAIL
SCALE: 1/2
F-3

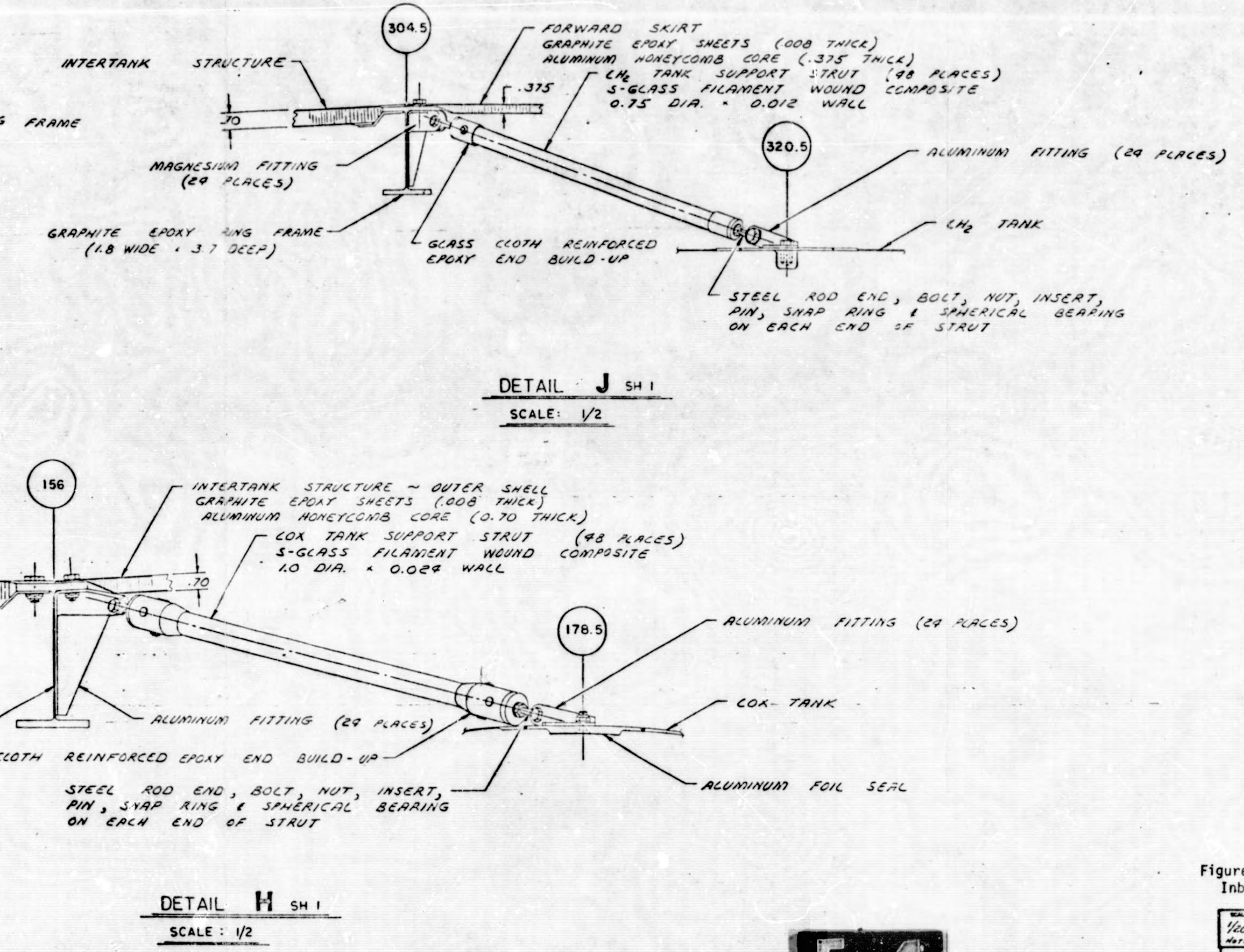


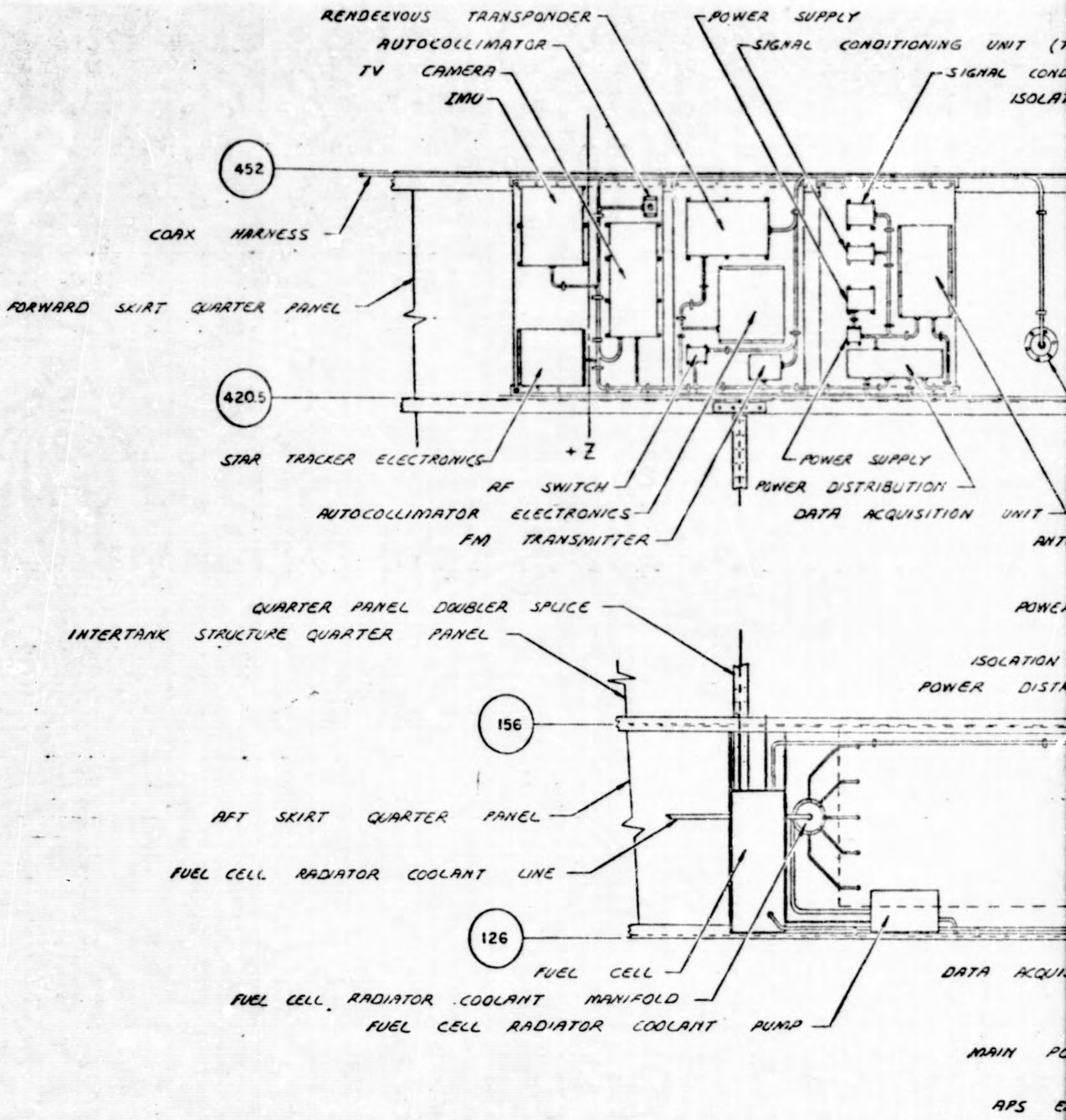
Figure 1.5-1
Inboard Profile Tug (Sheet 4 of 5)

F-4

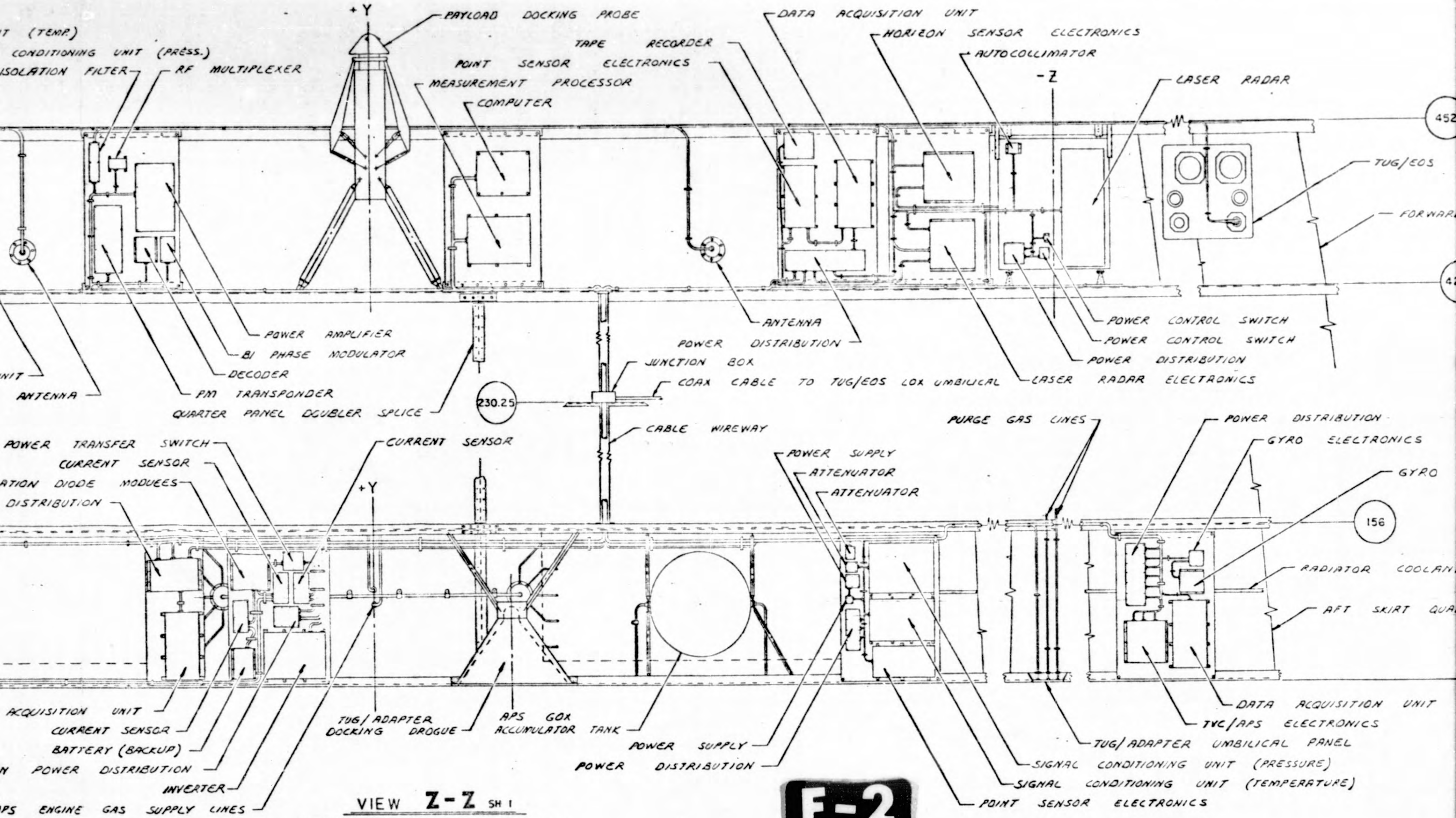
1 - 15, 1 - 16

SCALE 1/20 ✓ NOTED	DATE 1-24-72 MODEL TUG	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
INBOARD PROFILE TUG		V7-923602 SH 4 OF 5

1-15 1-16


RENDEZVOUS TRANSPONDER
AUTOCOLLIMATOR
TV CAMERA
IMU
POWER SUPPLY
SIGNAL CONDITIONING UNIT
452
COAX HARNESS
FORWARD SKIRT QUARTER PANEL
420.5
STAR TRACKER ELECTRONICS
+Z
RF SWITCH
AUTOCOLLIMATOR ELECTRONICS
FM TRANSMITTER
POWER SUPPLY
POWER DISTRIBUTION
DATA ACQUISITION UNIT
QUARTER PANEL DOUBLER SPLICE
INTERTANK STRUCTURE QUARTER PANEL
156
AFT SKIRT QUARTER PANEL
FUEL CELL RADIATOR COOLANT LINE
126
FUEL CELL
FUEL CELL RADIATOR COOLANT MANIFOLD
FUEL CELL RADIATOR COOLANT PUMP
ISOLATION
POWER
DATA

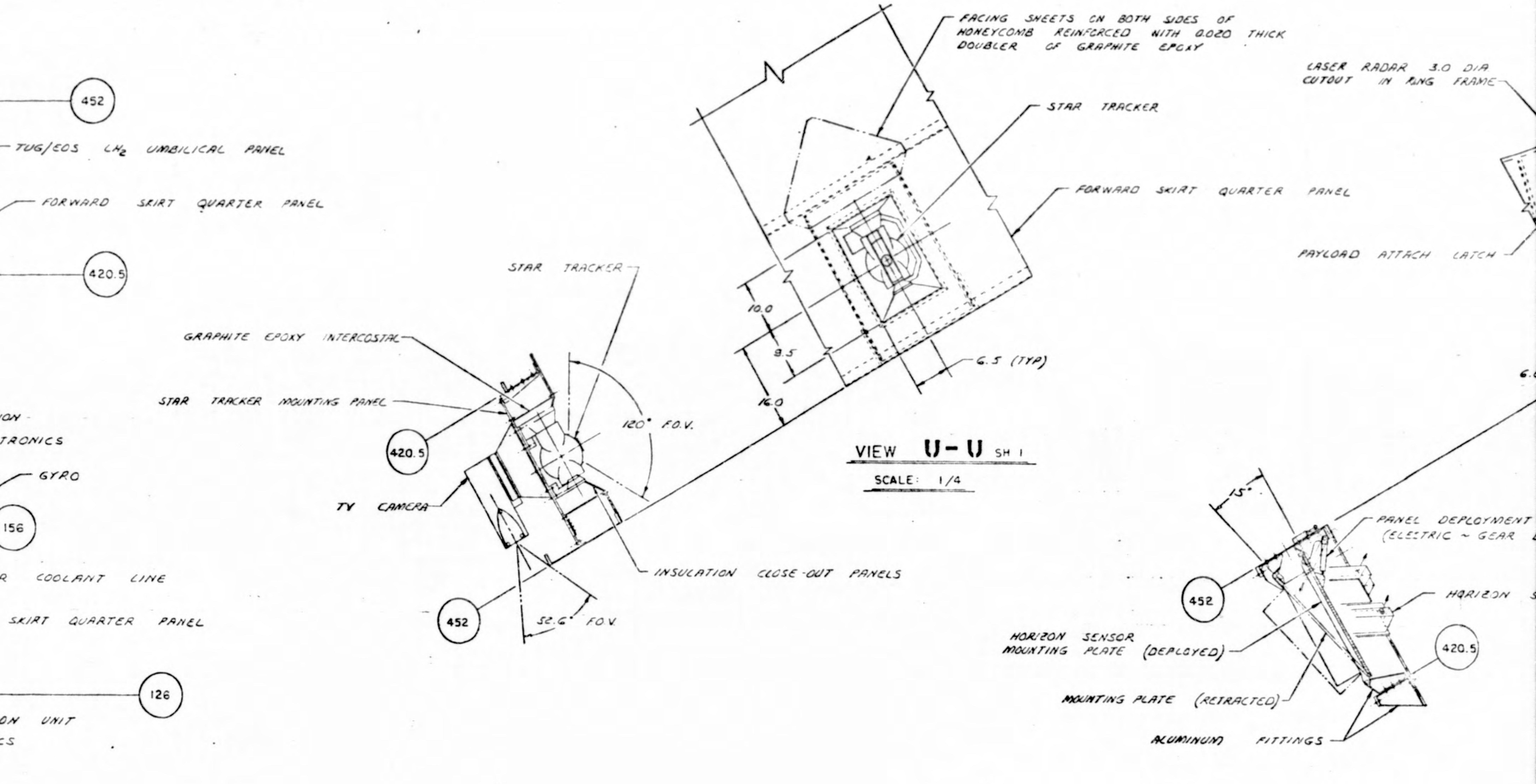
F-1



VIEW Z-Z SH 1

FORWARD & AFT SKIRT EQUIPMENT SHOWN ONLY IN FLAT PATTERN — SCALE: 1/8

F-2



FACING SHEETS ON BOTH SIDES OF HONEYCOMB REINFORCED WITH 0.020 THICK DOUBLER OF GRAPHITE EPOXY
LASER RADAR 3.0 DIA CUTOUT IN RING FRAME
452
TUG/EOS LH_2 UMBILICAL PANEL
STAR TRACKER
FORWARD SKIRT QUARTER PANEL
FORWARD SKIRT QUARTER PANEL
PAYLOAD ATTACH LATCH
420.5
STAR TRACKER
10.0
GRAPHITE EPOXY INTERCOSTAL
8.5
6.5 (TYP)
STAR TRACKER MOUNTING PANEL
16.0
120° F.O.V.
TRONICS
420.5
VIEW U-U SH 1
SCALE: 1/4
GYRO
TV CAMERA
15°
156
PANEL DEPLOYMENT
(ELECTRIC ~ GEAR
INSULATION CLOSE-OUT PANELS
COOLANT LINE
452
HORIZON S
452
52.6° F.O.V.
SKIRT QUARTER PANEL
HORIZON SENSOR MOUNTING PLATE (DEPLOYED)
420.5
126
MOUNTING PLATE (RETRACTED)
ON UNIT
ALUMINUM FITTINGS

F-3

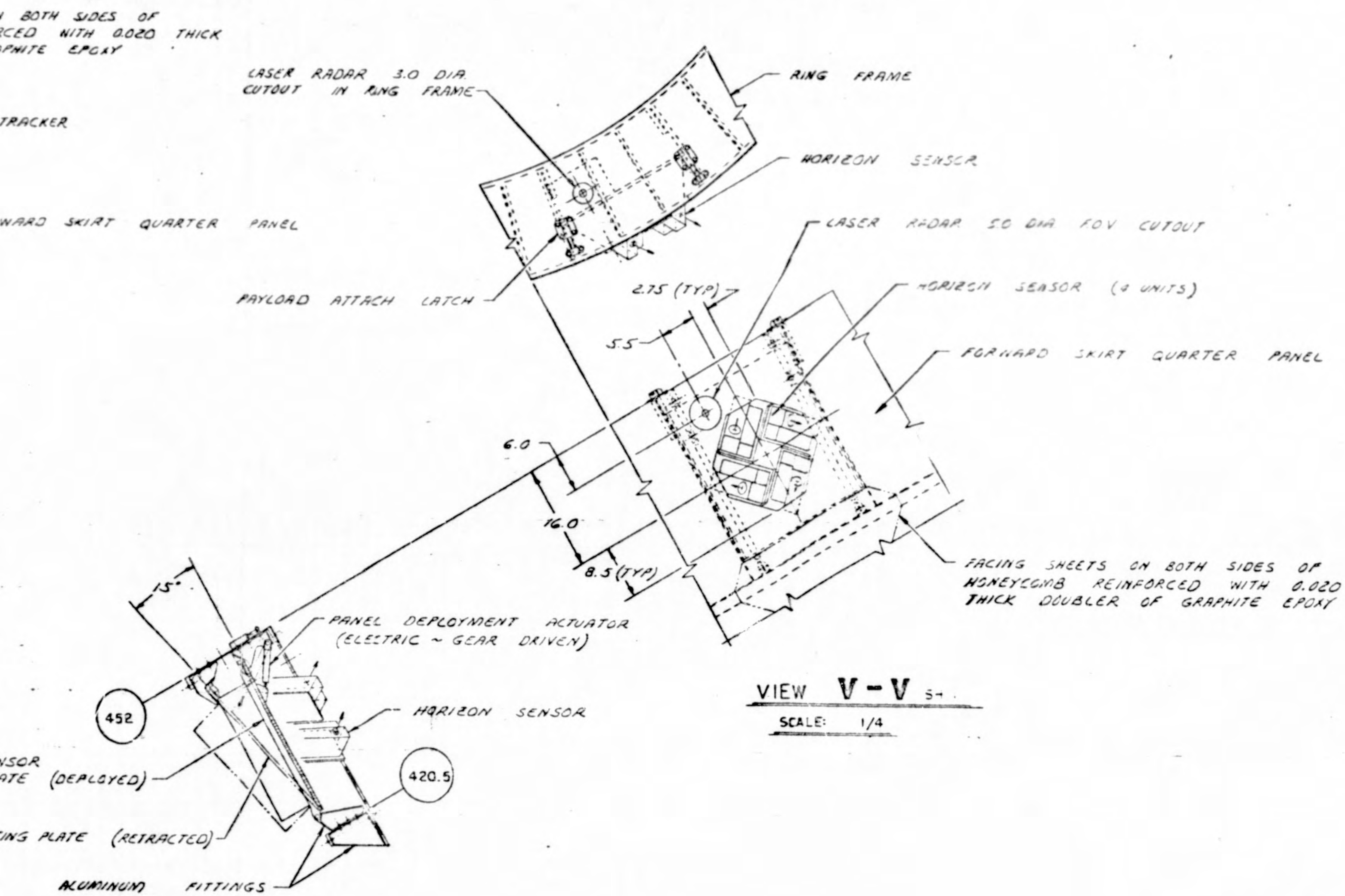


Figure 1.5-1
Inboard Profile Tug (Sheet 5 of 5)

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1 - 17, 1 - 18

SCALE 1/20 & Noted / DATE 1-24-72 / MODEL TUG	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
INBOARD PROFILE TUG		V7-923602 SH 5 OF 5

1-17 1-18

SPACE TUG

STRUCTURE (S)

LH$_2$ TANK (SHT)

1. BULKHEADS
2. SIDEWALLS
3. ACCESS DOOR
4. TANK SUPPORTS
5. BAFFLES
6. START TANK SUPPORTS

LOX TANK (SOT)

1. BULKHEADS
2. ACCESS DOOR
3. TANK SUPPORTS
4. BAFFLES
5. START TANK SUPPORTS

THRUST STRUCTURE (ST)

1. THRUST BLOCK SUPPORTS
2. THRUST BLOCK
3. SKIN PANELS
4. FRAMES
5. SUPPORT STRUTS
6. SYSTEM SUPPORT

FORWARD SKIRT (SFS)

1. FRAMES
2. SIDEWALL
3. SHUTTLE ORBITER ATTACH FITTINGS
4. SYSTEM SUPPORT

INTERTANK (SI)

1. FRAMES
2. SIDEWALL
3. ACCESS DOOR
4. SYSTEM SUPPORT

AFT SKIRT (SAS)

1. FRAMES
2. SIDEWALL
3. SYSTEM SUPPORT

DOCKING (SD)

1. LATCHING MECHANISM
2. DOCKING PROBE (PAYLOAD)

F-1

AVIONICS (A)

DATA MANAGEMENT (ADM)

1. DIGITAL COMPUTER
2. DATA ACQUISITION UNIT
3. MEASUREMENT PROCESSOR UNIT
4. INTERFACE UNIT
5. TAPE RECORDER

GUIDANCE, NAVIGATION & CONTROL (AGN)

1. INERTIAL MEASURING UNIT
2. STAR TRACKER
3. HORIZON SENSOR
4. AUTO COLLIMATOR
5. ENGINE CONTROL ASSEMBLY
6. BACKUP STABILIZATION ASSEMBLY

COMMUNICATIONS & INSTRUMENTATION (ACI)

1. TRANSPONDER - PM
2. TRANSMITTER - FM
3. POWER AMPLIFIER
4. BI-PHASE MODULATOR
5. DECODER
6. OMNI ANTENNA
7. HYBRID JUNCTION
8. DIRECTIONAL COUPLER
9. ISOLATION FILTER
10. R.F. SWITCH
11. R.F. MULTIPLEXER
12. STRAIN GAGE PRESSURE TRANSDUCER
13. POTENTIOMETRIC PRESSURE/POSITION TRANSDUCER
14. TEMPERATURE TRANSDUCER
15. CURRENT SENSOR
16. FLOW METER
17. LIQUID LEVEL POINT SENSOR
18. STRAIN GAGE SENSOR AMPLIFIER
19. TEMPERATURE SENSOR AMPLIFIER
20. CURRENT SENSOR ELECTRONICS
21. LIQUID LEVEL POINT SENSOR ELECTRONICS
22. ATTENUATOR
23. 5 VDC REGULATED POWER SUPPLY
24. ± 28 VDC POWER SUPPLY

ELEC POWER GENERATION, CONTROL & DIST (AEP)

1. FUEL CELL
   1. REACTOR STACK
   2. ISOLATION VALVE
   3. PUMP
   4. REGULATORS
   5. CONTROL VALVES
   6. CONDENSER
   7. WATER TANK
   8. CHECK VALVES
   9. VENT VALVES
   10. SCRUBBERS
   11. EJECTORS
   12. GAS SEPARATOR
2. BACKUP BATTERY
3. POWER TRANSFER SWITCH
4. POWER BUS MODULE
5. RESISTOR & DIODE MODULE
6. POWER CONTROL SWITCH

RENDEZVOUS AND DOCKING (ARD)

1. LAZER RADAR
2. TELEVISION CAMERA
3. RENDEZVOUS TRANSPONDER
4. OMNI ANTENNA
5. POWER DIVIDER

PROPULSION (P)

MAIN PROPULSION (PMP)

1. LOW SPEED INDUCER
2. TURBOPUMP
3. ISOLATION VALVE
4. PREBURNER
5. CONTROL VALVES
6. REGULATOR
7. RECEIVER
8. CHECK VALVE
9. HEAT EXCHANGER
10. ENGINE

PROPELLANT FEED FILL & DRAIN (PPF)

1. THERMAL ISOLATION SEGMENT
2. PREVALVES
3. FILL & DRAIN VALVES
4. DISCONNECTS
5. J-T EXPANSION VALVES
6. REGULATORS
7. CONTROL VALVES

PROPELLANT MANAGEMENT (PPM)

1. LEVEL SENSORS
2. SENSOR SUPPORT MAST
3. CAP PROBE

PROPELLANT ORIENTATION (PPO)

1. AUXILIARY PROPELLANT TANKS
2. PROPELLANT LEVEL SENSORS

THRUST VECTOR CONTROL (PTC)

1. PUMP MOTOR ACTUATOR
2. RESERVOIR/ACCUMULATOR ACTUATOR

PRESSURIZATION (PPR)

1. CHECK VALVE
2. REGULATORS
3. DIFFUSERS
4. DISCONNECTS
5. CONTROL VALVES

SAFING

1.
2.
3.
4.
5.
6.
7.
8.

REACTIO

1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
15.
16.

SPACE TUG

PROPULSION (P)

THERMAL (T)

ORBITER (O)

AEP)

MAIN PROPULSION (PMP)

1. LOW SPEED INDUCER
2. TURBOPUMP
3. ISOLATION VALVE
4. PREBURNER
5. CONTROL VALVES
6. REGULATOR
7. RECEIVER
8. CHECK VALVE
9. HEAT EXCHANGER
10. ENGINE

PROPELLANT FEED FILL & DRAIN (PPF)

1. THERMAL ISOLATION SEGMENT
2. PREVALVES
3. FILL & DRAIN VALVES
4. DISCONNECTS
5. J-T EXPANSION VALVES
6. REGULATORS
7. CONTROL VALVES

PROPELLANT MANAGEMENT (PPM)

1. LEVEL SENSORS
2. SENSOR SUPPORT MAST
3. CAP PROBE

PROPELLANT ORIENTATION (PPO)

1. AUXILIARY PROPELLANT TANKS
2. PROPELLANT LEVEL SENSORS

THRUST VECTOR CONTROL (PTC)

1. PUMP MOTOR ACTUATOR
2. RESERVOIR/ACCUMULATOR ACTUATOR

PRESSURIZATION (PPR)

1. CHECK VALVE
2. REGULATORS
3. DIFFUSERS
4. DISCONNECTS
5. CONTROL VALVES

SAFING & VENTING (PSV)

1. RELIEF VALVES
2. VENT VALVES
3. PRESSURE SWITCHES
4. SELECTOR VALVES
5. CONTROL VALVES
6. DISCONNECTS
7. CHECK VALVES
8. NON-PROPULSIVE NOZZLES

REACTION CONTROL (PRC)

1. BURST DISCS
2. SQUIB VALVES
3. ISOLATION VALVES
4. SELECTOR VALVES
5. CONTROL VALVES
6. REGULATORS
7. THRUSTERS
8. CHECK VALVES
9. PRESSURE SWITCHES
10. DISCONNECTS
11. NON-PROPULSIVE NOZZLES
12. GAS GENERATORS
13. TURBOPUMPS
14. HEAT EXCHANGERS
15. ACCUMULATORS
16. FLOW CONTROLLERS

FUEL CELL (TFC)

1. RADIATORS
2. PUMP
3. ISOLATION VALVE

MULTILAYER INSULATION (TMI)

1. LOX TANK
2. LH_2 TANK
3. FEED LINE
4. INSULATION SUPPORTS
5. APS TANKS

PURGE BAG (TPB)

1. VENT VALVE
2. PURGE VALVE
3. PURGE LINES
4. MANIFOLD
5. SELECTOR VALVE
6. PRESSURE SWITCH

PLUME IMPINGEMENT INSULATION (TPI)

HARDWARE CONDITIONING (THC)

1. COLD PLATES
2. HEATERS

SAFING (OS)

1. RECEIVERS
2. ISOLATION VALVE
3. DISCONNECT
4. RELIEF VALVE

INSULATION REPRESSURIZATION (OIR)

1. RELIEF VALVE
2. RECEIVERS
3. ISOLATION VALVE
4. CHECK VALVE
5. DISCONNECT
6. REGULATOR

DATA MANAGEMENT (ODM)

1. STATUS CONTROL PANEL
2. DISCONNECTS

TUG/SHUTTLE SUPPORT (OTS)

1. FRAMES
2. SIDEWALL
3. LOADS EQUILIZER MECHANISM
4. SYSTEM SUPPORT

DOCKING STRUCTURE (ODS)

1. LATCHING MECHANISM
2. DOCKING PROBE (TUG)

INTERFACE CONNECTIONS (OIC)

1. ELECTRICAL DISCONNECTS
2. FLUID DISCONNECTS
3. UMBILICAL PLATES
4. ACTUATOR MECHANISMS

Figure 1.5-2 Hardware Tree

F-2

1-27
1-28

The forward skirt accommodates eight panels of avionics equipment which is supported between the forward interface frame and the first frame aft of this. The panels are rectangular shaped 0.5 inch thick aluminum honeycomb with 0.020 inch thick aluminum skins. Two panels supporting the star tracker and communications require louvers on the back side of the panel to reject heat to the outer shell. Each panel does, however, require a heater to maintain the equipment within acceptable temperature limits during non-operating phases. These panels are attached to the shell with thermal isolation blocks which are mounted through brackets and fittings to the ring frames. The star tracker and horizon tracker panels are mounted 180° apart and are located on the Tug Z-Z axis. This axis is 30 degrees from the orbiter Z-Z axis, a condition which allowed both sensors a clear field of view with respect to the orbiter attach fittings. A cable wire way is provided at about the Tug Y-Y axis and is supported from the stability frames. In this location it is centrally located between the 8 avionics panels. In various positions within the shell structure, clips and brackets are provided on the inboard caps of the stability ring frames to attach the various lines and controls associated with the main propulsion system, auxiliary propulsion system and pressurization and vent subsystems.

A purge bag/meteoroid shield is provided at the forward end of the forward skirt. This shield is a spherically contoured diaphragm of rubber impregnated glass cloth. It is attached near the inboard cap of the 6-inch frame, aft of the interface. The diaphragm serves as both the container for the purge gas, as well as a meteoroid bumper. All of the avionics panels are forward of this barrier and consequently are accessible from the forward end of the Tug.

The forward frame provides for the support of a payload docking system. Consequently, docking and latching provisions are incorporated on this ring frame. The payload is attached to the Tug through 24 electrically actuated latches. Acquisition and docking is accomplished by the use of 3 Apollo type probes. These probes would be simplified versions of the Apollo-LEM docking probes since many of the complex operational requirements associated with the Apollo missions are not requirements for the Tug. Three off-center probes were selected since a single centrally located probe could not be installed on the Tug without greatly increasing the length of the vehicle. The probes are supported between the forward two frames and therefore extend forward of the interface plane to accomplish initial docking with the payload.

The final hard latching is accomplished by the 24 latching fingers spaced around the periphery of the forward frame. The latches are located at 15 degrees interval to assure uniform distribution of loading. The latching fingers are translated by gearing and are electrically activated. Additional redundancy may be obtained by the use of dual, parallel drive motors or by the use of an override pyrotechnic device system.

The Tug/EOS LH_2 umbilical panel is located on the forward skirt between the two forward frames. The panel is located such that when in the orbiter, the panel faces toward the lower belly corner of the orbiter. The LH_2 fill and drain line passes through a sealed cutout in the purge bag and interfaces

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with the panel. In addition, the panel accommodates pressurization and vent lines, as well as electrical connections. The various plumbing lines associated with the main propulsion, APS, and pressurization and vent system, are covered with insulation to reduce heat leaks to the main tanks and minimize boil-off.

The LH_2 tank is 168 inches in diameter with hemispherical bulkheads. A short cylindrical section (37.5 inches) separates the two bulkheads. The bulkheads are made up of six preformed gore sections and a central circular section which are all butt-welded together. These gore sections are 0.080 inch thick 2014 T651 aluminum sheet chem-milled to a thickness of 0.020 inches between the weld bands. The forward bulkhead has a 24-inch diameter opening access door located in the center of the circular section. The aft bulkhead has an identical circular section but contains a dollar weld section rather than an access door.

The short cylindrical section is made up of three 2014-T651 aluminum sheets (0.10 inch thick) which are butt welded into the cylindrical shape. The cylinder is then chem-milled to a thickness of 0.045 inch between the weld lands. At the forward end, the cylinder is butt-welded directly to the forward bulkhead. The aft end of the cylinder, however, is butt-welded to a heavier ring segment which is then butt-welded to the aft bulkhead. This heavier ring provides the thickness necessary to attach the strut truss support fittings. It also serves to adequately distrbute the loads induced into the tank from the tubular strut truss support which suspends the tank within the outer shell structure. Various bosses are welded to the tank to accommodate the tank penetrations required for feed, fill and drain, pressurization, vent and electrical lines. These penetrations have been grouped together to minimize the number of bosses within a gore section. The forward bulkhead access door contains the fill and drain penetration and also accommodates the vent deflector, which is an integral part of the door.

The APS LH_2 auxiliary tank is mounted inside the main LH_2 tank on the aft bulkhead. Its' centerline is located 3 inches from the center of the tank and is supported 1 inch from the bulkhead. It is supported on 6 lugs which are welded to the gore section weld lands. These lugs are separated 1.375 inches so as not to interfere with the inspection of the gore section welds. Other clips and lugs are similarly welded within the tank on the weld lands to provide attach points for the settling baffle support struts and the point sensor stillwell support guy wires.

The LH_2 tank is completely covered with multi-layer insulation. The insulation can be divided into four sections: the forward access door segment, the aft dollar weld segment, the aft bulkhead segment (up to the support struts) and the forward bulkhead and cylindrical section. The forward and aft bulkhead and cylindrical section are made up of 24 pie-shaped slices which are staggered when installed on the tank. The forward section covers the forward bulkhead, the cylindrical section, and contains the penetrations for the outer shell/tank support struts. The aft section covers only the aft bulkhead. The dollar weld section contains the penetrations for the feed, vent, APS fill and feed and the electrical lines, and is

configured as a circular section. The MLI is spaced from the tank wall, a distance of one inch by foam filled honeycomb pads which are bonded directly to the tank wall. An aluminum wire mesh internal support membrane is then fastened to the pads through threaded inserts in the pads. This wire mesh is stretch formed to the correct contour of the tank. The MLI layers are then attached with fiberglass epoxy tube posts which are installed to the pads through threaded inserts (not the same as used for the wire mesh). An external tension membrane of Nomex mesh is then applied over the entire surface to prevent balooning of the insulation. This membrane is applied in quadrants for the forward and aft sections, and a band for the area where the struts penetrate the MLI.

Twenty-four aluminum fittings are attached to the LH_2 tank outer wall to accommodate the support struts. The S-glass filament wound composite tubular struts suspend the tank within the shell. There are a total of 48 struts in the support truss. The ends of the tubes are reinforced with glass cloth epoxy build-up layers and are attached to the tank fitting through steel rod ends which are adjustable. The rod ends are mounted through spherical bearings to the fittings. The tubes are 0.75 inch in diameter with a 0.012 inch wall. At the outer shell they are attached with identical mounting at the tank end to magnesium fittings which are attached to a graphite epoxy composite "I" beam frame at the forward end of the intertank structure. The forward skirt is attached at this ame frame with "Hi Lok" fasteners. The "I" beam frame thus functions as the field/manufacturing joint interface between the forward skirt and the intertank shell.

The intertank shell structure is a truncated cone of 180 inches in diameter at the forward end and 162 inches in diameter at the aft end. It is 148.5 inches long. The construction is identical to that of the forward skirt (aluminum honeycomb core with graphite epoxy skins). The shell is 0.70 inches thick and in addition to the "I" beam frame at the forward end, incorporates three channel section ring frames of graphite epoxy which are equally spaced and measure 0.5 inch by 1.75 inches deep. As in the forward skirt, these frames are primarily for stability and in addition provide for mounting of clips and brackets to support the various propulsion system plumbing lines and controls, and the electrical cable wire way and junction box.

Several penetrations of this shell are accommodated for the Shuttle/Tug LOX umbilical, purge venting and an access door to allow servicing of the intertank area. The access door is located between the two aft ring frames and is a 30-inch square opening. Two six-inch diameter opening vent valves are located 180 degrees apart in the forward area of the shell. These valves vent the purged intertank area to allow rapid evacuation of the MLI during orbiter ascent. The LOX umbilical panel is located between the two forward stability frames and is oriented such that it faces the belly or lower portion of the orbiter when the Tug is installed in the cargo bay.

At the aft end of the shell, located 90 degrees apart are four 30 x 48-inch panels of dynaflex R-F 2400 insulation covered with 0.002 inch thick stainless steel. These panels are to accommodate the forward firing

APS engine plume impingement in this area. The panels are attached to the outer surface of the shell. As in the forward skirt the intertank shell is made in quarter panels and the splicing of these panels is identical to that described for the forward skirt. The intertank structural shell is attached at its aft end to the aft skirt, with "HI Lok" fasteners. It is attached to a graphite composite "I" beam which is part of the aft skirt.

The aft skirt is a cylindrical shell assembly 162 inches in diameter and 30 inches long. It is identical in construction to the other outer shell assemblies but is 1.12 inches thick. It is made up of quarter panels of graphite epoxy skins over aluminum honeycomb core. There are no intermediate or stability frames in this skirt, only the forward "I" beam frame and the aft channel frame. The aft frame mates with an identical counterpart on the aft adapter, which supports the Tug within the orbiter cargo bay. This skirt also accommodates the mounting of three docking drogues for the Tug/Shuttle adapter docking. These drogues are supported by an intercostal to the aft frame and stabilized by a tubular strut truss between the forward and aft frame. Two APS gas accumulator tanks and an engine purge system helium supply tank are mounted on the aft skirt with simple girth attached "VEE" strutt trusses with a torsional stability strut. These tubular struts are also attached to the frames. Four avionics panels, identical to those in the forward skirt, are also mounted between the frames in the aft skirt. None of these panels require louvers but all have heaters to maintain temperatures within acceptable limits during non-operating phases of the missions. The avionics equipment located in the aft skirt is that which is propellant or propulsion oriented. The aft frame interface is also the purge system disconnect panel between the Shuttle mounted adapter and the Tug. A fuel cell and associated coolant pump and controls are mounted on intercostals between the two frames. The coolant is manifolded to four radiator panels located on the aft skirt outer skin 90 degrees apart. These radiator panels are connected in series to maintain a constant temperature to the fuel cell regardless of the orientation of the Tug with respect to the sun.

Two five-engine APS modules are mounted on the outer surface of the aft skirt between the radiator panels. The two quints are 180 degrees apart and between them are located two dual engine modules. The quints are located on the Tug Z-Z axis while the duads are located on the Y-Y axis. The duads have forward and aft firing engines. The quints are composed of conventional type quad firing engine patterns with the addition of a single engine firing radially outward. Various clips and support brackets are provided by the aft skirt for attaching and supporting plumbing lines, cables and an electrical cable wire way and junction box. The aft skirt contains 24 latch receptacle fittings at the aft end to mate the Tug with the Shuttle mounted adapter latching system. Twenty-four aluminum fittings at 15 degrees interval, are attached to the forward frame to provide attachment of the LOX tank support struts.

Forty-eight tubular struts attach the LOX tank to the outer shell. These struts are S-glass filament wound tubes which are identical to those described for the LH_2 tank except for the larger diameter (1 inch) and wall thickness (0.024 inches). The ends of the struts contain adjustable rod ends and

spherical bearings. At the LOX tank the struts are attached to aluminum fittings at the girth of the tank.

The LOX tank is an oblate spheroid 150 inches in diameter and 105 inches high and is manufactured from 2014-T651 aluminum. The tank is made up of the two bulkheads and a girth ring which are all butt-welded together. Each bulkhead is composed of six preformed gore sections. There is a large diameter (34 inches) dollar weld on the aft bulkhead and a small diameter (14 inch) dollar weld on the forward bulkhead. The 6-inch girth ring is 0.30 inches thick. Adjacent to this on the forward bulkhead the tank thickness tapers from 0.125 to 0.040 inches which is the basic thickness for this bulkhead. The aft bulkhead is basically 0.030 inches thick but tapers to 0.125 inches thick in the area where the thrust structure is attached.

The penetrations to the tank are grouped into one area on the aft bulkhead and all bosses are located on a circular thickened section in one gore section. In the forward bulkhead, the fill and drain inlet is a similarly handled penetration. An access door is located off center in the forward bulkhead and 30 degrees from the Tug + Z axis. This door offers a 20-inch opening into the LOX tank.

An APS LOX auxiliary tank is located on the aft bulkhead within the main tank. This tank is similar to that in the LH_2 tank. It is put into the tank in halves through the opening in the forward bulkhead. It is attached to the aft bulkhead by lugs which are welded to the gore section weld lands and the tank is mounted one inch from the inside wall of the bulkhead. Other tabs welded to the weld lands and thickened sections allow attachment and support points for the settling baffles (three baffles) and the sensor probe stillwell guide cables.

The LOX tank, like the LH_2 tank, is completely covered with MLI. This insulation is applied in 24 gore segments in a staggered configuration. The MLI is divided into the forward bulkhead section, the aft bulkhead section, and a dollar section for the forward and aft apex. The forward section extends beyond the tank support struts and consequently contains the penetrations and seals for these struts. These MLI panels are applied in exactly the same manner as those described for the LH_2 tank (foam filled honeycomb hardspots, aluminum wire mesh, 0.50 inch thick MLI and an external tension membrane). In the area of the access door, special provisions have been made to remove the insulation and screen mesh to permit opening of the tank. Twelve penetrations of the aft bulkhead insulation accommodate the thrust structure struts. The one-inch gap between the tank wall and the insulation serves as the purge gas annulus for the insulation and is purged through an aluminum tube manifold at the girth of the tank. This manifold distributes the gas under the MLI. This concept is identical to that used on the LH_2 tank.

The thrust structure is attached directly to the aft bulkhead on the LOX tank. The primary load carrying structure is composed of the 12 tubular struts which attach the engine gimbal block mount to the LOX tank. These struts are 1.5 inches in diameter and are made up of a combination of boron epoxy longitudinal fibers and graphite epoxy circumferential fibers. The

struts are 60 inches long with built-up ends to accommodate the end fittings for attachments. A glass cloth reinforced epoxy bathtub type fitting on the aft end attaches to the engine gimbal block mount aluminum spider fitting with "Hi Lok" fasteners. At the forward end, the tube terminates in another glass cloth reinforced epoxy bathtub type fitting which attaches directly to the LOX tank, at a tangency point. The thrust structure tubes are flattened on two sides to accommodate the attachment of ring frames and a truncated torsional stability cone.

The stability cone is a glass cloth reinforced epoxy skirt 0.010 inch thick. It is attached to the flat areas on the struts. Two graphite epoxy channel ring frames are attached to the thrust structure with blind pull rivets. One frame is attached to the inner portion of the cone and one to the outer surface. The tubes are built up in the areas where these frames are attached. Brackets and tabs on the frames and the cone structure provide for mounting of APS equipment. Two APS conditioning units are attached to the thrust structure, as well as controls, plumbing and electrical lines. Two of the struts provide attach fittings for the main engine gimbal actuators. The thrust structure also provides aft meteoroid protection for this vehicle, as well as serving as the aft purge bag. To continue the meteoroid protection/purge bag, an additional truncated cone of graphite epoxy skins (0.008 inches thick) over aluminum honeycomb core (0.25 inches thick) is provided between the thrust structure and the outer shell. This cone is attached with "Hi Lok" fasteners to the aft skirt forward frame and is joined to the thrust structure at the forward channel frame through a flexible seal, which does not allow this structure to carry thrust loads.

The feed lines to the main engine are covered with multi-layer insulation as are the fill and drain lines for the main tanks. This insulation is applied directly to the lines and does not provide a purge annulus. The vent lines and APS feed lines are also insulated. The APS accumulator tanks mounted on the aft skirt are also covered with MLI. This insulation is applied directly to the tank. A thermal coating is applied to the aft facing surface of this insulation. All of the insulation on the plumbing lines and on the accumulator tanks is held in place with zippered tension membranes similar to that on the main tanks.

The Tug purge bag consists of the outer shell structure, the forward spherically shaped membrane, the thrust structure, and the small closeout between the thrust structure and the outer shell. To insure proper sealing of this structure/purge bag and reduce leakage of the purge gas, the entire inner surface of the purge bag is covered with a sealant coating of RTC epoxy resin. The large area around and between the main tanks is purged by the same system that purges the main tanks and the insulation. Gas diffusers are located in the intertank area for this purpose. The entire outer surface of the Tug vehicle is covered with a thermal control coating of zinc oxide in RTV methyl silicone. This coating is applied to the outer surface of the outer shell, the forward meteoroid shield/purge bag outer surface, the outer surface of the thrust structure and the inner surface of the aft skirt.

The Tug is attached inside the orbiter by a cylindrical shell Shuttle mounted adapter. The Tug is upside down when installed in the orbiter cargo bay, and is supported from the forward bulkhead. This adapter is 162 inches in diameter to mate with the aft skirt and is 81 inches long. It is assembled from four panels which are made up of aluminum honeycomb core with graphite epoxy composite face sheets. The honeycomb core is 1.12 inches thick and the facing sheets are 0.058 inches thick in the major load carrying areas and 0.008 inches thick in the lower stressed areas.

The shell is reinforced by four equally-spaced channel section, graphite epoxy ring frames. The aft frame is 12 inches deep and the intermediate frames are 4 inches deep each. The shell is comprised of two panels which subtend an angle of 65.4 degrees each and two which subtend an angle of 114.6 degrees each. Titanium is used for fittings and doublers for assembling the cylinder.

The adapter provides the main structural attachment for the Tug to the orbiter cargo bay and contains Tug deployment and docking provisions. Once the Tug is separated from the orbiter, this adapter structure remains with the orbiter. The launch loads are transmitted to the adapter structure through two interconnected, load equalizing hydraulic struts which interface with the adapter at two trunnions located on the aft end of the adapter and slightly aft of the aft frame. Pitch loads are introduced tangentially through these fittings and yaw loads are introduced tangentially through a fitting at the bottom or belly portion of the cylinder.

The adapter houses the helium purge system for the main tanks and insulation. Eight 6-cubic feet tanks are mounted near the aft end of the adapter for the helium storage. Various controls and plumbing are accomodated, as well as a disconnect panel which interfaces with the Tug through the aft skirt disconnect. For deployment of the Tug vehicle out of the orbiter cargo bay, a shaft located near the upper forward corner of the orbiter cargo bay is inserted into receiver fittings attached to the adapter. The adapter and Tug combination is then rotated out of the cargo bay about this shaft.

At the Tug/adapter interface, 24 latches are located at 15 degree interval around the perimeter of the adapter. Sixteen of these latches are folded away after initial deployment, leaving only the 8 independently actuated latches which are used for redocking and reattachment of the Tug for reentry, when the Tug is empty and the loads are consequently lower. The sixteen latches are compression dogs with an overcenter link which are initially locked on the ground and are released in space with a pull cable system. The eight remateable latches are hook latch type with overcenter linkage and are electrically driven.

Once the Tug mission is completed, docking of the Tug to the adapter is accomplished by three probes in the adapter. When extended, the probe arms are free to deflect laterally under slight pressure. The probes engage drogures in the aft skirt and are then retracted to draw the Tug to the

adapter. Spherical bearings and an axial alignment system within the probes insure that the Tug is aligned correctly with the adapter for engagement of the eight hook latches. When the Tug/adapter interface has been made, the eight latches are actuated and the final attachment is accomplished. The Tug/adapter combination is then rotated back into the orbiter cargo bay, the trunnion attachment fittings are engaged and the deployment shaft removed from the fittings on the adapter.

1.6 HARDWARE TREE

The Hardware Tree shown in Figure 1.5.2 is a summary of all the components comprising the four basic systems of the Tug, with the exception of lines, wiring and component mounting hardware. The tree also lists as a separate category those items which are chargeable to the Tug but remain in the Shuttle orbiter.

The tree outlines the level of hardware detail investigated in the design analysis as well as the breakdown employed.

The design definition studies are presented in two volumes. This volume includes the definition of Propulsion and Mechanical Subsystems; Avionics Subsystems; Thermal Control for TVC hydraulics and fuel cell; and Electrical Power Subsystem.

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2.0 PROPULSION AND MECHANICAL SUBSYSTEMS

This section of the report will present the function, options considered, system and operational description of the following subsystems:

Propellant Feed, Fill and Drain
Safing and Venting
Pressurization
Propellant Acquisition
Propellant Management
Main Propulsion (MPS)
Thrust Vector Control (TVC)
Auxiliary Propulsion (APS)

The individual requirements for these subsystems as well as the fuel cell have been integrated where practical. Figure 2.0-1 shows schematically how this has been accomplished. Table 2.0-1 is a list of all components (except for orifices and lines) which are used in the mechanical and propulsion subsystems. The fuel cell and insulation repressurization subsystems have been included on both the schematic and list of components because of their direct tie-in to the mechanical subsystems. The subsystem discussions, in most cases, employ separate schematics on which the components have been numbered as an aid in understanding the system operational descriptions. The integrated numbering system shown in Figure 2.0-1 has not been used in the individual subsystem sections in order to simplify the callouts on the schematics.

A discussion of electrically versus pneumatically operated valves will be presented at the beginning of this section. The choice of electrically operated valves will affect the power requirements of the various subsystems.

A hardware and operational description of the Tug/Orbiter interfaces is also included.

Table 2.0-1. Propulsion and Mechanical Subsystems Components

PMP - MAIN PROPULSION	
1.1	LOX Low Speed Inducer
1.2	LH_2 Low Speed Inducer
2.1	LOX Turbopump
2.2	LH_2 Turbopump
3.1	Helium Purge Isolation Valve(s)
3.2	LH_2 Main Engine Isolation Valve(s)
4.	Preburner
5.1	Thrust Control Valve (M.O.)

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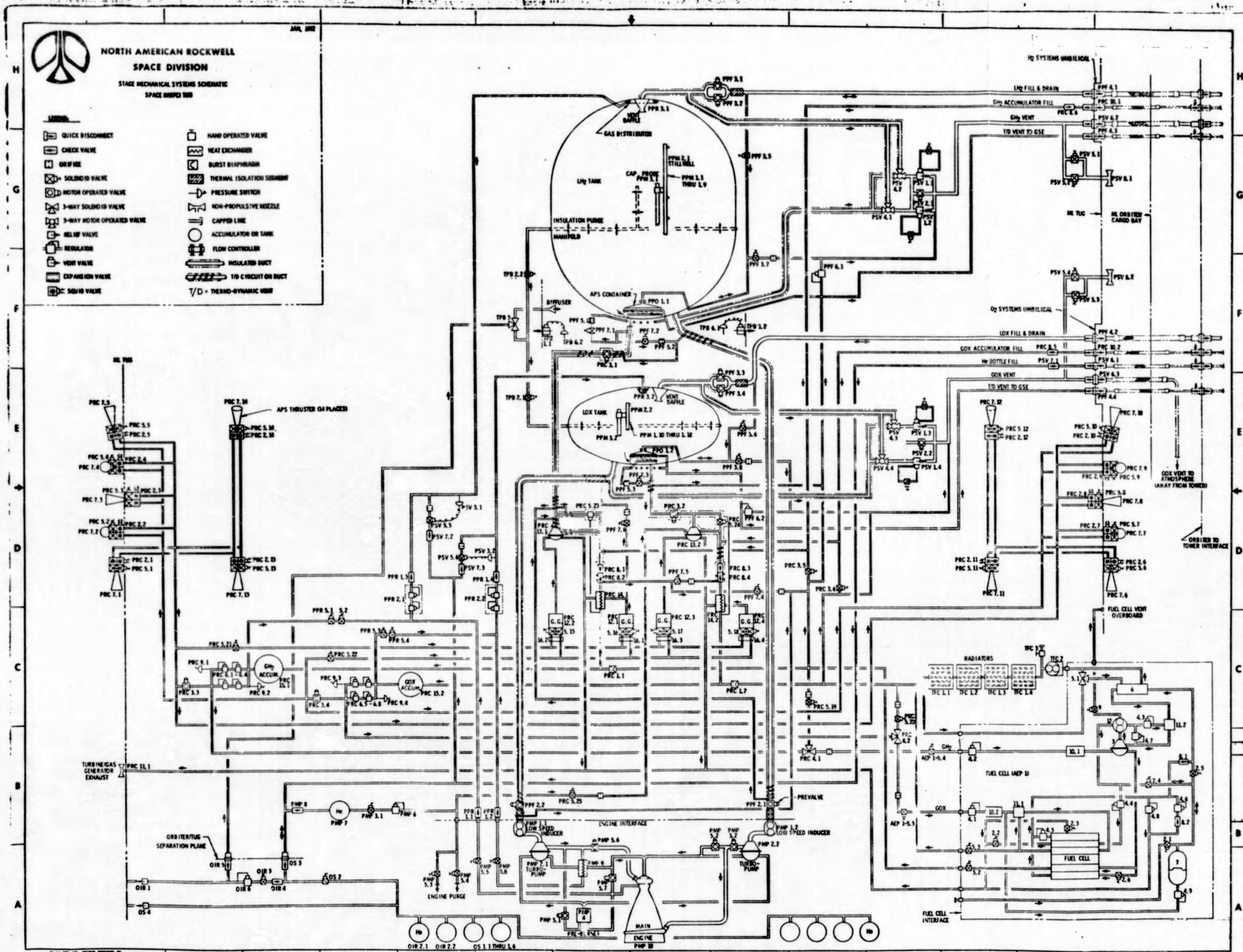

Figure 2.0-1 Stage Mechanical Systems Schematic Space (MSFC) Tug

Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

PMP - MAIN PROPULSION (continued)

5.2	Start By-Pass Valve (M.O.)
5.3	Purge Control Valve(s)
5.4	Purge Control Valve(s)
5.5	LH_2 Tank Pressurization Control Valve(s)
5.6	Mixture Ratio Control Valve (M.O.)
5.7	Turbine Temp Control Valve (M.O.)
5.8	LOX Tank Pressurization Control Valve(s)
6.	Helium Purge Regulator
7.	Helium Purge Receiver
8.	Helium Purge Receiver Fill Check Valve
9	LOX Heat Exchanger
10.	Engine

PPF - PROPELLANT FEED, FILL AND DRAIN

1.1	LH_2 Feed Line Thermal Isolator
1.2	LOX Feed Line Thermal Isolator
1.3	LH_2 Fill and Drain Line Thermal Isolator
1.4	LOX Fill and Drain Line Thermal Isolator
2.1	LH_2 Prevalve (M.O.)
2.2	LOX Prevalve (M.O.)
3.1	LH_2 Fill and Drain Valve (M.O.)
3.2	LH_2 Fill and Drain Valve (M.O.)
3.3	LOX Fill and Drain Valve (M.O.)
3.4	LOX Fill and Drain Valve (M.O.)
3.5	LH_2 Aux Prop Tank Fill Valve(s)
3.6	LOX Aux Prop Tank Fill Valve(s)
3.7	LH_2 Aux Prop Tank Vent Control Valve(s)
3.8	LOX Aux Prop Tank Vent Control Valve(s)
4.1	LH_2 Fill and Drain Disconnect
4.2	LOX Fill and Drain Disconnect
4.3	LH_2 T/D Vent Disconnect
4.4	LOX T/D Vent Disconnect
5.1	LH_2 Aux Prop Tank T/D Vent *J-T Valve*
5.2	LH_2 MPS and APS Feed Line T/D Vent *J-T Valve*
5.3	LOX Aux Prop Tank T/D Vent J-T Valve
6.1	LH_2 T/D Vent Back-Pressure Regulator
6.2	LOX T/D Vent Back-Pressure Regulator
7.1	LH_2 Aux Prop Tank T/D Vent Control Valve(s)
7.2	LH_2 MPS and APS Feed Line T/D Vent Control Valve(s)
7.3	LOX Aux. Prop Tank T/D Vent Control Valve(s)
7.4	LH_2 MPS and APS Feed Line T/D Vent Outlet Control Valve(s)
7.5	LOX APS Feed Line T/D Vent Control Valve(s)
7.6	LOX MPS Feed Line T/D Vent Control Valve(s)

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Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

PPM - PROPELLANT MANAGEMENT	
1.1 - 1.9	LH_2 Tank Level Sensors
1.10 - 1.18	LOX Tank Level Sensors
2.1	LH_2 Tank Sensor Mast (Stillwell)
2.2	LOX Tank Sensor Mast (Stillwell)
3.1	LH_2 Tank Cap Probe
3.2	LOX Tank Cap Probe
PPO - PROPELLANT ORIENTATION (ACQUISITION)	
1.1	LH_2 Auxiliary Propellant Tank
1.2	LOX Auxiliary Propellant Tank
2.1	LH_2 Aux. Prop. Tank Level Sensor
2.2	LH_2 Aux. Prop. Tank Level Sensor
2.3	LOX Aux. Prop. Tank Level Sensor
2.4	LOX Aux. Prop. Tank Level Sensor
PTC - THRUST VECTOR CONTROL	
1.	Pump/Motor Actuator
2.	Reservoir/Accumulator Actuator
PPR - PRESSURIZATION	
1.1	LH_2 Press. Engine Isolation Check Valve
1.2	LOX Press. Engine Isolation Check Valve
1.3	LH_2 Tank Pressure Regulator Isolation Check Valve
1.4	LOX Tank Pressure Regulator Isolation Check Valve
2.1	LH_2 Tank Pressure Regulator
2.2	LOX Tank Pressure Regulator
3.1	LH_2 Tank Pressurization Gas Diffuser
3.2	LOX Tank Pressurization Gas Diffuser
4.	Deleted Disconnects
5.1	LH_2 Tank Prepressurization Control Valve(s)
5.2	LH_2 Tank Prepressurization Control Valve(s)
5.3	LOX Tank Prepressurization Control Valve(s)
5.4	LOX Tank Prepressurization Control Valve(s)
PSV - SAFING AND VENTING	
1.1	LH_2 Tank Relief Valve
1.2	LH_2 Tank Relief Valve
1.3	LOX Tank Relief Valve
1.4	LOX Tank Relief Valve
2.1	LH_2 Tank Vent Valve (M.O.)
2.2	LOX Tank Vent Valve (M.O.)
3.1	LH_2 Helium Purge Pressure Switch
3.2	LOX Helium Purge Pressure Switch

Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

PSV - SAFING AND VENTING (continued)	
4.1	LH_2 Tank Vent Selector Valve (M.O.)
4.2	LH_2 Tank Vent Selector Valve (M.O.)
4.3	LOX Tank Vent Selector Valve (M.O.)
4.4	LOX Tank Vent Selector Valve (M.O.)
5.1	GH_2 Non-Propulsive Vent Control Valve(s)
5.2	GH_2 Non-Propulsive Vent Control Valve(s)
5.3	GOX Non-Propulsive Vent Control Valve(s)
5.4	GOX Non-Propulsive Vent Control Valve(s)
5.5	LH_2 Tank Helium Purge Control Valve(s)
5.6	LOX Tank Helium Purge Control Valve(s)
6.1	Helium Purge Receiver Fill Disconnect
6.2	GH_2 Vent Disconnect
6.3	GOX Vent Disconnect
7.1	Helium Purge Receiver Fill Check Valve
7.2	LH_2 Tank Helium Purge Check Valve
7.3	LOX Tank Helium Purge Check Valve
8.1	GH_2 Non-Propulsive Vent
8.2	GOX Non-Propulsive Vent
OS - SAFING (ORBITER)	
1.1 - 1.6	Helium Purge Receivers
2.	Helium Purge Isolation Valve(s)
3.	Helium Purge Disconnect
4.	Helium Purge Relief Valve
PRC - REACTION CONTROL (AUXILIARY PROPULSION)	
1.1	GH_2 Burst Disc
1.2	GOX Burst Disc
2.1 - 2.14	Thruster Squib Valve
3.1	LH_2 Feed Isolation Valve (M.O.)
3.2	LOX Feed Isolation Valve (M.O.)
3.3	GH_2 Isolation Valve (M.O.)
3.4	GOX Isolation Valve (M.O.)
3.5	LH_2 System Bleed Valve(s)
3.6	LOX System Bleed Valve(s)
4.1	GH_2 Fuel Cell/Vent Selector Valve(s)
4.2	GOX Fuel Cell/Vent Selector Valve(s)
5.1 - 5.14	Thruster Bi-Propellant Valve(s)
5.15	LH_2 Turbopump Gas Generator Bi-Propellant Valve(s)
5.16	LH_2 Heat Exchanger Gas Generator Bi-Propellant Valve(s)
5.17	LOX Turbopump Gas Generator Bi-Propellant Valve(s)
5.18	LOX Heat Exchanger Gas Generator Bi-Propellant Valve(s)
5.19	GH_2 Vent Valve(s)
5.20	GOX Vent Valve(s)

Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

PRC - REACTION CONTROL (AUXILIARY PROPULSION) (continued)

5.21	Gas Generator GH_2 System Control Valve(s)
5.22	Gas Generator GOX System Control Valve(s)
5.23	LH_2 Turbopump By-Pass Control Valve
5.24	LOX Turbopump By-Pass Control Valve
5.25	LOX Turbopump Purge Valve(s)
6.1 - 6.4	GH_2 Pressure Regulator
6.5 - 6.8	GOX Pressure Regulator
7.1 - 7.14	Thrusters
8.1 - 8.2	GH_2 Isolation Check Valve
8.3 - 8.4	GOX Isolation Check Valve
8.5	GOX Accumulator Fill Check Valve
8.6	GH_2 Accumulator Fill Check Valve
9.1	GH_2 Regulator Output Pressure Switch
9.2	GH_2 Accumulator Pressure Switch
9.3	GOX Regulator Output Pressure Switch
9.4	GOX Accumulator Pressure Switch
10.1	GH_2 Accumulator Fill Disconnect
10.2	GOX Accumulator Fill Disconnect
11.	Non-Propulsive Exhaust
12.1	LH_2 Heat Exchanger Gas Generator
12.2	LH_2 Turbopump Gas Generator
12.3	LOX Turbopump Gas Generator
12.4	LOX Heat Exchanger Gas Generator
13.1	LH_2 Turbopump
13.2	LOX Turbopump
14.1	LH_2 Heat Exchanger
14.2	LOX Heat Exchanger
15.1	GH_2 Accumulator
15.2	GOX Accumulator
16.1	LH_2 Heat Exchanger Gas Generator Flow Controller
16.2	LH_2 Turbopump Gas Generator Flow Controller
16.3	LOX Turbopump Gas Generator Flow Controller
16.4	LOX Heat Exchanger Gas Generator Flow Controller

AEPI - FUEL CELL

1.	Reactor Stack
2.1	Main Water Fill Valve(s)
2.2	Water Purge Valve(s)
2.3	Steam Purge Valve(s)
2.4	Water Storage Isolation Valve(s)
2.5	Secondary Water Fill Valve(s)
2.6	By-Product Water Chamber Purge Valve(s)
3.	Water Pump
4.1	GOX Inlet Pressure Regulator
4.2	GH_2 Inlet Pressure Regulator

Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

AEPI - FUEL CELL (continued)

4.3	Condenser Pressure Regulator
4.4	Water/Steam ΔP Regulator
4.5	Steam Pressure Regulator
4.6	Steam Vent Pressure Regulator
4.7	Water Pump By-Pass Control Regulator
4.8	Water Vapor Vent Regulator
4.9	Water Storage Pressure Regulator
5.1	Condenser Temp. Control Valve(s)
5.2	GH_2 Purge Control Valve(s)
5.3	GOX Purge Control Valve(s)
5.4	GH_2 Reactant Control Valve(s)
5.5	GOX Reactant Control Valve(s)
6.	Condenser/Subcooler
7.	Water Storage
8.1	Coolant Loop Check Valve
8.2	Steam Vent Check Valve
9.	Gas Separation Vent Valve(s)
10.1	GH_2 Scrubber
10.2	GOX Scrubber
11.1	GOX Recycle Ejector
11.2	Water Ejector
12.	Gas Separator

TFC - FUEL CELL

1.1 - 1.4	Radiator
2.	Pump
3.	Freon Fill Valve(H)

OIR - INSULATION REPRESSURIZATION

1.1	Relief Valve
1.2	Relief Valve
2.1	Helium Receiver
2.2	Helium Receiver
3.	Isolation Valve(s)
4.	Check Valve
5.	Disconnect
6.	Pressurization Regulator

TPB - PURGE BAG

1.1	Vent Valve
1.2	Vent Valve
2.1	LOX Tank Insulation Purge Valve(s)
2.2	LH_2 Tank Insulation Purge Valve(s)

Table 2.0-1. Propulsion and Mechanical Subsystems Components (Cont)

TPB - PURGE BAG (continued)	
3.	Purge Lines
4.1	LH_2 Tank Insulation Purge Manifold
4.2	LOX Tank Insulation Purge Manifold
5.	Intertank/Insulation Selector Valve(s)
6.1	Relief Valve Pressure Switch
6.2	Relief Valve Pressure Switch

Key:

M.O. - Motor Operated
S - Solenoid Operated
H - Hand Operated

2.1 REQUIREMENTS

The assumptions and guidelines used in designing the propulsion and mechanical subsystems were those supplied by NASA in their original study plan, subsequent revisions and numerous letters and telecons. While many of the guidelines had an influence on the designs described in the following sections, the ones listed below had a significant impact on the selected system configuration.

1. The Tug shall be capable of safely venting propellant boiloff gases while on the launch pad, during launch and flight, in orbit and during re-entry while still in the shuttle cargo bay.

2. The Tug shall be capable of being loaded with pressurant and other fluid reactants while in the shuttle cargo bay on the launch pad. Subsystems shall be separate from shuttle, but accessible with shuttle on the pad in the vertical position and with the payload doors closed.

3. The only orbital operations to be assumed are Tug undocking and re-docking with the shuttle and the payload and minimum functional tests of the Tug prior to its separation from the shuttle. After retrieval from orbit and re-docking with the shuttle, the Tug/Orbiter propellant vent and purge interface must be re-established.

4. In the event of an abort, the Tug shall have the capability of safely dumping propellants prior to shuttle orbiter landing. Propellant dump provisions should be provided only during the orbital coast phase of an abort to orbit mode. All propellant dump will be accomplished with the Tug in the cargo bay. Acceleration for Tug propellant settling will be provided by shuttle.

5. All tug propulsion systems will be required to be in a safe condition prior to reentry from orbit in the shuttle.

6. After remating with the shuttle orbiter, the propellants will be dumped and the tanks vented to approx 1 psia, then repressurized with helium. Again the propellant tanks will be vented to approx 1 psia and repressurized with helium for reentry and landings. The helium will be stored in the shuttle cargo bay.

7. Time for Tug propellant dump during orbital phase of an abort from orbit mode is 21 minutes (until 3 minutes before apogee deorbit maneuver) then an additional 27 minutes (3 minutes after de-orbit maneuver until reentry).

8. A single 10,000 lb thrust LOX/LH_2 engine with a minimum guaranteed Isp of 470 sec and a mixture ratio of 6.1 will be used

   a. 120 sec of idle mode is required for engine chilldown

   b. Propellant must be settled for idle mode operation

   c. Engine NPSH requirements are:

      LH_2 - 15 feet (0.5 psi NPSP)
      LOX - 2 feet (1.0 psi NPSP)

9. The system Isp for the GOX/GH_2 Auxiliary Propulsion System (APS) shall be 380 sec.

10. All subsystems shall be designed Fail Safe with an additional requirement for the APS of 30 minutes of attitude hold capability in the event of a failure.

11. The cargo bay internal wall temperature is assumed to be +200°F max.

12. The Tug will be positioned in the cargo bay with the engine forward.

In addition to the NASA established guidelines, the design of the subsystems was influenced by requirements established by other NR disciplines during the study. In all cases, minimum system weight, operational flexibility and redundancy to increase system reliability were the deciding factors if more than one choice of subsystem configuration existed which would satisfy all design requirements.

2.2 ELECTRIC AND PNEUMATIC VALVE ACTUATION SUBSYSTEM EVALUATION

When comparing pneumatically operated valves with electrically operated valves to minimize system weight, all components and system operating parameters must be taken into consideration. In comparing these two modes of operation for a given size valve, the assumption is made that the basic valve weight will be independent of the mode of actuation and only the weights of the actuators and actuation systems need to be evaluated.

For pneumatic actuation, the weight of the actuator housing, piston, return spring, bellcranks or rack and pinion, plus auxiliary interconnecting and attaching hardware must be compared with the weight of the electrical motor, gear reduction box and auxiliary attaching hardware. Preliminary analysis on two to three inch line size cryogenic valves (range to be used in Tug) indicates that the total weight for electrical actuation is no greater than the total weight for pneumatic actuation. This is due primarily to the size of the pneumatic actuator housing, the weight of the actuator piston and heavy return spring as compared to a gear box and an electrical motor sized for intermittent operation at ambient conditions and continuous duty at cryogenic conditions.

This then indicates that the weight of the pneumatic supply system as compared to the weight of the electrical supply system will be the deciding factor. In the pneumatic system, actuation solenoid valve leakage will be the deciding factor in determining the capacity of the storage spheres. For a 1976 technology it has been assumed that actuation solenoid valve leakage will not exceed one standard cubic inches per minute (scim) of helium at the end of a ten mission life, including all prelaunch checkouts and orbital operations. In addition to the storage sphere, additional equipment such as a regulator, relief valve, burst disc, check valves and actuation solenoid valves will be required. The weight of the electrical system must consider the additional weight of the electrical wiring, larger electrical controls and increased fuel cell size. Because the electrical control motors will be of the reversible type with limit switches to control the stroke in both direction, electrical power will only be required during the actuation and deactuation cycle which has been assumed to be 500ms or less. For this short period of actuation the existing fuel cells are capable of supplying this momentary additional power with no increase in size.

As shown in Table 2.2-1, this method of evaluation results in a net weight savings of 57 pounds by utilizing electrical motor actuation in lieu of the more conventional pneumatic actuation for valves in the two to three inch line size. Thus, the decision has been made to use electrically actuated valves in all applications on the Tug.

2.3 PROPELLANT FEED, FILL AND DRAIN SYSTEM

The propellant feed, fill, and drain system (Figures 2.3-1 and 2.3-2) consists of the Tug vehicle mounted plumbing to accomplish ground propellant servicing, flight dumping of propellants, ground and flight fill and drain of the auxiliary tanks, MPS and APS propellant feed, and propellant line conditioning. The plumbing is optimized in size considering line pressure drop and line weight to achieve a minimum system weight for the stage. The system is designed to minimize propellant lost at engine shutdown and to provide for efficient propellant temperature control. The installation layout of the system is shown in Figures 2.3-3 and 2.3-4.

2.3.1 Operational Summary

For ground operations, each main propellant tank is filled or drained through a three-inch line with two parallel-redundant 2-inch valves located

Space Division
North American Rockwell

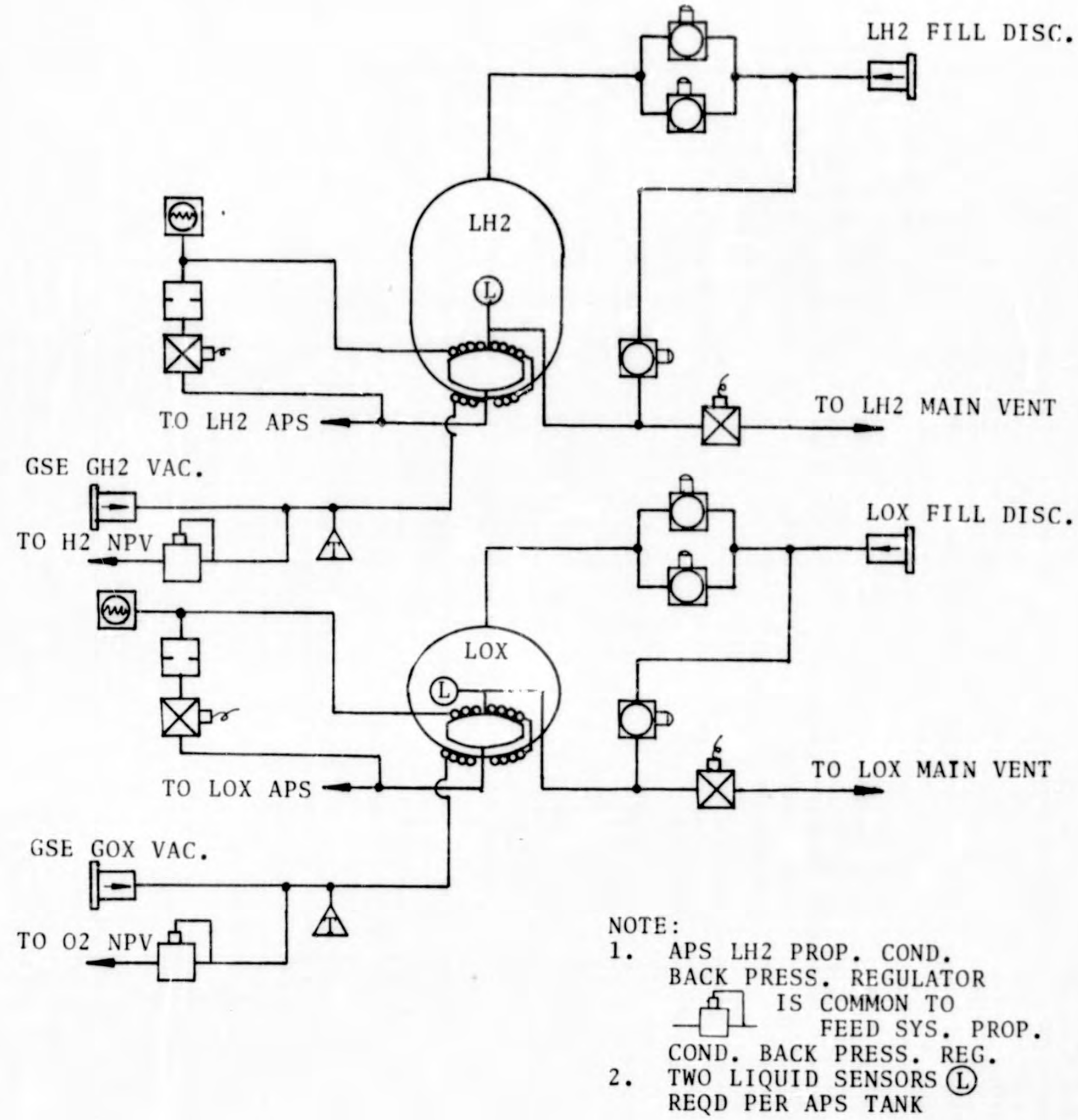

Figure 2.3-1 Tug Propellant Fill & Drain & APS Prop. Tank Conditioning

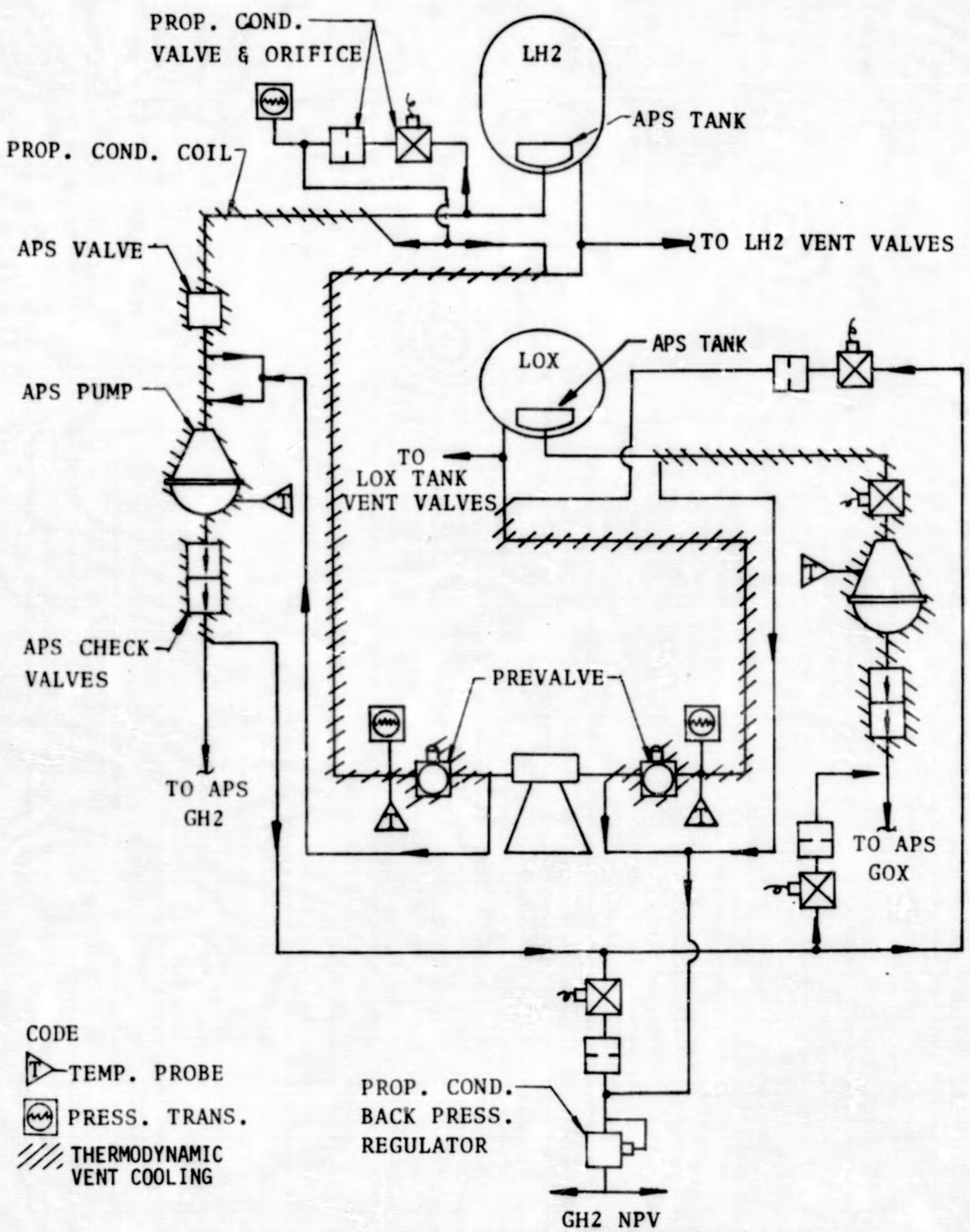

Figure 2.3-2 Tug Propellant Feed & Conditioning System

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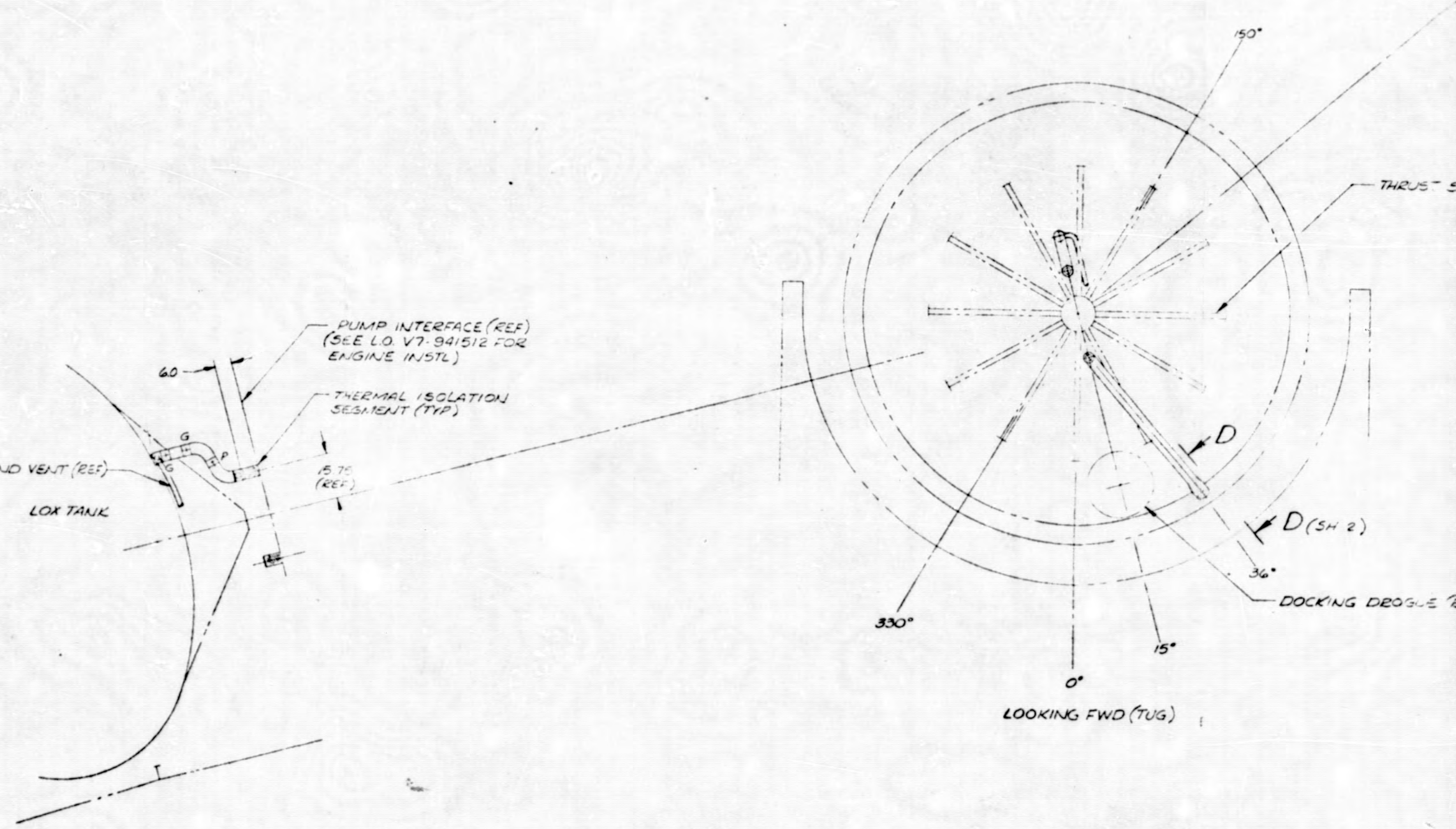
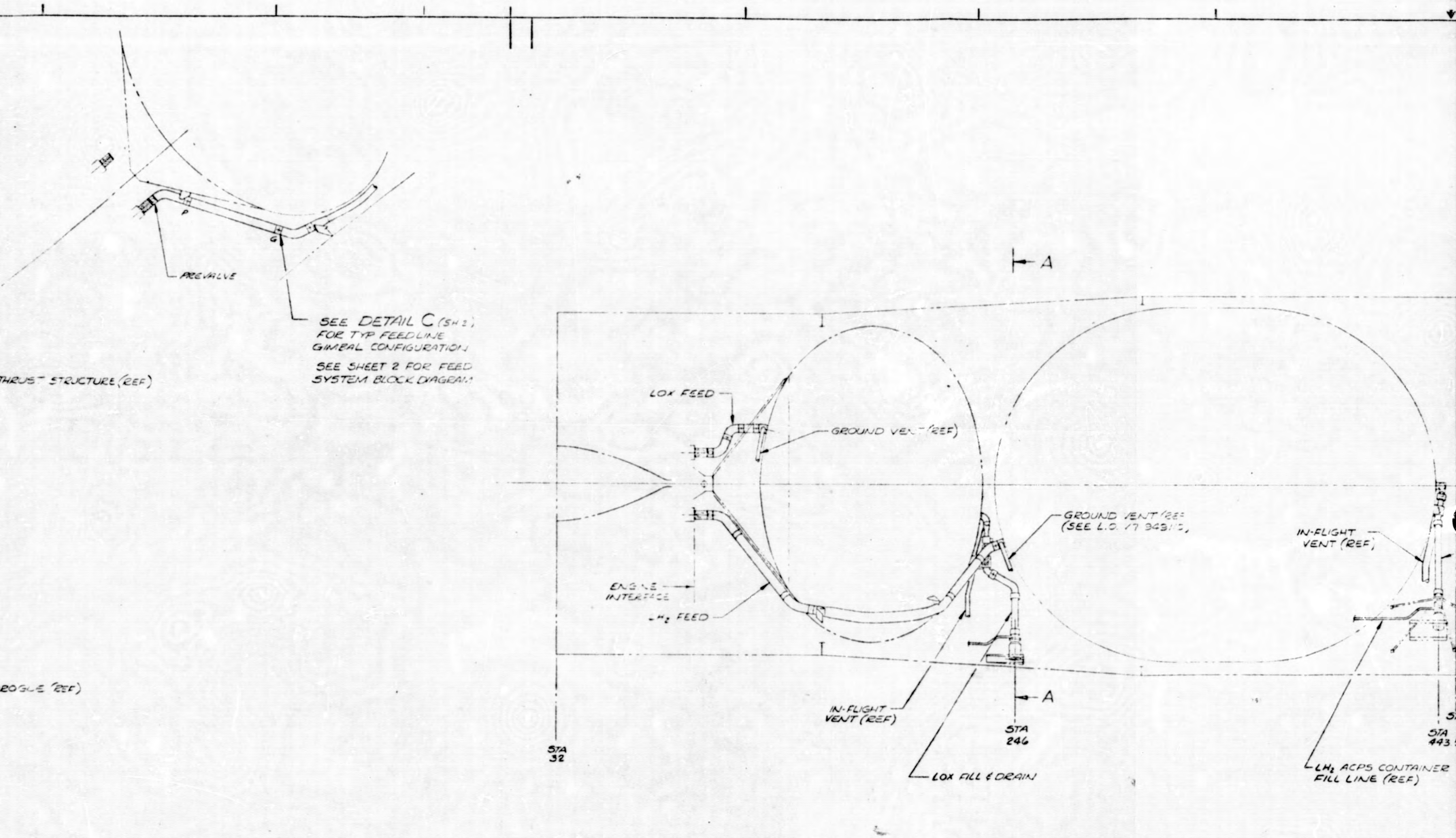


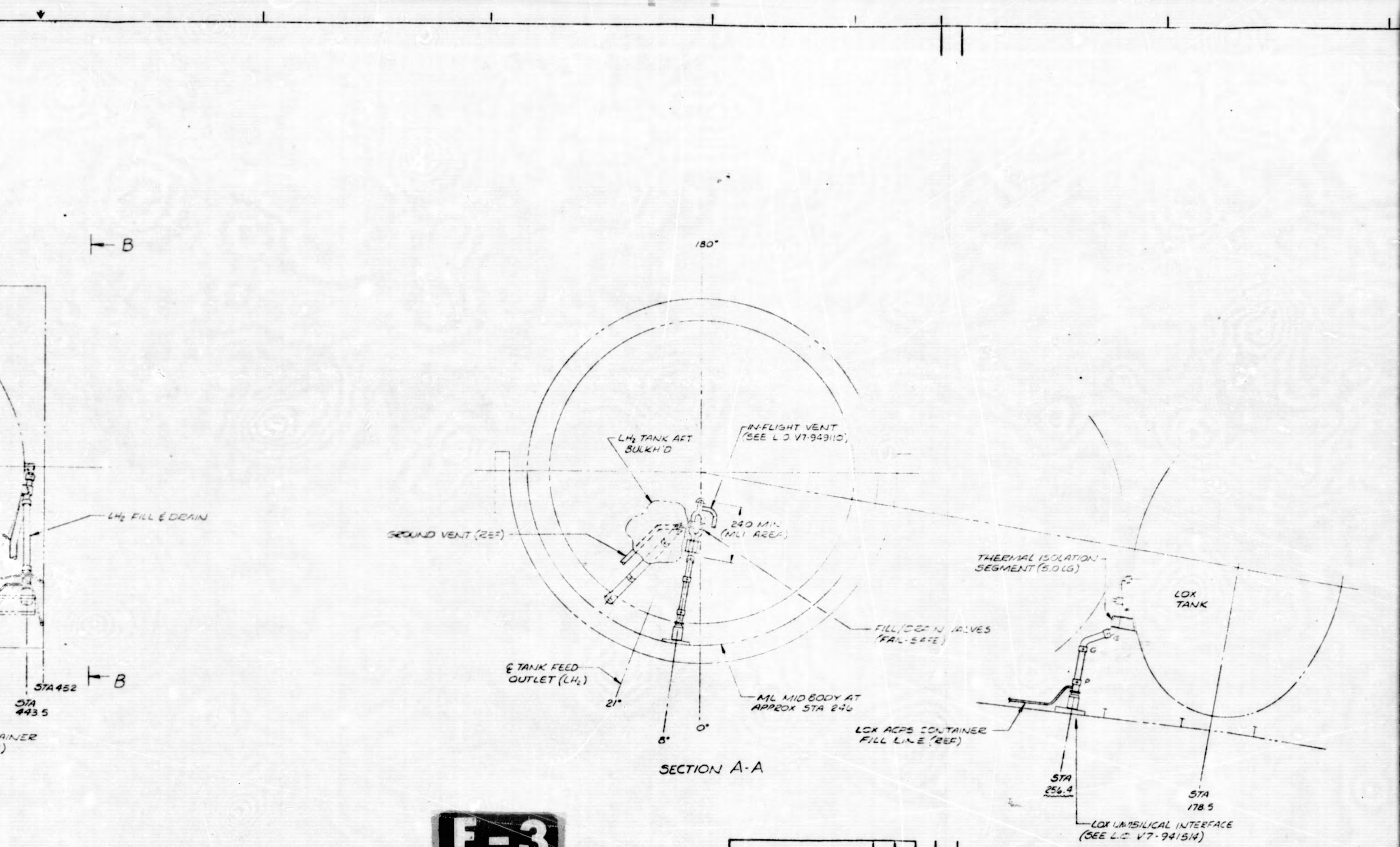
PUMP INTERFACE (REF)
(SEE L.O. V7-941512 FOR
ENGINE INSTL)
60
THERMAL ISOLATION
SEGMENT (TYP)
G
G
P
15.75
(REF)
GROUND VENT (REF)
LOX TANK
150°
THRUST ST
D
D (SH 2)
36°
DOCKING DROGUE (R
330°
15°
0°
LOOKING FWD (TUG)

F-1



P
G
PREVALVE
SEE DETAIL C (SH 2) FOR TYP FEEDLINE GIMBAL CONFIGURATION
SEE SHEET 2 FOR FEED SYSTEM BLOCK DIAGRAM
THRUST STRUCTURE (REF)
ROGUE (REF)
A
LOX FEED
GROUND VENT (REF)
ENGINE INTERFACE
LH_2 FEED
GROUND VENT (REF
IN-FLIGHT VENT (REF)
IN-FLIGHT VENT (REF)
A
STA 32
STA 246
STA 443
LOX FILL & DRAIN
LH_2 ACPS CONTAINER FILL LINE (REF)

F-2



SECTION A-A

F-3

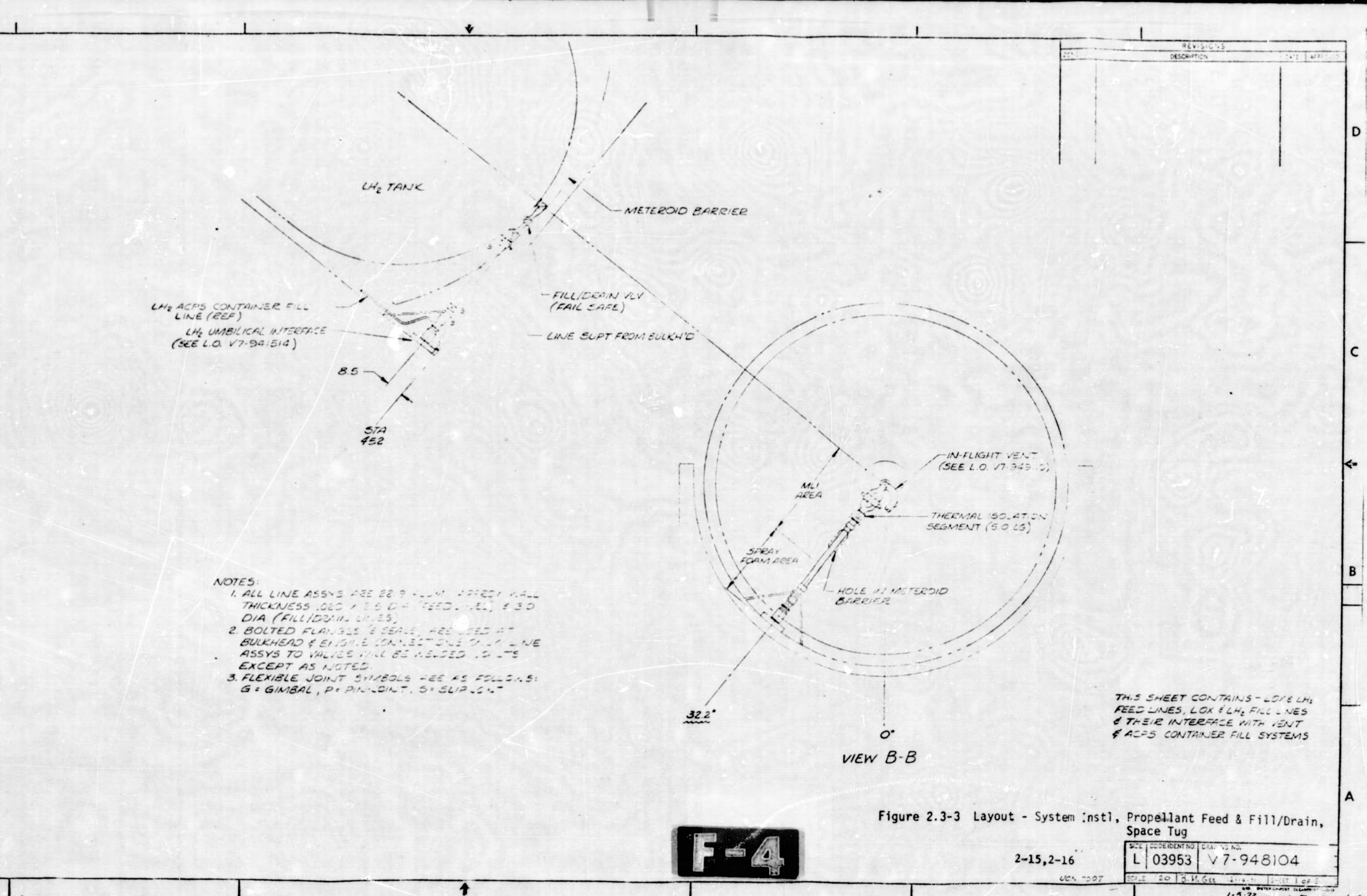
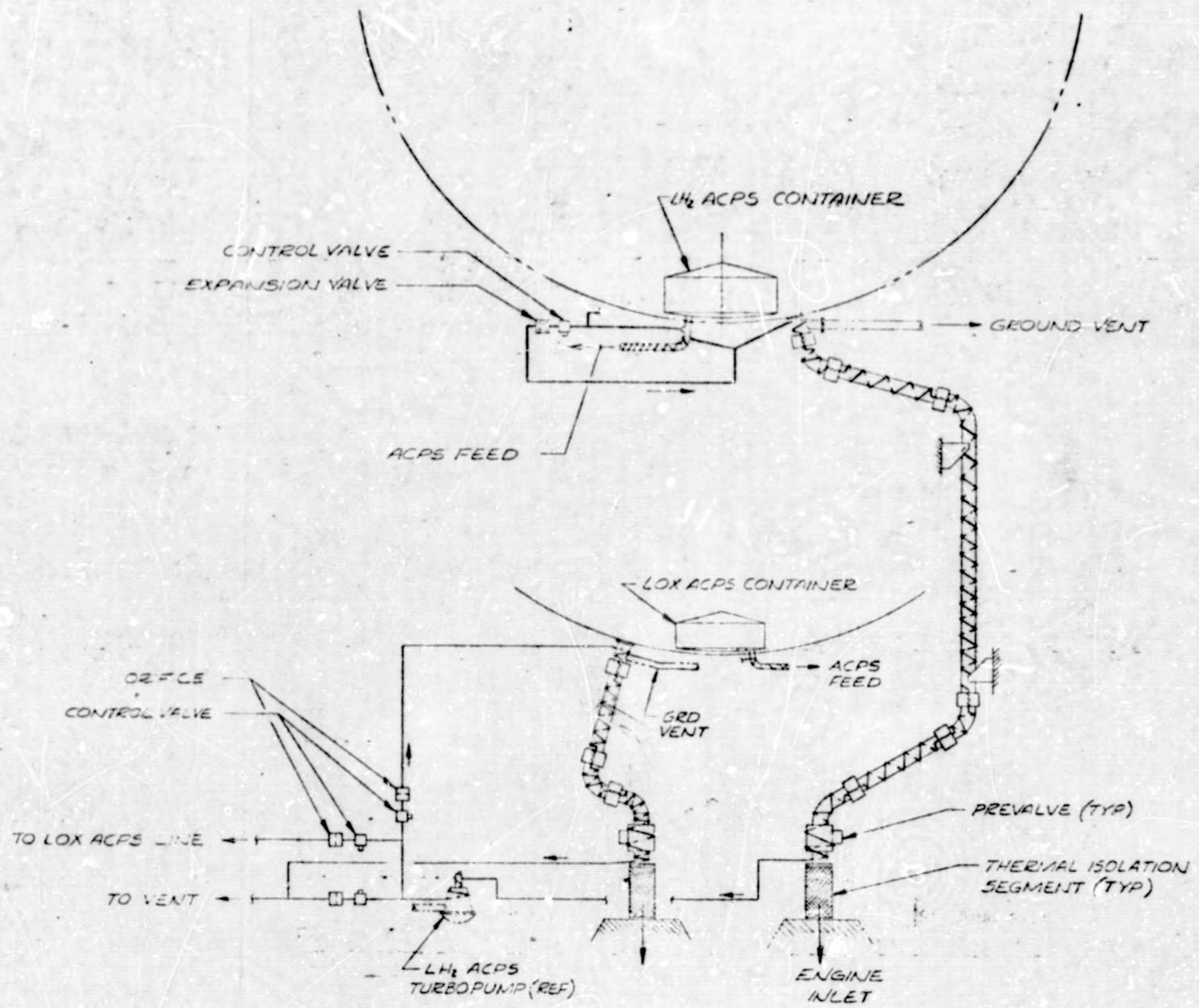


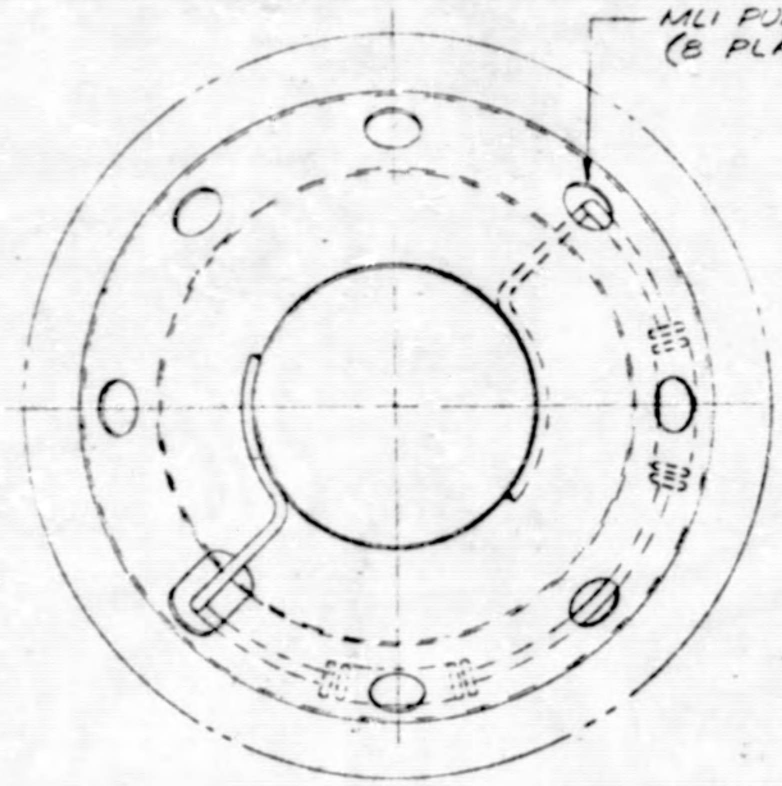
Figure 2.3-3 Layout - System Instl, Propellant Feed & Fill/Drain, Space Tug

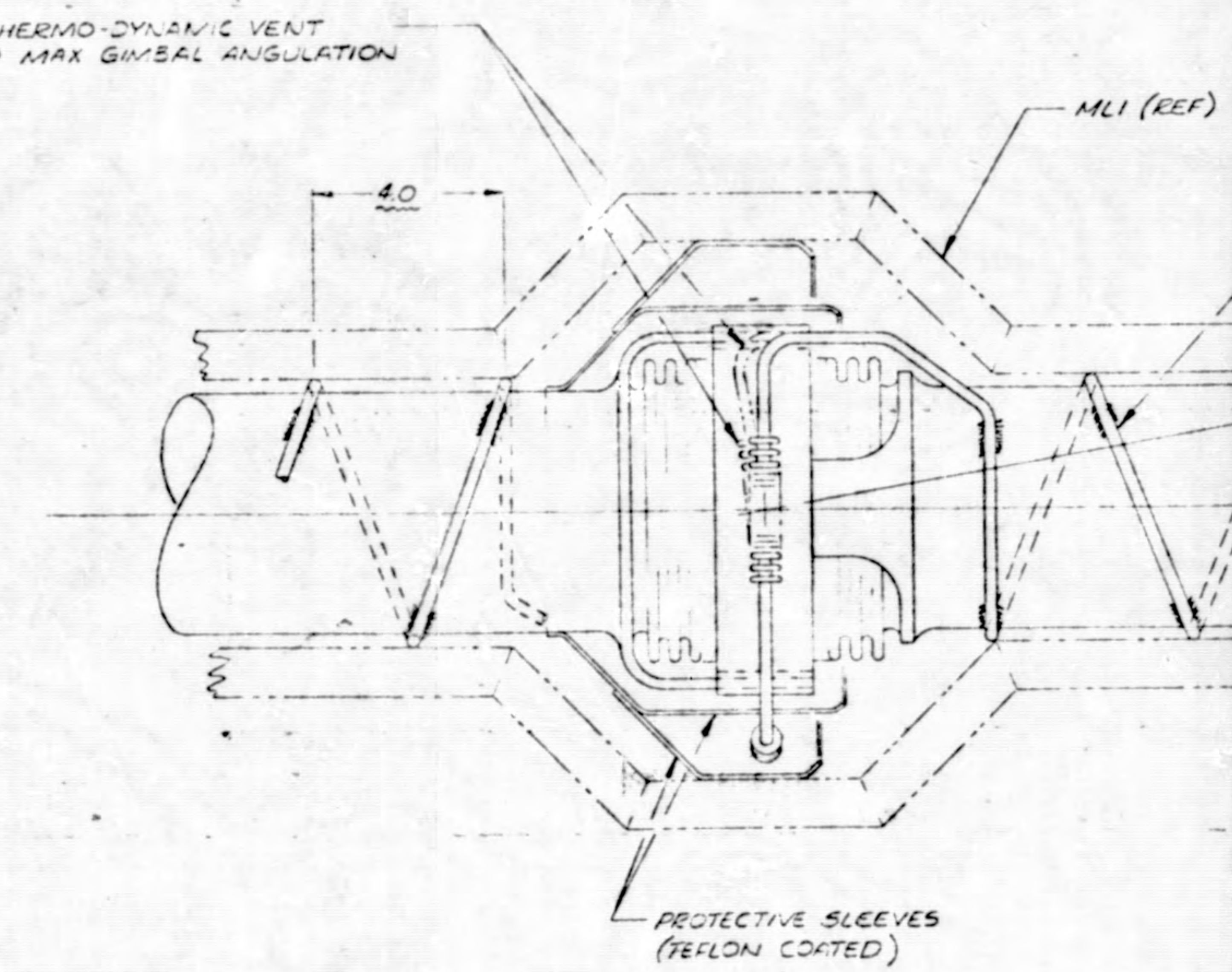
F-4



BLOCK DIAGRAM
FEEDLINE & THERMO-DYNAMIC VENT CIRCUIT

F-1



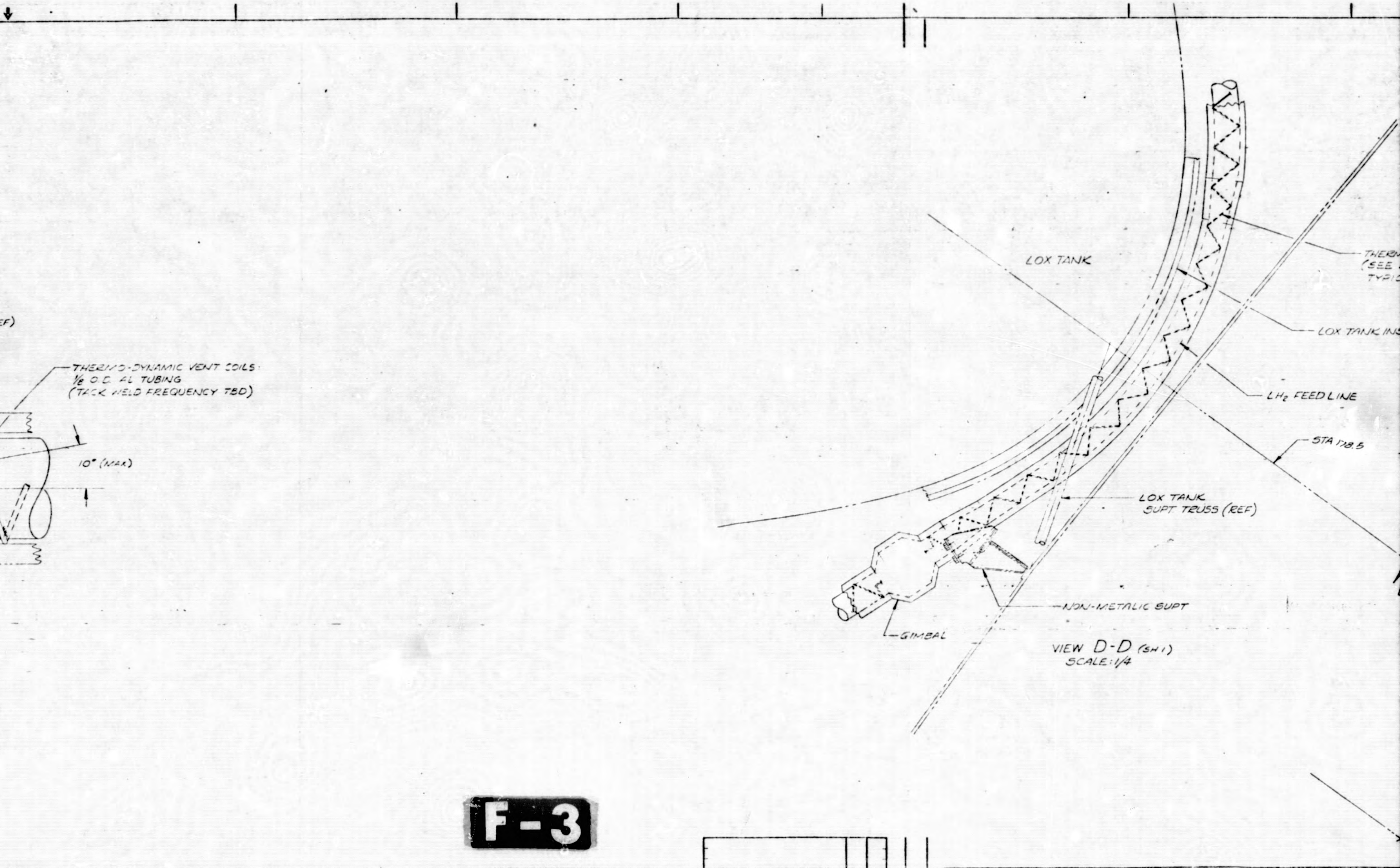


DETAIL C (SH 1)
SCALE: 1/1

THERMO-DYNAMIC VENT CONFIGURATION
AT A TYP GIMBAL OR PIN-JOINT

NOTE: IF FLOW REQMTS PERMIT, AN INTERNAL
GIMBAL WOULD GREATLY REDUCE
ENVELOPE & WEIGHT.

F-2



EF)
THERMO-DYNAMIC VENT COILS
1/2 O.D. AL TUBING
(TACK WELD FREQUENCY TBD)
10° (MAX)
LOX TANK
THERM
(SEE
TYPIC
LOX TANK INS
LH_2 FEED LINE
STA 178.5
LOX TANK
SUPT TRUSS (REF)
NON-METALIC SUPT
GIMBAL
VIEW D-D (SH 1)
SCALE: 1/4

F-3

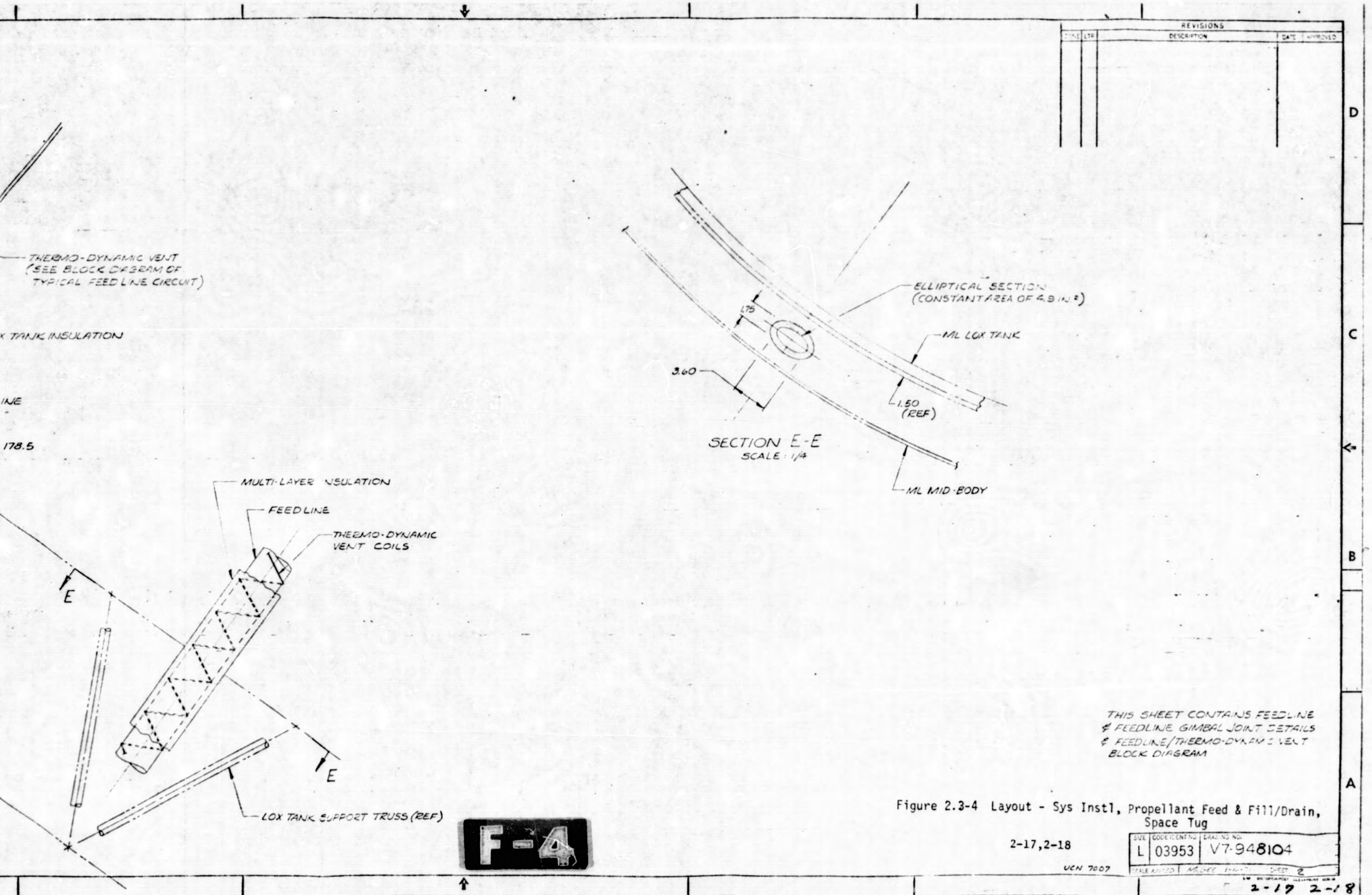


Figure 2.3-4 Layout - Sys Instl, Propellant Feed & Fill/Drain, Space Tug

Table 2.2-1. Subsystem Weights

Components	Pneumatic Actuation	Electrical Actuation (Additional)
Lines	15.00 lbs	NA
Control Solenoids	15.00	NA
Regulator	2.25	NA
Relief Valve	0.50	NA
Burst Disc	0.25	NA
Ch. Valves	1.00	
He Receiver (3/4 ft3, 3250 PSI)	28.00	NA
Wiring	NA	5 lbs
Total	62 lbs	5 lbs
Net Saving		57 lbs

NA - Not applicable

at the top of the tank. This same system will be used for in-flight propellant dumps since they occur with the Tug in the cargo bay and with the propellants settled by thrust from the orbiter. Parallel valves are required to assure abort drain capability and the 3-inch line allows drain of the propellants within 27 minutes. Ground fill and drain of the APS auxiliary propellant tanks is accomplished during the main tank fill operation by tapping off the main fill line through a one-inch auxiliary tank fill valve. In-flight refill of this tank through the screened inlet port occurs only during periods of positive thrust (APS or MPS). After the propellant has been settled, opening the auxiliary tank vent valve initiates refill and the ullage gas is vented overboard through the non-propulsive vent system. Refill is terminated when the vent line liquid level point sensors indicate wet. To minimize the heat leak into these tanks they are wrapped with small diameter (1/8 inch with 4 inch pitch) tubing which comprises part of a thermodynamic vent cooling circuit. To permit these circuits to function on the ground they are connected to a GSE vacuum source. In flight they are connected to their respective non-propulsive vent systems (either GH_2 or GOX). (Refer to Figure 2.3-1).

Propellants are delivered to the main engines through insulated 2-1/2 inch feedlines and prevalves, which are the optimum size from a stage weight standpoint. The entire feed system is wrapped (inside the MLI) with small diameter (1/8 inch with 4 inch pitch) tubing which is part of the thermodynamic vent cooling circuit. Use of this system plus the insulation will minimize boiloff when the MPS is not operating and will minimize system chilldown time

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and propellant losses during each engine start. Refer to Figure 2.3-2 for a fluid schematic of the propellant feed and conditioning system.

Propellant conditioning control is accomplished by sensing the feed system conditioning propellant pressure and four system temperatures to control the conditioning system operation through four control valves in accordance with the schematic of Figure 2.3-5.

2.3.2 System Selection Rationale

The diameter of the LOX and LH_2 feed lines was optimized to minimize the overall stage weight. As the feed line diameter increases, the weight of the feed line components increase, but the structural and ullage pressurant weight decreases since the pressure drop decreases. Figure 2.3-6 shows the results of this study.

To minimize the heat leak into each propellant tank from the engine during the coast periods, a prevalve and a thermal isolation segment are required in each feed line. The use of the prevalve, closed during the coast period, will eliminate the heat leak due to conduction through the fluid. The use of the thermal isolator will reduce the heat leak due to conduction through the line wall.

To minimize the weight of the propellants lost after each main engine shutdown (trapped on the engine side of the prevalve), the prevalve must be located as near the engine as possible while still permitting effective thermal isolation. The weight of propellant per foot of line is 2.42 lb for LOX and 0.15 lb for LH_2. Assuming a length below the tank outlet of 17 ft for LH_2 and 3 ft for LOX and a 6-burn mission (five shutdown losses), the weight loss is 49 lbs for tank mounted prevalves. With the prevalves located within 24 inches of the engine (to allow for a thermal spacer and venting) the weight loss is 25 lbs or a weight savings of 24 lbs, assuming no weight increase for the thermal spacer.

To maintain subcooled liquid in the MPS and APS feedlines and to reduce the weight penalty caused by LOX boiloff losses, a feedline propellant conditioning system, employing a thermodynamic vent cooling concept will be used. Assuming the heat inputs listed below for a 144-hour mission, 49 lbs of LH_2 and 83 lbs of LOX will be lost due to boiloff (no cooling of the lines and pump).

ASSUMED HEAT INPUTS

	LH_2 BTU/Hr	LOX BTU/Hr
MPS Feedline	12	2
APS Feedline	4	.6
APS Pump	50	50

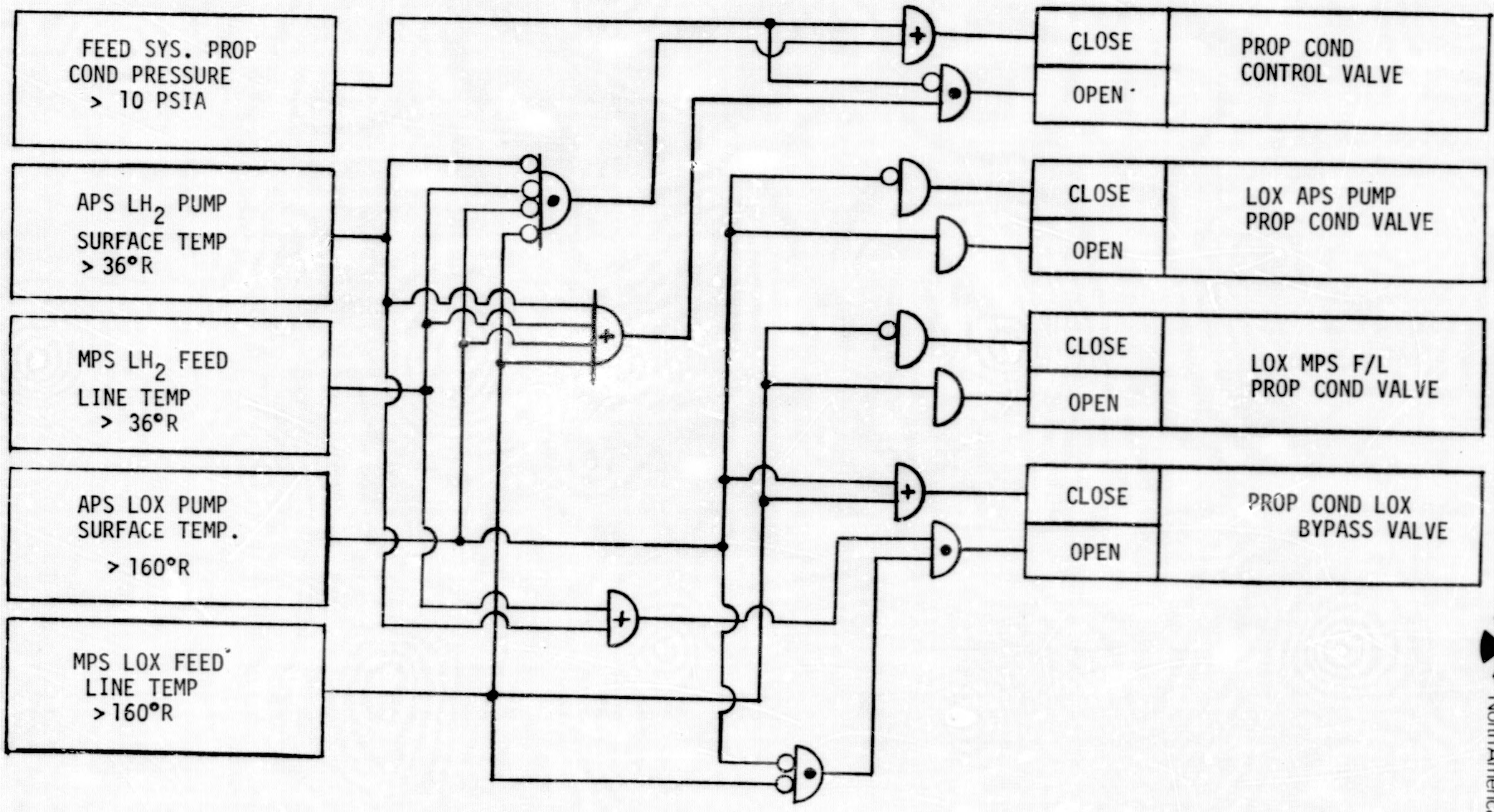

Figure 2.3-5 Propellant Feed - Propellant Conditioning System, Control Requirements

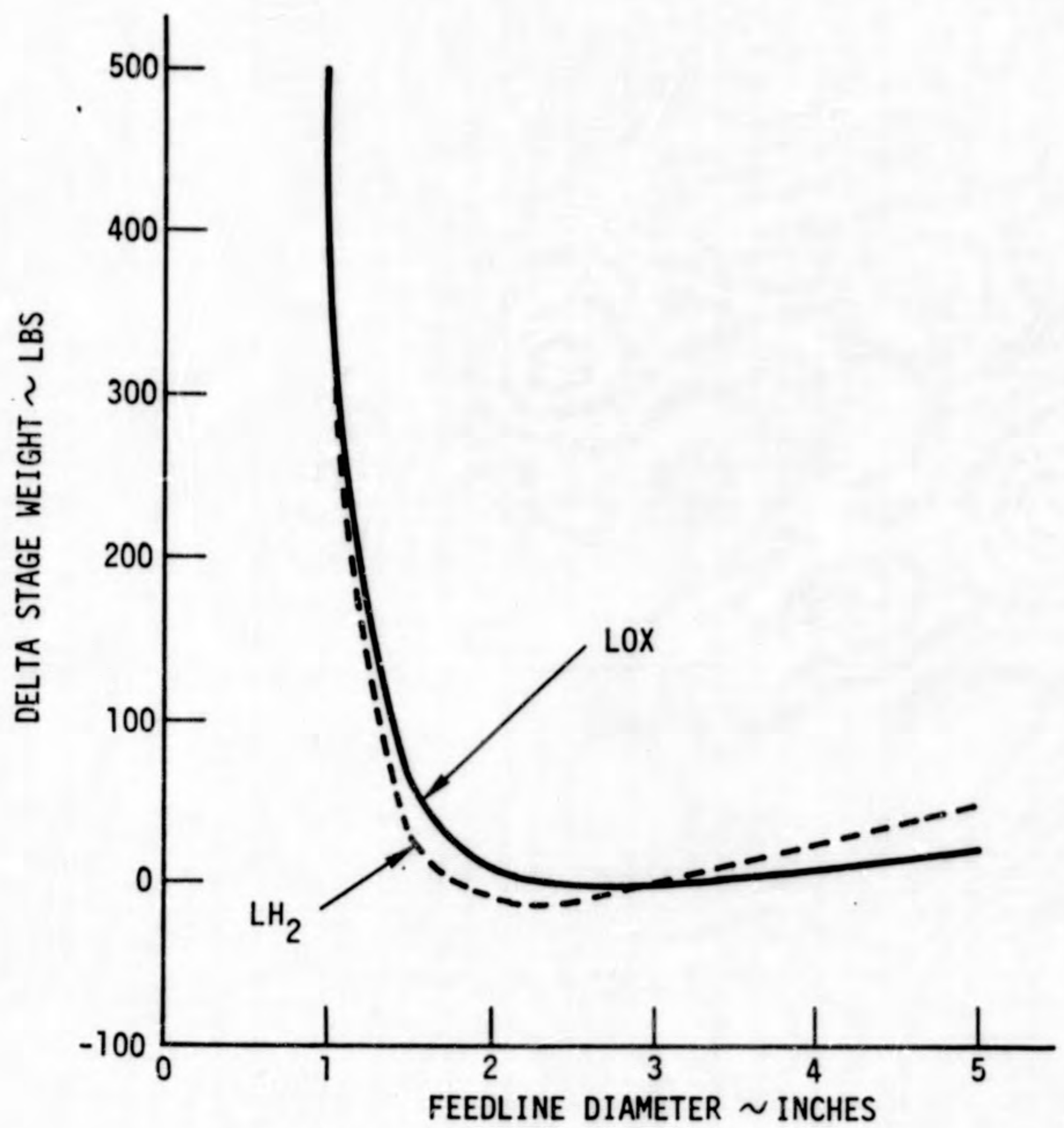

Figure 2.3-6 Feedline Optimization

The conditioning system uses a LH_2 coolant flow rate of 0.38 lbs/hr. The resulting propellant loss for the 144-hour mission is 54 lbs., which is all LH_2. There is no LOX boiloff loss due to the above heat inputs with the use of thermodynamic vent cooling system. Thus the weight savings is 78 lbs. This savings, however, is slightly reduced by the propellant conditioning system hardware weight.

2.4 SAFING AND VENTING SUBSYSTEM

The primary function of the Tug safing and venting subsystem is to provide the necessary venting capability for prevention of over-pressurization of the main propellant tanks and still hold the propellant tank pressures at a level high enough to assure engine operations. Another function of the venting subsystem is to maintain a positive pressure in the propellant tanks to maintain structural integrity.

The tug safing subsystem is required for safing the Tug propellant tanks prior to re-entry. The safing subsystem utilizes the Tug fill and drain subsystem for the liquid dump portion of the propellant tanks safing operations. A separate stored helium subsystem is also a part of the Tug safing subsystem. In addition to providing helium gas for propellant tanks safing operations, the stored helium subsystem also provides helium repressurization gas to the Tug insulation subsystem during re-entry. Another function of the tug safing and venting subsystem is to vent the main propellant tanks either to atmosphere or a burn stack during tanking operations on the ground.

Since the tug can have the liquids in the main propulsion tanks at either end of the tanks depending on whether or not the tug is in the cargo bay of the orbiter or outside the orbiter, the vent subsystem provides venting capability from either the top or bottom of the tanks.

A separate venting subsystem is provided for the APS for use during its fill and purge operations.

2.4.1 Functional Operation and Electrical Requirements

a. Subsystem Functional Requirements

The tug safing and venting subsystem is shown schematically in Figure 2.4-1. The subsystem installation layout is shown on Figure 2.4-2. To minimize relief valve complexity, the vent subsystem is comprised of two separate subsystems. One vent subsystem consists of two, two-inch, dual mode relief valves (2) that relieves pressure during flight and in-orbit. The two relief valves are redundant and either valve is sized to handle total subsystem vent requirements during flight and in orbit. The other vent subsystem consists of a two position shutoff valve (4) that is opened on the ground during loading and ground hold operations. Due to the large boiloff rate during loading of 666 lbs/hr on the LH_2 propellant tank this subsystem consists of a three-inch shutoff valve and three-inch lines. On the LOX tank the boiloff rate is 478 lbs/hr and the LOX ground vent subsystem consists of a two-inch shutoff valve and two-inch lines.

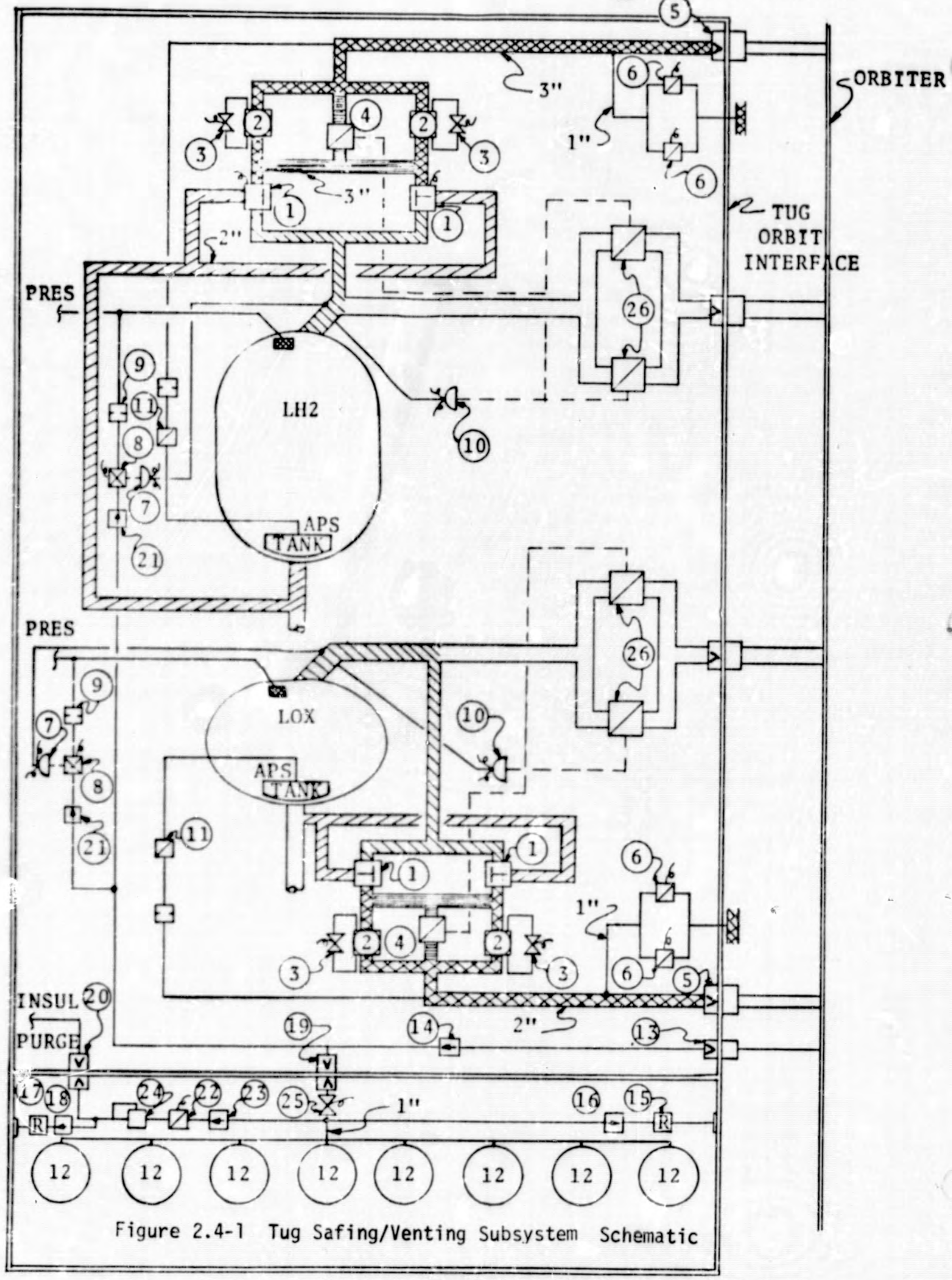

Figure 2.4-1 Tug Safing/Venting Subsystem Schematic

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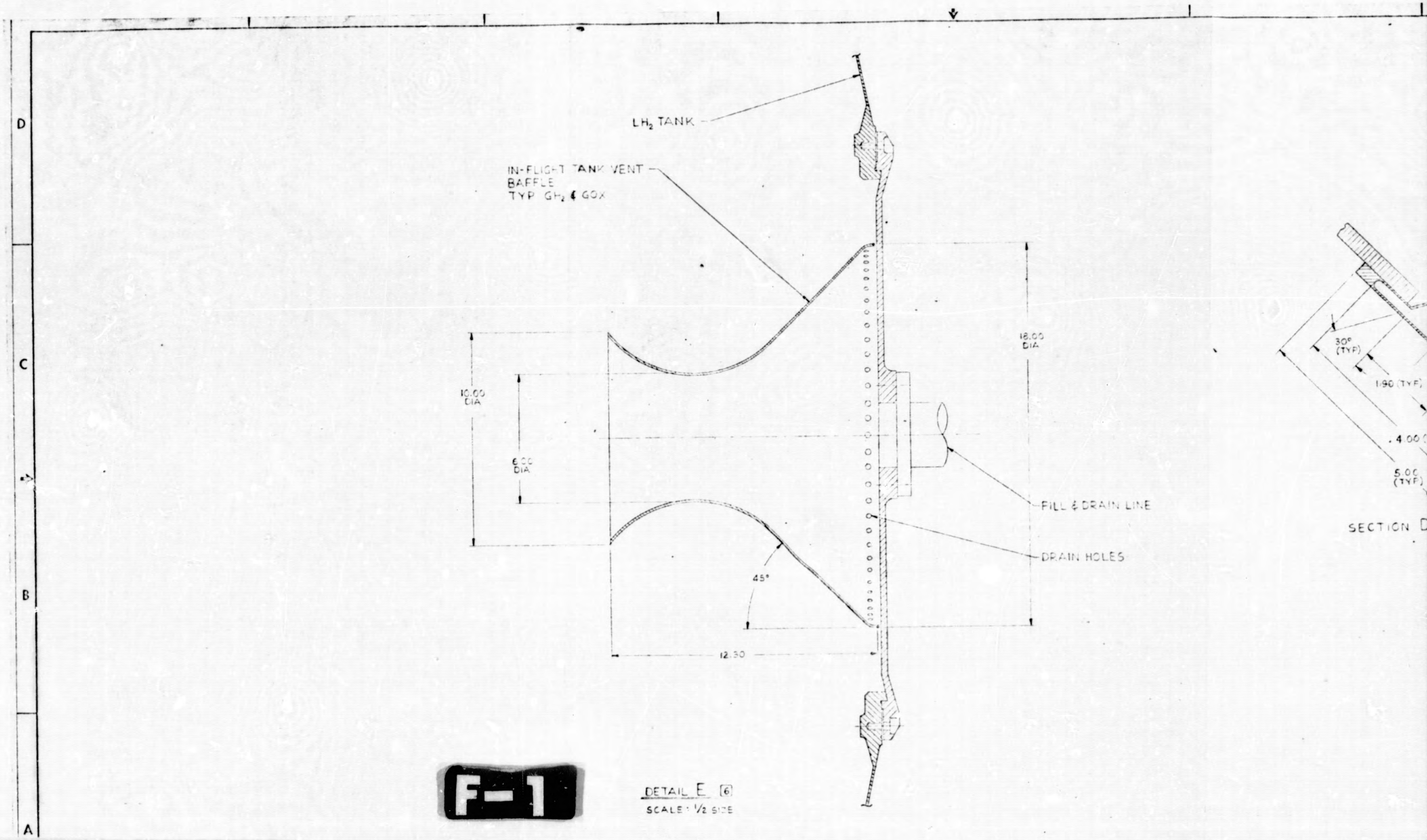
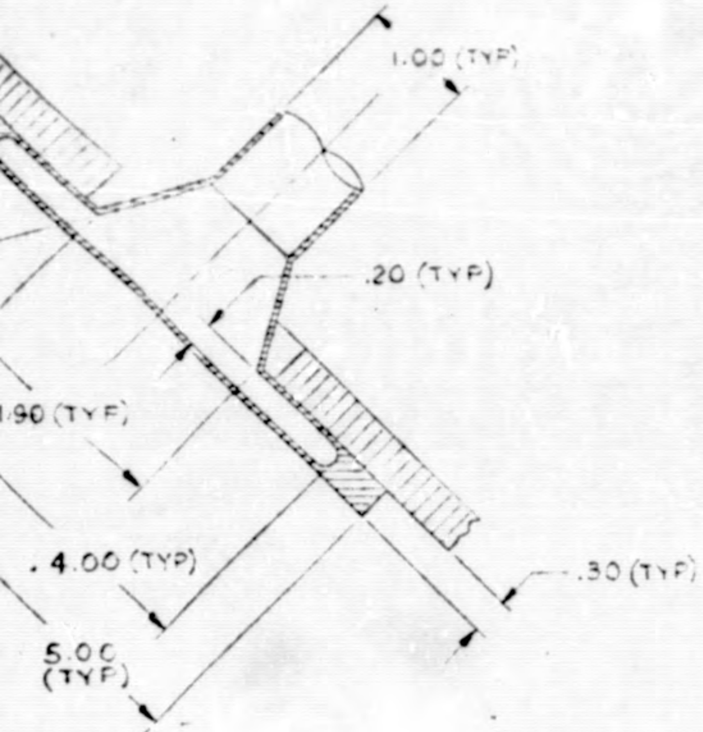
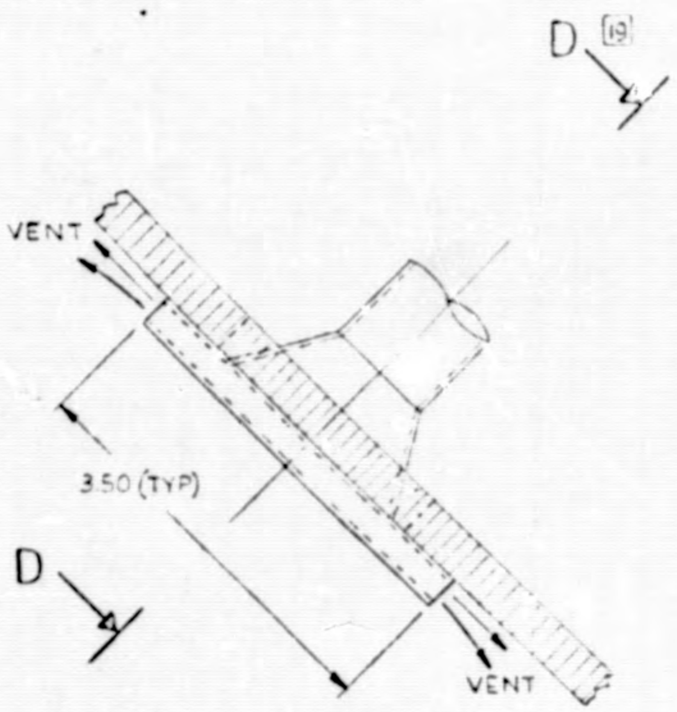


DETAIL E [6]
SCALE: 1/2 SIZE



TION D-D 16



DETAIL C 14
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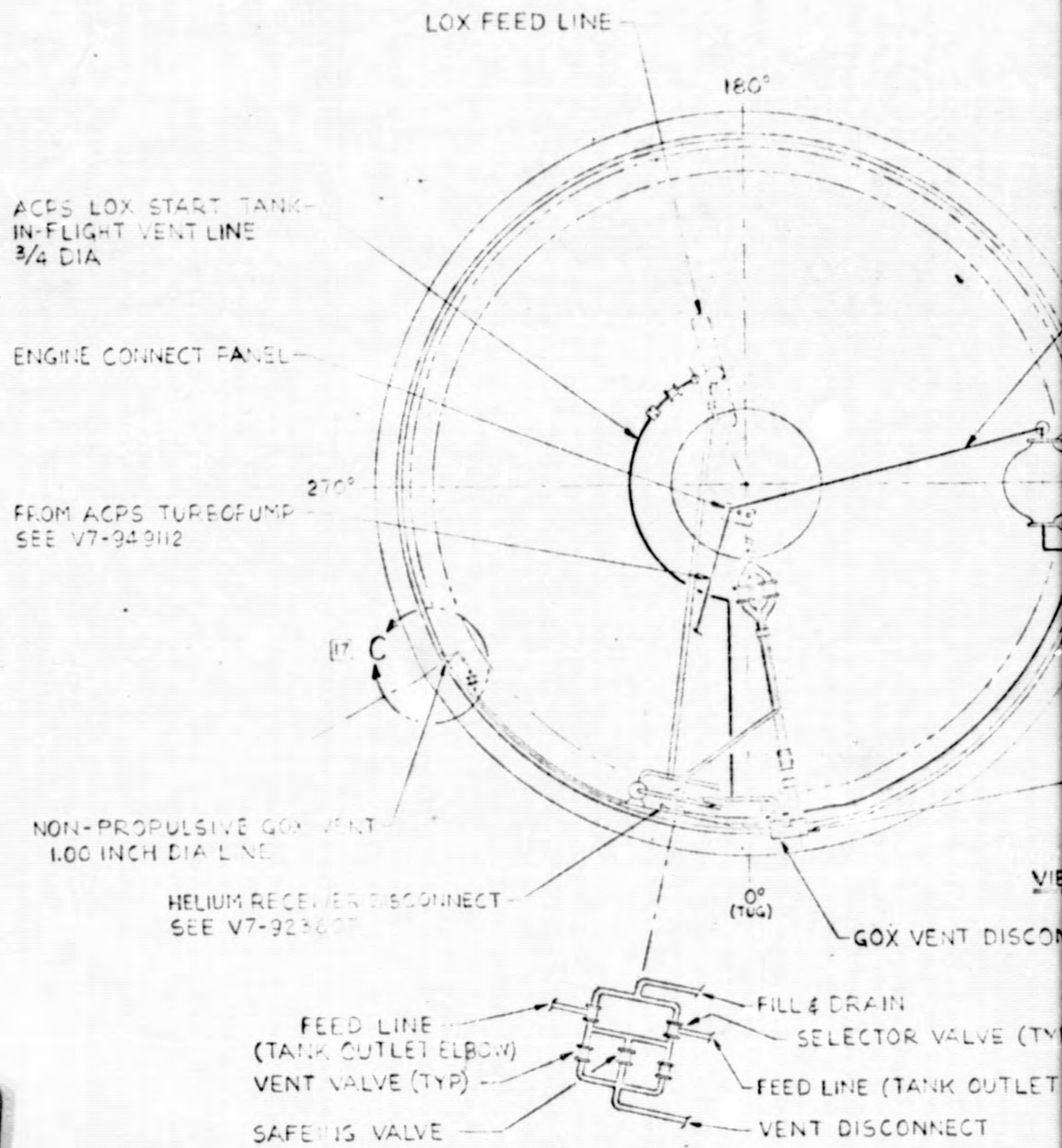
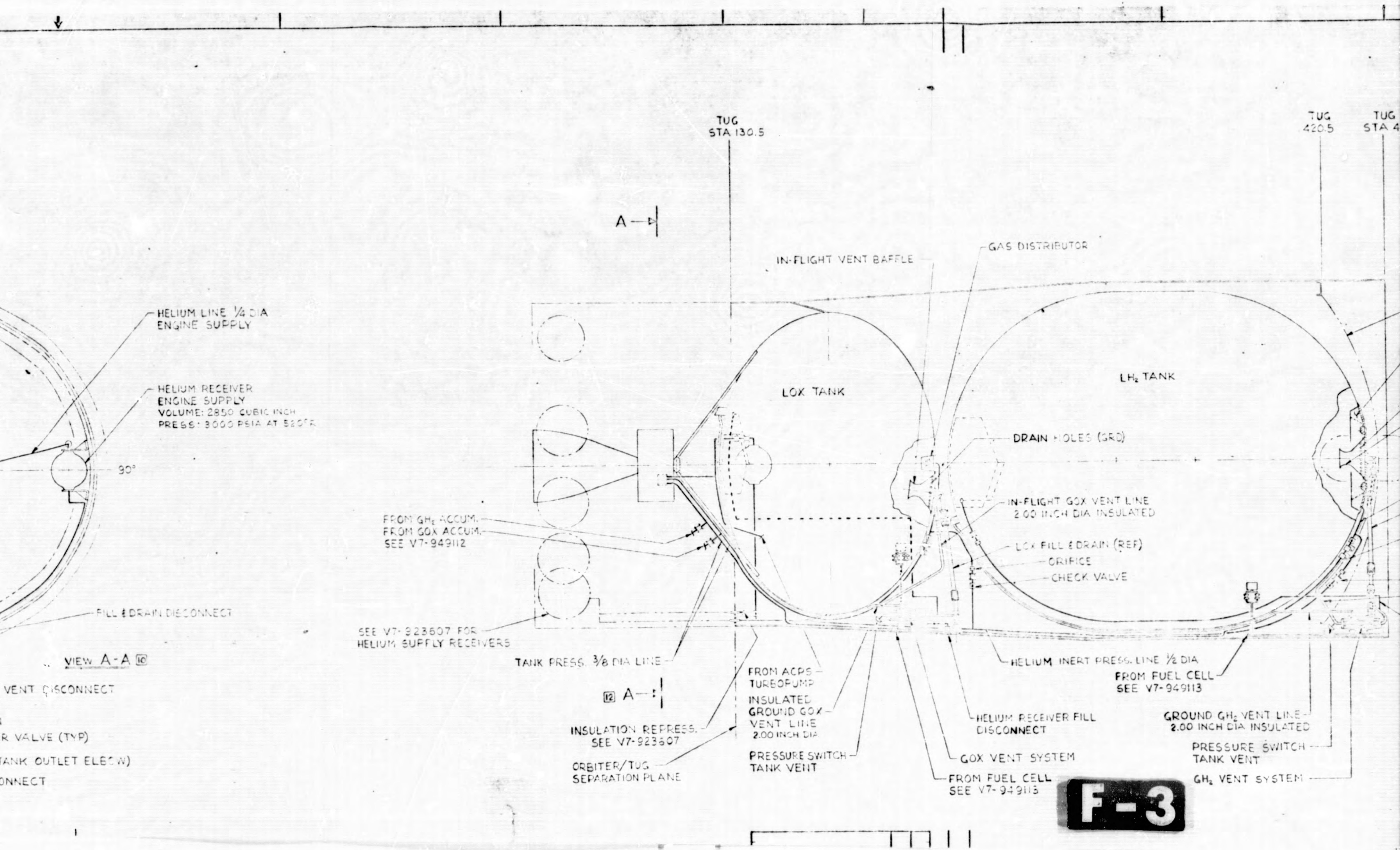
TYP NON-PROPULSIVE VENT

F-2



TYP VENT SYSTEM



TUG
STA 130.5
TUG
420.5
TUG
STA 4
A
HELIUM LINE 1/4 DIA
ENGINE SUPPLY
HELIUM RECEIVER
ENGINE SUPPLY
VOLUME: 2850 CUBIC INCH
PRESS: 3000 PSIA AT 520°R
90°
FILL & DRAIN DISCONNECT
VIEW A-A 10
VENT DISCONNECT
R VALVE (TYP)
TANK OUTLET ELBOW)
ONNECT
IN-FLIGHT VENT BAFFLE
GAS DISTRIBUTOR
LOX TANK
LH2 TANK
DRAIN HOLES (GRD)
IN-FLIGHT GOX VENT LINE
2.00 INCH DIA INSULATED
LOX FILL & DRAIN (REF)
ORIFICE
CHECK VALVE
FROM GH2 ACCUM.
FROM GOX ACCUM.
SEE V7-949112
SEE V7-923607 FOR
HELIUM SUPPLY RECEIVERS
TANK PRESS. 3/8 DIA LINE
12 A
INSULATION REPRESS.
SEE V7-923607
ORBITER/TUG
SEPARATION PLANE
FROM ACPS
TURBOPUMP
INSULATED
GROUND GOX
VENT LINE
2.00 INCH DIA
PRESSURE SWITCH
TANK VENT
HELIUM INERT PRESS. LINE 1/2 DIA
FROM FUEL CELL
SEE V7-949113
HELIUM RECEIVER FILL
DISCONNECT
GOX VENT SYSTEM
FROM FUEL CELL
SEE V7-949113
GROUND GH2 VENT LINE
2.00 INCH DIA INSULATED
PRESSURE SWITCH
TANK VENT
GH2 VENT SYSTEM

F-3

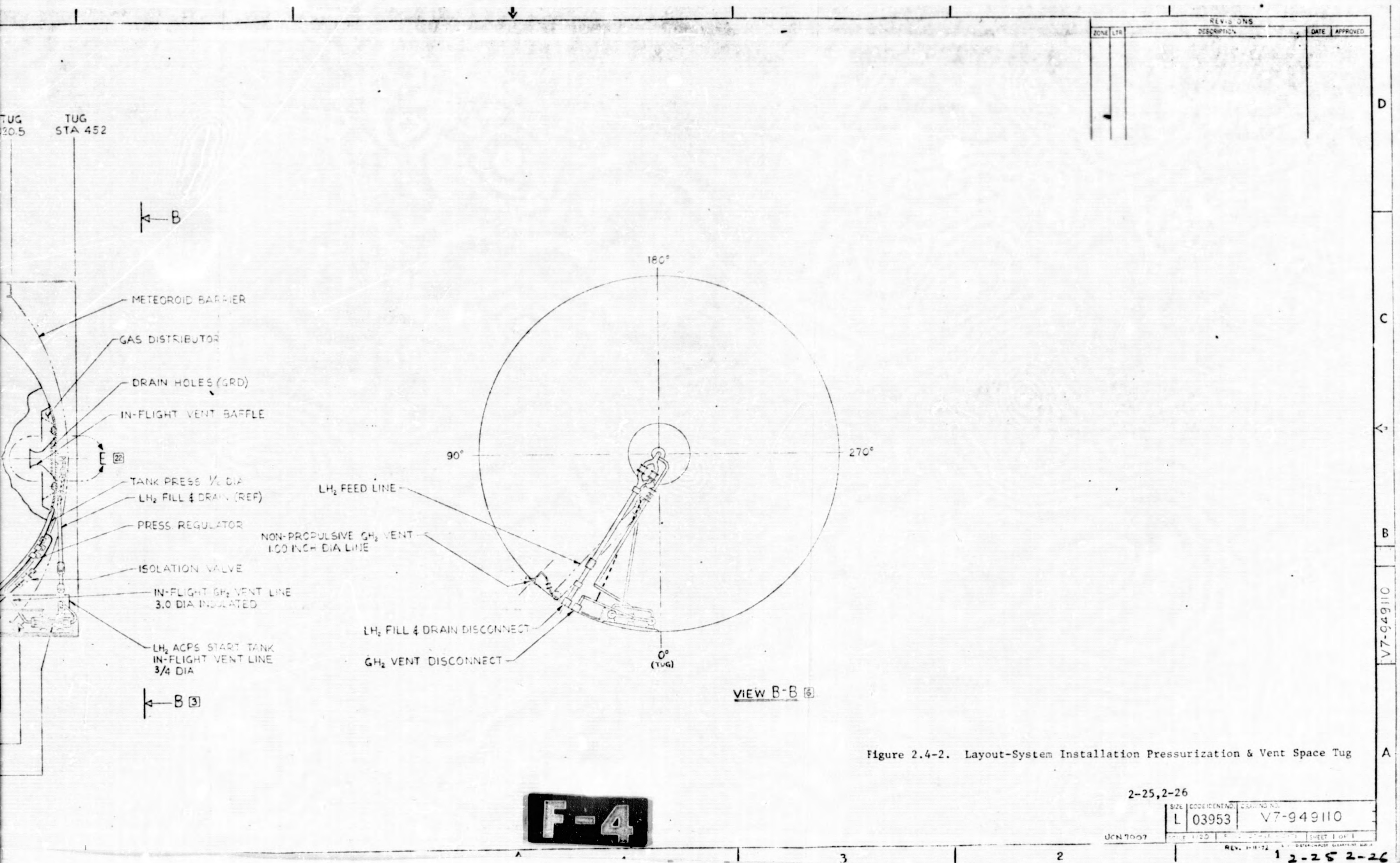


Figure 2.4-2. Layout-System Installation Pressurization & Vent Space Tug

F-4

The ullage pressure during ground hold is predicted to be 15.2 psia for the LH_2 propellant tank and 15.1 psia for the LOX propellant tank. During liftoff and up to orbit the relief valves on the LH_2 propellant tank are in the high mode of operation (24 psia relief pressure). The relief valves on the LOX propellant tank, due to structural considerations, are in the low mode of operation (15-16 psia). The low mode of operation pressure setting for the LH_2 tank relief valves is 16-17 psia. During on-orbit and orbit to orbit operations, all propellant tank relief valves will be in the high mode of operation except for short periods of time when the bulk liquids on the tug require cooling. This will be accomplished by switching to the low mode of operation on both sets of relief valves and the bulk liquids will cool due to evaporation of the bulk liquid as the liquid stabilizes at the lower vapor pressure and temperature.

Safing the propellant tanks is required prior to reentry. This may occur either due to an abort situation or a normal return to earth. The liquids in the propellant tanks are dumped utilizing liquid vapor pressure only and no additional pressurization of the tanks is required.

Since the vent rates during orbit to orbit operations are predicted to be small, the forces created by plume effects against the side of the Tug are small and within the capability of the APS system to maintain attitude control.

b. Electrical Requirements

Tables 2.4-1 and 2.4-2 are tabulations of subsystem valves electrical power requirements.

Actual time during orbiter flight that applied power is required is described in the following vent-safing subsystems functional description.

c. Vent-Safing Subsystem Functional Description (Ref. Figure 2.4-1).

The vent/safing subsystems functions are to provide necessary ullage pressure in the tank for main engine operations, to provide structural integrity for the Tug, and to safe the propellant tanks.

The tank propellant vent subsystems have the capability of venting the propellant tanks from either top or bottom. This capability is provided by a two-inch motor driven shuttle valve (1), in the vent subsystem. Two shuttle valves are provided in each tank vent subsystem to provide a fail safe subsystem. Two relief valves (2), are provided on each propellant tank. The relief valves are dual mode valves and are in parallel to provide a fail-safe subsystem. The low mode of operation is 16-17 psia for the LH_2 tank and 15-16 psia for the LOX tank. The high mode of operation is 23-24 psia. This mode is required for engine burn operations. High mode is the normal mode for the relief valves. A solenoid valve (3), mounted on each relief valve provides the capability of switching the relief valve mode of operation from high mode to low mode. These solenoids are latching solenoids.

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Table 2.4-1. NR Tug EPS Input Data Sheet
LH_2 Tank Safing and Venting Subsystem

Subsystem -			Operational Phase -				
			Power Required (In Watts)				
Component	Voltage/ Frequency Required	Regulation Required (Percent)	Max Demand Inrush/ Peak	Operational Usage (Avg)	Standby Usage (Avg)	Emergency Power	Duration (Duty Cycle)
Relief Valve Solenoid (2)	28 VDC		None	85W-Cryo 85W-Cryo	0	85W-Cryo 85W-Cryo	1 min 1 min
2 Way Shuttle Valve (2)	28 VDC		None	200W-Cryo 200W-Cryo	0	200W-Cryo 200W-Cryo	1 min
Vent Valve (2)	28 VDC		None	200W-Cryo	0	200W-Cryo	1 min
Orbital Vent Valves (2)	28 VDC		None	85W-Cryo 85W-Cryo	0	85W-Cryo 85W-Cryo	10 min 10 min
Helium Purge S/O Sol. Valve (1)	28 VDC		None	85W-Cryo	0	85W-Cryo	30 min
Pressure Switch (2)	28 VDC		None	-	0	-	-
Stored Helium Sys Isolation Sol Valve (1)	28 VDC		None	85W-Cryo	0	85W-Cryo	200 min

Table 2.4-2. NR Tug EPS Input Data Sheet
LOX Tank Safing and Venting Subsystem

Subsystem -			Operational Phase -				
			Power Required (In Watts)				
Component	Voltage/ Frequency Required	Regulation Required (Percent)	Max Demand Inrush/ Peak	Operational Usage (Avg)	Standby Usage (Avg)	Emergency Power	Duration (Duty Cycle)
Relief Valve Solenoid (2)	28 VDC		None	85W-Cryo 85W-Cryo	0	85W-Cryo 85W-Cryo	1 min 1 min
2 Way Shuttle Valve (2)	28 VDC		None	200W-Cryo 200W-Cryo	0	200W-Cryo 200W-Cryo	1 min 1 min
Vent Valve (1)	28 VDC		None	200W-Cryo	0	200W-Cryo	1 min
Orbital Vent Valves (2)	28 VDC		None	85W-Cryo 85W-Cryo	0	85W-Cryo 85W-Cryo	10 min 10 min
Helium Purge S/O Valve (1)	28 VDC		None	85W-Cryo	0	85W-Cryo	30 min
Pressure Switch (2)	28 VDC		None	-	0	-	-
Insulation Purge S/O Valve (1)	28 VDC		None	45W-Cryo	0	45W-Cryo	180 min

A separate, motor driven shutoff valve (4), is provided in the vent subsystem for reducing the tank ullage pressure to near ambient pressure for tanking purposes. This valve will also be used for in orbit safing operations prior to re-entry. It is a 3-inch valve on the LH_2 subsystem and a 2-inch valve on the LOX subsystem.

The vent subsystem plumbing is designed to provide a dual function. When in the cargo bay, the subsystem is connected to a shuttle (orbiter) overboard vent subsystem through a disconnect (5). When out of the cargo bay, the vent subsystem is switched over to an on-board low flow non-propulsive vent subsystem through two 1-inch direct acting latching valves (6). The orbital venting subsystem discharges vented gases overboard through non-propulsive vent discharge nozzles.

The propellant tanks safing operations can only be accomplished when the Tug is in the orbiter cargo bay and the vent and fill and drain subsystems are attached to associated systems on the orbiter through disconnect interfaces. It is assumed that propellant settling by the orbiter has positioned all liquids at the top of the propellant tanks. The safing of the propellant tanks is accomplished as follows:

In either an abort or normal mode of operation for safing, first all liquids are dumped overboard through the fill and drain system (26), utilizing only vapor pressure. The fill and drain valves are kept open until the tank pressure is reduced to 1 psia or below. Then the fill and drain valves (26), are closed. The propellant tanks are then pressurized to 17 psia from a helium supply subsystem through the helium solenoid valve (8), controlled by a pressure switch (7), and orifice (9). The vent subsystem selector valve (1) is switched to vent the subsystem from the bottom of the propellant tanks and the tanks are blown down through the shutoff valve (4). The APS tank vent subsystem valve (11) is also opened to purge the APS tanks at the same time. The main propellant tanks are repressurized again to 17 psia prior to re-entry.

The helium supply subsystem is comprised of a bank of eight 6 cubic feet receivers (12), manifolded together. The helium supply subsystem is pressurized on the ground through disconnect (13) that is connected to an associated orbiter disconnect. The isolation solenoid valve (25) prevents inadvertent blowdown of the helium subsystem. Check valve (14) provides a redundant shutoff of the subsystem to prevent inadvertent blowdown of the subsystem in the event of a disconnect (13) malfunction. Pressure relief capability is provided in the helium supply subsystem.

The high pressure relief system is comprised of a 3000 psig relief valve (15) and a redundant check valve (16) to provide thermal relief for the system. The low pressure relief system protects the insulation repressurization system in the event of a regulator malfunction. This system is also comprised

of a relief valve (17) and a redundant check valve (18). Both relief systems discharge directly into the orbiter cargo bay.

The helium supply system is attached to the tug through two disconnects. Disconnect (19) provides for pressurizing the system on the ground and also ties into the propellant tanks for propellant tanks safing operations. Redundant check valves (21) are provided in the propellant tanks helium pressurization lines to prevent propellant contamination of the helium system. Disconnect (20) attaches the tug insulation repressurized system to the helium supply system. The helium system for repressurization of the insulation during re-entry is comprised of a solenoid shutoff valve (22), redundant check valve (23), and a two stage pressure regulator (24).

Circuit logic during safing operations will be such that the gas will not be turned on until the low-pressure pressure switch (10), closes the propellant tank vent valve and both the fill and drain valves. The fill and drain valves (26) will be opened for propellant liquids dump and for the first propellant tank blowdown.

When the low-pressure pressure switch (10) actuates, the fill and drain valves are closed and simultaneously the propellant tanks helium purge solenoids (8) will energize to allow pressurization of the propellant tanks. When the tank pressure switch (7) deactuates, the helium purge solenoid (8) shuts off the helium supply to the propellant tanks. The vent valves (4) will immediately open and propellant tank blowdown will occur.

When the low-pressure pressure switch (10), again actuates at the low pressure in the propellant tanks, the tank vent valves will close and repressurization of the propellant tanks will occur again. At the end of the propellant tanks repressurization when the tank pressure switch (7) closes the helium purge solenoid valve, tank safing will have been completed.

2.4.3 System Technical Discussion

a. Sizing Ground Vent System

Sizing of the ground vent system lines was based on a predicted boiloff rate of 666 lbs/hr on the LH_2 propellant tank and 478 lbs/hr on the LOX tank. Back pressure requirements in the LOX tank is 15.2 psia and 16.0 psia in the LH_2 tank. Using the relationship of

$$\Delta P = \frac{4fL}{D} \frac{V^2}{2g} \frac{\rho_{aver}}{144}$$

where

ΔP = pressure drop psi

f = friction factor

V = Velocity ft/sec

g = gravimetric constant 32.2 ft/sec-sec

ρ_{aver} = average density of fluid lbs/ft^3

L = equivalent length of tubing (ft)

D = diameter of tubing (ft)

it is determined that the back pressure during ground hold will be 15.1 psia in the LOX tank with a 2-inch vent line and 15.2 psia in the LH_2 tank using a 3-inch line.

b. Helium Requirements for Safing and Repressurizing

Table 2.4-3 shows a summary of all the stored helium required to safe and pressurize the tug prior to re-entry. In addition 24 lbs of stored helium is required for repressurization of the MLI (insulation) during re-entry. Helium requirements were computed based on the following relationship:

$$M_{He} = \frac{M_r \, CV_r \left(T_2 - T_1\right)}{CP_r \left(T_{He} - T_1\right) - CV_{He} \left(T_2 - T_1\right)}$$

where:

Mr = Mass residual gas in tank - lbs

CV_r = Specific heat at constant volume of residual gas. BTU/lb-F

CP_r = Specific heat at constant pressure of residual gas. BTU/lb - F

T_{He} = Initial temperature of stored helium - R

T_1 = Initial temperature of residual gas - R

T_2 = Final temperature of mixed gases - R

CV_{He} = Specific heat at constant volume of pressurant helium

Since P_2 was known, 17 psia, several T_2 were assumed until the proper P_2 was obtained using the following relationship:

$$P_2 = \frac{\left(R_r M_r + R_{He} \, M_{He}\right) T_2}{144 \, V}$$

where

P_2 = Final tank pressure after repress - 17 psia

R_r = Gas constant of residual gas - ft-lb/lb - R

Table 2.4-3. Tug Safing/Venting System

Blowdown Cycle		Helium Requirements for Safing								
		Helium Pressurant Req.				Concentration (Vol)				Total GHe
		LH_2 Tank		LOX Tank		GH_2		GOX		
Liquid Residuals		1st	2nd	1st	2nd	1st	2nd	1st	2nd	
LH_2 Tank	LOX Tank									
lbs	lbs	lbs	lbs	lbs	lbs	%	%	%	%	lbs
0	0	13.85	13.85	6.0	5.9	67.3	24.2	25.9	4.3	39.60
0	50	13.85	13.85	12.0	5.62	67.3	24.2	41.1	26.1	45.32
10	0	16.0	16.77	6.0	5.9	75.1	34.2	25.9	4.3	44.67
10	50	16.0	16.77	12.0	5.62	75.1	34.2	41.1	26.1	50.39

R_{He} = Gas constant of helium - 386 ft/lb - R

V = Volume of propellant - Ft3

The computation for determining residual gas concentrations after each blowdown and repressurization was based on the following relationship -

$$\%\ \text{Residual gas by volume} = \frac{\frac{M_r}{m_r}}{\frac{M_r}{m_r} + \frac{M_{He}}{m_{He}}}$$

where

m_r = Molecular weight residual gas

m_{He} = Molecular weight helium - 4.0

After the first blowdown and repressurization, the above relationships were modified to accommodate the fact that the residual gas was comprised of a mixture of propellant gases and helium. The quantity of each of the residual gases was determined by the following relationship:

$$N_m = \frac{(P_1\ 144)\ (V)}{(1544)\ \ (T_2)}$$

where:

N_m = mols remaining in the tank

P_1 = pressure in the tank - psia

m = molecular wt

then:

N_r = % Residual gas X N_m

N_{Her} = $N_m - N_r$

where:

N_r = Mols residual gas

N_{Her} = Mols residual helium in the tank

$M_r = m_r \times N_r$

$M_{Her} = m_{Her} \times M_{He}$

For the second repressurization:

$$M_{He2} = \frac{(M_r\ C_{Vr} + M_{Her}\ CV_{He})\ \ (T_2 - T_1)}{CP_{He}\ (T_{He} - T_1) - CV_{He}(T_2 - T_1)}$$

$$P_2 = \frac{\left[(R_{He})\ (M_{He2} + M_{Her}) + R_r M_r\right] T_2}{144V}$$

$$\%\ \text{Residual gas} = \frac{\frac{M_r}{m_r}}{\frac{M_r}{m_r} + \frac{M_{He2} + M_{Her}}{m_{He}}}$$

c. Evaluation for Helium Receiver Capacity

The basis of this analysis is the assumption that there will be 0 lbs residual liquids in the LH_2 propellant tank and 50 lbs of liquid residuals in the LOX propellant tank.

From Table 2.4-3:

LH_2 Tank Helium requirements	=	27.70 lbs
LOX Tank Helium requirements	=	17.62 lbs
Insulation Purge System	=	24.00 lbs
Total Helium Required		69.32 lbs

Assume residual pressure in the receiver bank of 300 psia.

Receivers will be pressurized to 3000 psia nom.

Then

$$M_2 = M_1 \left[\frac{P_2}{P_1}\right]^{\frac{1}{K}}$$

$$M_2 = 17.35 \text{ lbs}$$

$$M_{TOT} = 69.32 + 17.35 = 86.67 \text{ lbs}$$

where

M_2 = Mass of residual helium remaining in the receivers - lbs

M_1 = Mass of helium required - lbs

M_{TOT} = Total helium storage required - lbs

Receiver size:

$$V_T = \frac{ZM_T R_{He} T_{He}}{144 P_{rec.}} = 45.5 \text{ ft}^3$$

where:

V_T = Receiver volume reqd. - ft^3

Z = Compressibility Factor (Dimensionless)

P_{rec} = Press. in receiver - psia (3000 psia)

T_{He} = Final Temperature in receiver - °R (530°R)

Install eight - 6 ft^3 receivers = 48 ft^3

$$\% \text{ Reserve} = \frac{48 - 45.5}{48} = 5\%$$

The choice of 6 ft^3 receivers was made based on the ease of installation and the better weight distribution in the support structure.

d. Helium Receiver Pressure Determinations after Each Blowdown

Initial Helium Available:

$$M_{He\ TOT} = \frac{VP}{ZRT} = 91.6 \text{ lbs}$$

Using the following relationship

$$P_2 = P_1 \left[\frac{M_2}{M_1}\right]^K$$

the following tabulation was computed

	M_1 lbs	M_2 lbs	P_1 psia	P_2 psia
1st Repress	91.6	65.75	3000	1730
2nd Repress	65.75	46.28	1730	965
Insul. Purge	46.28	22.28	965	288

e. Propellant Tanks Safing Times Evaluations Under Abort Conditions

In the following computations only average temp, pressures, flowrates, etc., were used and only end points computed. No attmpet was made to describe transient profiles.

Prior to inerting the propellant tanks it is necessary to dump all of the liquids in the propellant tanks. All these computations are based on the assumption that the Tug is in the payload bay of the orbiter and that all liquids have been settled to the forward end of the propellant tanks.

(1) Propellant Liquid Dump Using Ullage Pressure. Liquid dump is accomplished using ullage pressure only without the use of any additional makeup gases. Figure 2.4-3 shows the relationship between ullage pressure remaining in the propellant tanks vs. propellant remaining. The development of Figure 2.4-3 was based on the following assumptions:

1. Liquid and gases in the propellant tanks are at saturation.
2. Adiabatic process.
3. Dumping 100 percent liquid (no two phase flow)
4. Initial internal energy of the tank fluid minus the mean enthalphy of the dumped fluid is equal to the internal energy of the residual fluid.

$$E_1 - \bar{H} = E_2 \qquad \text{Eq. (1)}$$

5. No contaminant fluids or pressurants in the tank.

The following relation develops from equation (1).

$$\left\{\frac{V_L}{V}\left[\rho_L (e_L - \bar{h}) - \rho_\sim (e_\sim - \bar{h})\right] + \rho_\sim (e_\sim - \bar{h})\right\}_{1}^{2} = 0 \qquad \text{Eq. (2)}$$

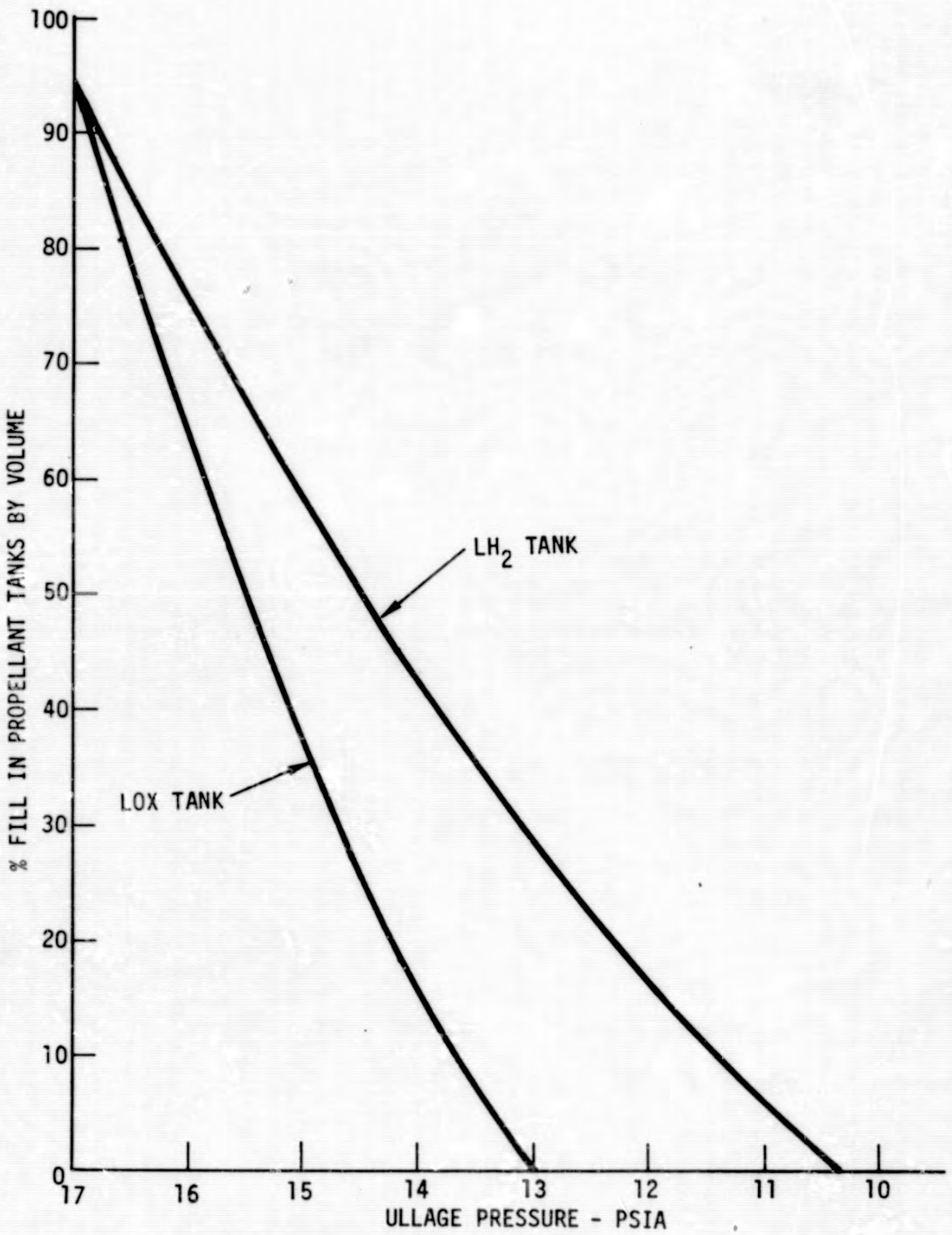

Figure 2.4-3 Propellant Tanks - Liquid Dump Ullage Pressure Profile Utilizing Only Vapor Pressure

Limitations of the above equation

$$0 \leq \frac{V_{L1}}{V} \leq 1 \geq \frac{V_{L2}}{V} \geq 0$$

For Full Tank Dump

Let $V_{L1}/V = 1$ and solve for V_{L2}/V. Then eq. (1) becomes:

$$\frac{V_{L2}}{V} = \frac{\rho_{L1}\,(e_{L1} - \overline{h}) - \rho_{\sim 1}\,(e_{\sim 1} - \overline{h})}{\rho_{L2}\,(e_{L2} - \overline{h}) - \rho_{\sim 2}\,(e_{\sim 2} - \overline{h})} \qquad \text{Eq. (3)}$$

For Empty Tank Fill

Let $V_{L1}/V = 0$ and solve for V_{L2}/V. Then eq. (1)becomes:

$$\frac{V_{L2}}{V} = \frac{\rho_{\sim 2}\,(e_{\sim 2} - \overline{h}) - \rho_{\sim 1}\,(e_{\sim 1} - \overline{h})}{\rho_{\sim 2}\,(e_{\sim 2} - \overline{h}) - \rho_{L2}\,(e_{L2} - \overline{h})} \qquad \text{Eq. (4)}$$

For 95% Full Tank Dump

Let $V_{L1}/V = 0.95$. Then eq. (1) becomes:

$$\frac{V_{L2}}{V} = \frac{0.95\left[\rho_{L1}\,(e_{L1} - \overline{h}) - \rho_{\sim 1}\,(e_{\sim 1} - \overline{h})\right] + \left[\rho_{\sim 1}\,(e_{\sim 1} - \overline{h}) - \rho_{\sim 2}\,(e_{\sim 1} - \overline{h})\right]}{\left[\rho_{L2}\,(e_{L2} - \overline{h}) - \rho_{\sim 2}\,(e_{\sim 2} - \overline{h})\right]}.$$

where:

ρ = Density (Consistent units)

e = Internal Energy } Consistent Units

$\overline{h} = \dfrac{h_1 + h_2}{2}$ Mean Enthalpy }

V_L = Volume of Initial Liquid Mass

V = Total Volume of Tank

V_L/V = Dimensionless Volumetric Ratio

Subscripts:

1 - Initial Condition

2 - Final Condition

L - Liquid

v - Vapor

① Condition of Initial Fluids

② Condition of Residual Fluids

Bracketed Expression Evaluated @ ② minus ①

(2) Propellant Liquid Dump Times. Liquid dump is accomplished through the fill and drain lines. Using the ΔP shown in Figures2.4-3 for propellant dump computations and determining line losses from layout drawings ($K = 4fL/D$). The following equation was used to compute average liquid flow rate (W) during dumping operations.

$$\dot{W} = A \left[\frac{2g\rho\Delta P}{144K} \right]^{\frac{1}{2}}$$

where

$\dot{W}$ = Dump Flowrate lbs/sec

g = gravimetric constant 32.2 ft/sec-sec

ΔP = system pressure drop - psia

K = Line loss constant - dimensionless - 4f L/D.

A = Cross-Sectional Area of Line - inches^2

then

W_{LH_2} = 6.35 lbs/second average

W_{LOX} = 30.5 lbs/second average

$t_{L_{dump}} = M_L/W$

where

$t_{L_{dump}}$ = dump time - liquids - seconds

M_L = Mass Liquid - lbs

For LH_2 tank:

$$t_{dump_{LH_2}} \doteq \frac{8105}{6.35} = 21.25 \text{ min.}$$

For LOX Tank:

$$t_{dump_{LOX}} = \frac{48633}{30.5} = 26.7 \text{ min.}$$

(3) After liquid dumping operations are completed the gases continue to blow down through the fill and drain system to 1 psia. Time for ullage gases blowdown computed as follows:

$$t_r = \frac{2V\left[\left(\frac{P_f}{Pi}\right)^{\frac{1-K}{2K}} - 1\right]}{(K-1)\ (C_dA)\ B\sqrt{R\ Ti}}$$

where

t_r = Time for blowdown - sec

V = Tank Volume - ft^3

P_f = Final Tank Pressure - 1 psia

Pi = Initial Tank Pressure - psia

K = Specific heats ratio - Dimensionless

C_dA = Effective Flow Area

R = Gas Constant - ft-lb/lb°R

Ti = Initial Temp - °R

$$B = \sqrt{Kg\left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}}$$

For 1st Blowdown

GH_2 Blowdown Time = 2.23 min.

GOX Blowdown Time = 2.06 min.

(4) Repressurization Time. Time to repressurize the tank to 17 psia after the first blowdown cycle was computed as follows: The repressurization of the propellant tanks is accomplished through the repress solenoid and orifice. The orifice I.D. = 0.1 inch. Line losses were determined from layout drawings (K = 4 fL/D). Flow through the orifice was computed using the following relationships:

$$\dot{W} = \frac{0.165\, C_d D^2\, Pu}{\sqrt{Ti}}$$

where:

$\dot{W}$ = Flowrate lbs/sec

C_d = Coefficient of discharge - dimensionless

D = diameter of orifice - inches

Pu = Pressure upstream orifice - psia

Ti = Initial temperature of pressurant - °R

Several test cases were compiled assuming different P_u and a curve was plotted of ullage pressure vs. flowrate. The flowrates computed above were tested in the line pressure drop equation:

$$\Delta P = \frac{144}{2g\rho} (K) \left[\frac{\dot{W}}{A}\right]^{\frac{1}{2}}$$

and a cross plot was made against the orifice flow curve. Where the two curves crossed is the balanced flow through the system. This average flowrate was 0.0628 lbs/sec. This flowrate holds true for both propellant tanks.

Time to repressurize - 1st cycle

LH_2 Tank = $M_{He}/\dot{W}$ = 3.67 min.

LOX Tank = $M_{He}/\dot{W}$ = 3.18 min.

(5) Second Propellant Tanks Blowdown. The second propellant tanks blowdown time was computed using the same equation as was used for the first blowdown,

$$t_r = \frac{2V\left[\left(\frac{P_f}{Pi}\right)^{\frac{1-K}{2K}} - 1\right]}{(K-1)\,(C_dA)\; B\sqrt{R\,Ti}}$$

except a new (R) and (B) had to be computed to account for the mixed gases in the tank.

Tank blowdown times 2nd cycles

LH_2 Tank = 2.22 min.

LOX Tank = 2.86 min.

(6) 2nd Propellant Tanks Repressurization. The second repressurization flowrate was computed in the same manner as was used for the first repressurization time using the lower helium receiver pressure available. This pressurant flowrate was 0.0412 lbs/sec.

2nd repressurization time

LH_2 Tank = 5.61 minutes

LOX Tank = 3.75 minutes

(7) Total tank required to safe propellant tanks in an abort condition.

LH_2 Tank = 34.98 minutes

LOX Tank = 38.55 minutes

Table 2.4-4 is a tabulation of all the delta times required for safing the propellant tanks under an abort condition.

f. Propellant Tanks Safing Times Evaluations for a Normal Mode

Safing times under normal conditions prior to reentry were computed in same manner as was used for the abort mode. Liquid residuals in the tanks were assumed to be 5 percent of original propellant load.

Total time for Safing -

LH_2 Tank = 14.45 minutes

LOX Tank = 13.36 minutes

Table 2.4-4 is a tabulation of all the delta times required for safing the propellant tanks under normal safing conditions.

Table 2.4-4. Safing Times - Based 0 lbs LH_2 and 50 lbs LOX Residuals (After dump)

		Liquid Dump Minutes	1st Blowdown Minutes	1st Repress Minutes	2nd Blowdown Minutes	2nd Repress Minutes	Total Time Minutes
Abort	LH_2 TK	21.25	2.23	3.67	2.22	5.61	34.98
Cycle	LOX TK	26.70	2.06	3.18	2.86	3.75	38.55
Normal	LH_2 TK	0.09	2.86	3.67	2.22	5.61	14.45
Cycle	LOX TK	1.25	2.32	3.18	2.86	3.75	13.36
Tank Ullage	LH_2 TK	36	25	287	92	552	
Temp °R	LOX TK	159	127	222	94	539	

g. Effect of Liquid Quality on Liquid Dump Times

All liquid dump times shown in this report were based on dumping liquid only. Figures 2.4-4 and 2.4-5 show the effect of vapor quality at the exit has on liquid dump times. Both figures show liquid dump times for both 2-inch and 3-inch lines to illustrate the effect of both line size and vapor quality. The curves clearly show that small changes in vapor quality are reflected in large increases in liquid dump times.

h. System Weight

Table 2.4-5 is a compilation of all of the detail system elements weights. Total system weight is 1010.75 lbs. This weight is based on components that will be available in 1976.

i. Reentry Venting

Since actual heating rates during reentry are not known, the effect of self-pressurization during reentry was not computed. Venting during reentry is almost certain. Putting the relief valves in high mode of operation of 23 - 24 psia would reduce the amount of venting during reentry.

j. Propellant Liquid Dump

A separate propellant liquid dump evaluation during flight to orbit was performed. Table 2.4-6 shows a summary of propellant liquid dump times vs. line size. The basis for this evaluation was that 3/4 of the propellant liquid would be dumped. To decrease the dump time, the tanks will be pressurized. The LH_2 ullage pressure was held constant at 23 psia and the LOX ullage pressure was held constant at 16 psia, by the use of makeup pressurant from the stored helium supply bank.

The weight penalty for increasing the flow capacity of the propellant liquid dump system is shown in Table 2.4-7.

k. Additional Purging of the Hydrogen Propellant Tank

One more blowdown-repressurization cycle can be performed on the LH_2 propellant tank if it is decided not to perform a purge of the LOX propellant tank. 17.62 lb of helium purge gas are required for the LOX tank purge and safe sequence. One more pressure-vent cycle on the LH_2 propellant tank would require 15.15 lbs of helium and the concentration of the hydrogen in the LH_2 propellant tank would be reduced to 5 percent from 24.2 percent after the second cycle. The LOX could still be dumped using only vapor pressure and the tank allowed to self pressurize back to 17 psia from 13 psia the pressure that the LOX tank would drop to during liquid dump operations.

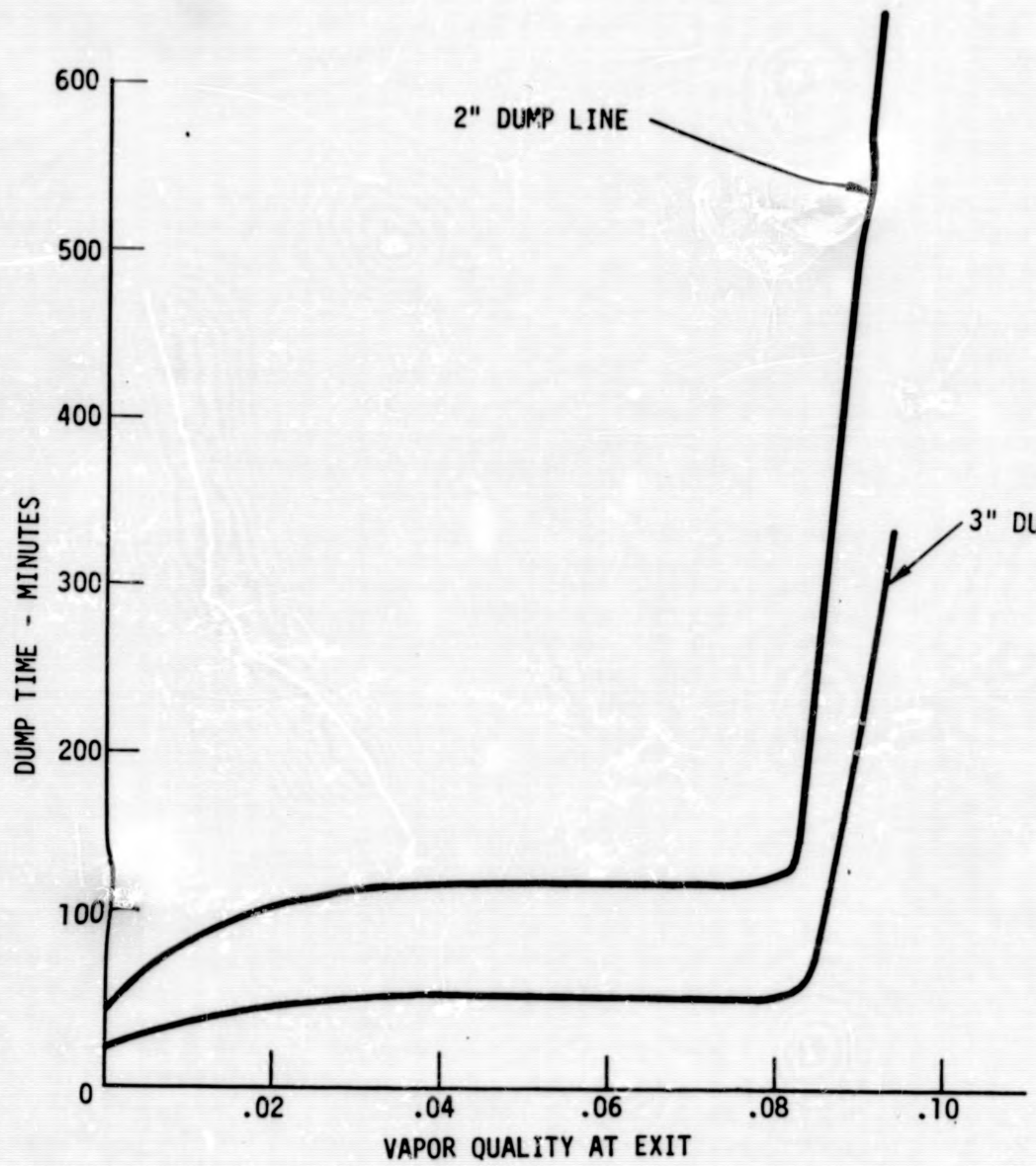

Figure 2.4-4 Effect of 2 Phase Flow on LH_2 Liquid Dump Time

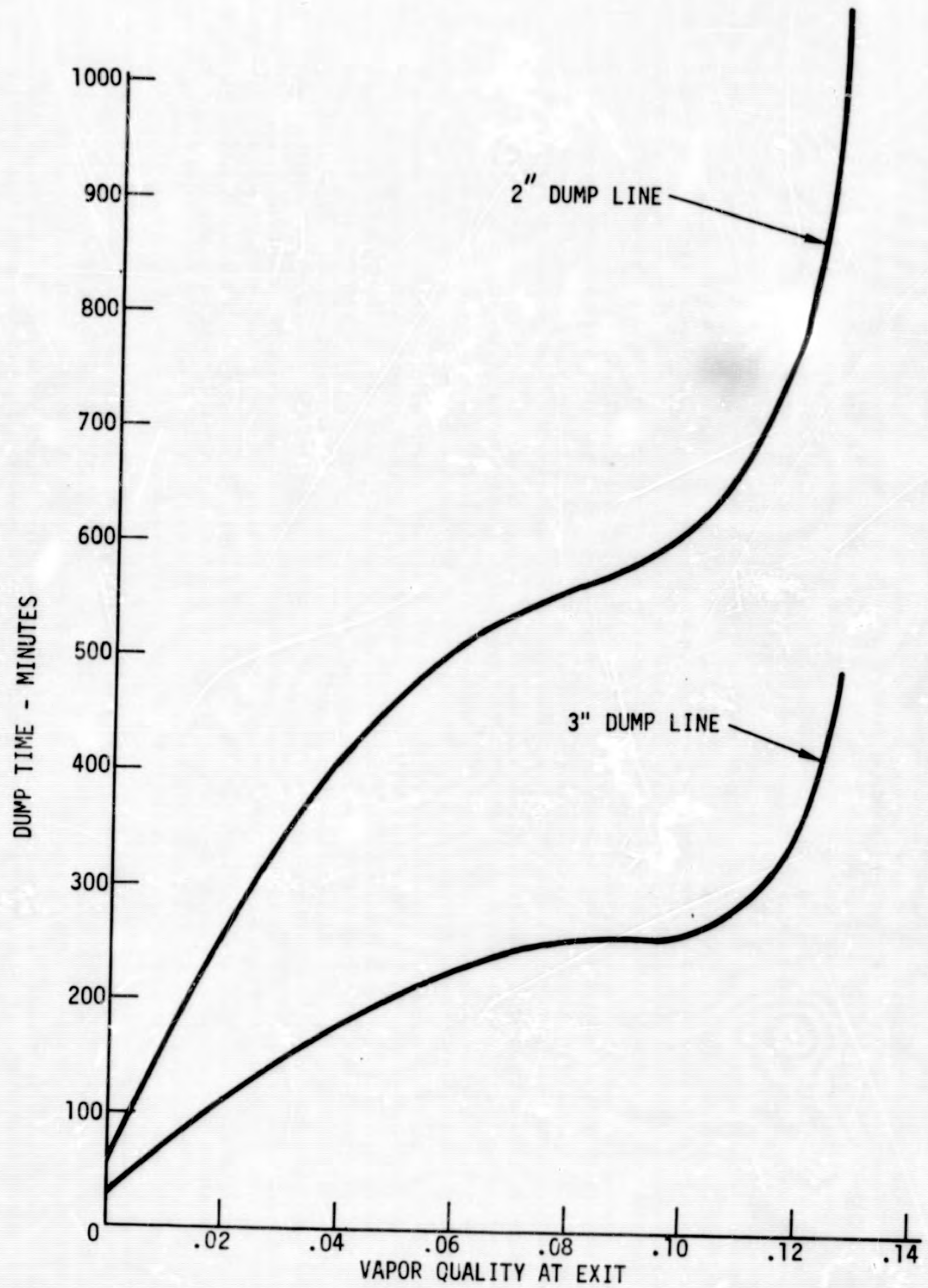

Figure 2.4-5 Effect of 2 Phase Flow on LOX Liquid Dump Time

Table 2.4-5. Tug Safe/Venting System Weight

Sch. No.	Description	Unit Weight lbs	No. Units	Total Weight lbs
1	Shuttle Valves	10	4	40.0
2	Relief Valves	10	4	40.0
3	Relief Valve Solenoid	*	-	-
4	Vent Valve	8	2	16.0
5	Vent Disconnect	4	2	8.0
6	Orbital Vent Valve	2.25	4	9.0
7	17 psia Pressure Sw.	1.50	2	3.0
8	Helium S.O. Solenoid Vlv.	2.25	2	4.5
9	Helium System Orifice	0.75	2	1.5
10	1 psi Pressure Switch	1.75	2	3.5
11	APS System S.O. Sol. Vlv.	2.0	1	2.0
12	Helium Receivers	76.5	8	612.0
13	Helium Sys. Disconnect	2.5	1	2.5
14	Helium Sys. Check Valve	0.75	1	0.75
15	3250 psi Relief Valve	0.75	1	0.75
16	3250 psi Check Valve	0.75	1	0.75
17	Low Pressure Relief Vlv.	0.75	1	0.75
18	Low Pressure Check Vlv.	0.75	1	0.75
19	Orbiter Helium Disconnect	1.75	1	1.75
20	Insulation Purge Disc.	1.75	1	1.75
21	Helium Purge Check Vlv.	0.75	2	1.50
22	Insul. Purge Isol. S.V.	2.25	1	2.25
23	Insul. Purge Check Vlv.	0.75	1	0.75
24	Insul. Purge Regulator	4.00	1	4.00
25	Helium Sys. Isol. S.V.	2.25	1	2.25
26	Fill and Drain Valves	7.50	Ref.	-
	Insulation Purge Sys	-	-	68.0
	Tank Purge Sys			18.75
	Stored Helium Sys.			153.00
	Tank Vent Sys.			11.00
	*An integral part of relief valve			
	Total Syst Wt			1010.75

Table 2.4-6. Summary of Propellant Liquid Dump Times vs Line Size

Dump Time - Minutes			
Propellant Tank	Line Size		
	3"	6"	7"
LH_2	9.74 min	2.37 min	1.74 min
LOX	18.5 min	4.18 min	3.24 min

Table 2.4-7. Weight Penalty for Increasing the Flow Capacity

Weight Penalty for Larger Capacity System		
	Wt Penalty	
	Line Size	
Propellant Tank	6"	7"
LH_2	36 lbs	48.6 lbs
LOX	31 lbs	43.6 lbs
Total Wt Penalty	67 lbs	92.2 lbs

2.4.4 Evaluative Comments

The propellant tanks safing/venting system can meet all of the ground rules established for this study. In several areas design compromises were made in the interest of weight saving that might require reconsideration as the Tug design becomes more firm. The ground rule decision to provide propellant liquid dump capability only while in the shuttle payload bay limits the safing options in orbit and should be evaluated further. Use of the fill and drain system for propellant liquid dump in orbit imposes the more severe flight requirements on a system that ordinarily only needs to meet ground requirements. However, this choice does provide the simpler and lighter weight system. GH_2 concentration level in LH_2 propellant tank after safing as defined in the guidelines does not meet the safety requirement of 4 percent for a completely safe tank.

Additional consideration should be given to the actual requirements for safing the propellant tanks. Possibly the LOX tank should require only a liquid dump cycle and the helium provided for safing the LOX tank used to provide an additional cycle in the hydrogen tank. Tank safing can be improved by long hold periods with tank vents open. Residual gases in the tanks are a function of how long gases are allowed to diffuse to outside space environment. Dumping of liquid residuals outside the orbiter payload bay should be reconsidered to provide more flexibility to stage operations.

2.5 PRESSURIZATION SUBSYSTEM

2.5.1 Function

The pressurization subsystem is required for providing a means of supplying and controlling pressurizing gases to the propellant storage tanks for both main engine and APS turbopumps start and run requirements.

2.5.2 Description and Operation

A. Subsystem Description

The pressurization subsystem selected is shown schematically on Figure 2.5-1 with installation details covered on Figure 2.5-2. The main tank vent valves will be closed after propellant loading and topping approximately 30 seconds prior to launch. The tanks will be allowed to self-pressurize to the main tank relief valve low mode operating bands of 15 to 16 psia for the LOX tank and 16 to 17 psia for the LH_2 tank.

In case of an on-the-pad abort that requires propellant off-loading, helium pressurant will be supplied to the main tanks from the Safing and Venting System's inerting and purging helium supply. The main tank pressures will be limited to a 17 psia maximum for both LOX and LH_2 tanks by pressure switches which are components of the Safing and Venting System.

At Shuttle launch, the LH_2 main tank relief valves will be switched to the high mode operating bands of 23 to 24 psia. The LOX main tank valves will remain in the low mode until the orbiter has achieved orbit, at which time they will be switched to the high mode operating band of 23 to 24 psia.

After the shuttle orbiter has achieved orbit, the Tug may be required to dump propellants in case of an orbiter abort. This dump will be done using vapor pressure, thereby retaining capability for inerting the propellant tanks with the available helium supply. At present, there is no requirement to dump propellants during Shuttle boost.

Prior to a main engine firing, the APS thrusters will be fired to settle propellants. Once the propellants have been settled, the main engine will start idle mode operation. Main tank pressurization is not required for idle mode operation. After initiation of main engine idle mode operation and approximately 105 seconds prior to mainstage operation, both main tanks will be pressurized to the regulator bands of 21 to 22 psia by opening the pressurization solenoid valves. The pressurant to each tank is supplied from the APS accumulators. Pressurant supply conditions will be within a pressure range of 575 to 1250 psia for both main tanks and at temperatures of 400°R for the LOX tank and 200°R for the LH_2 tank. The pressurant flow will be controlled by the pressurization regulators. Approximately 5 seconds prior to mainstage operation, the prepressurization solenoid valves will be closed, thus isolating the APS from the high pressure pressurant supply that exists during mainstage operation.

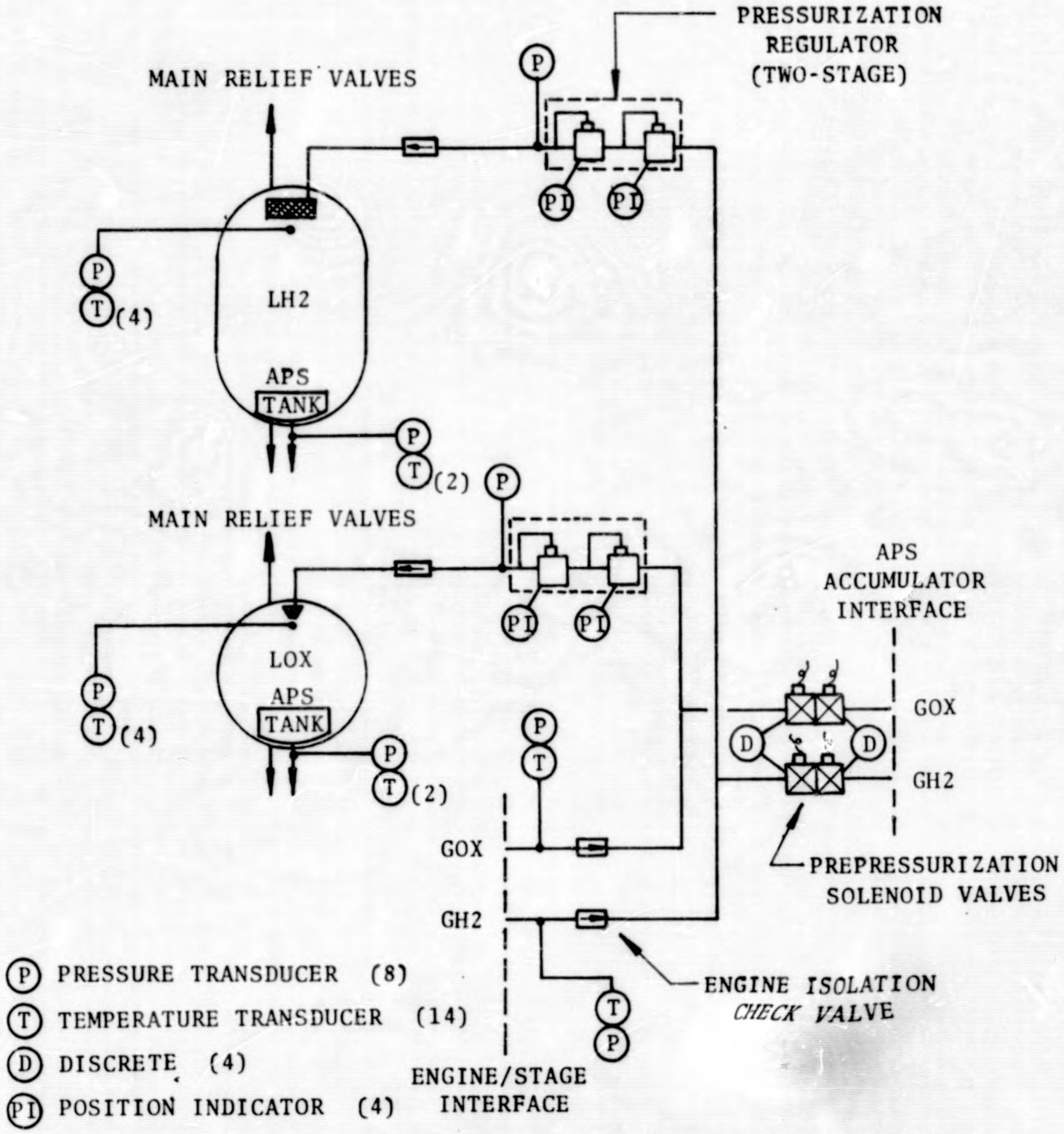

Figure 2.5-1 Pressurization System Schematic

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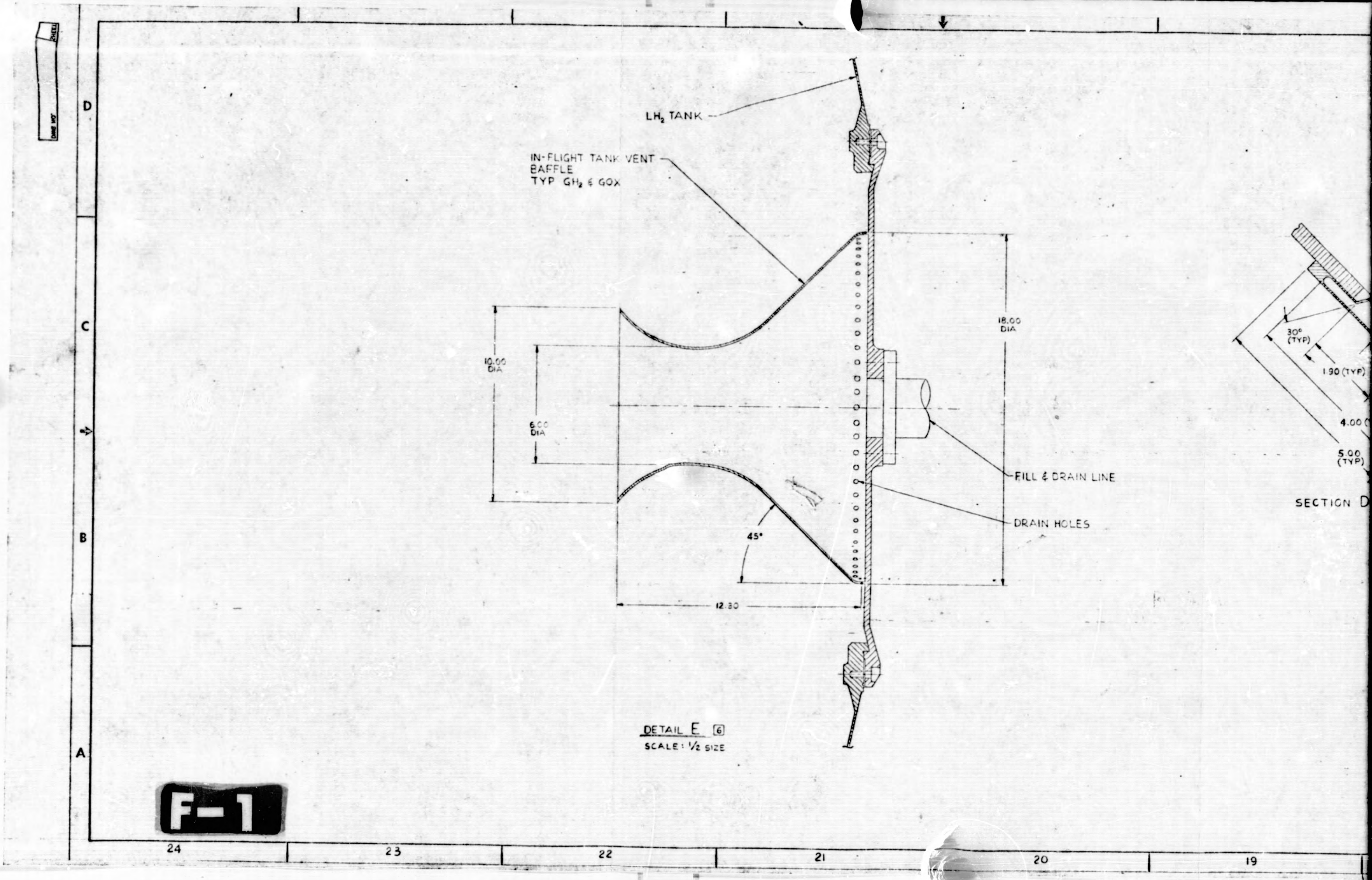
LH2 TANK
IN-FLIGHT TANK VENT
BAFFLE
TYP GH2 & GOX
18.00
DIA
10.00
DIA
6.00
DIA
FILL & DRAIN LINE
DRAIN HOLES
45°
12.30
30°
(TYP)
1.90 (TYP)
4.00
5.00
(TYP)
SECTION D
DETAIL E 6
SCALE: 1/2 SIZE
F-1
D
C
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A
24
23
22
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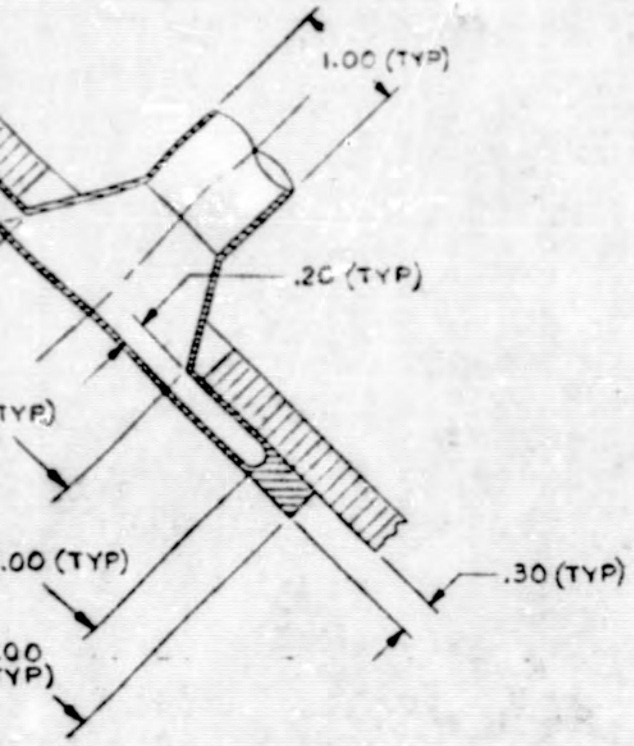
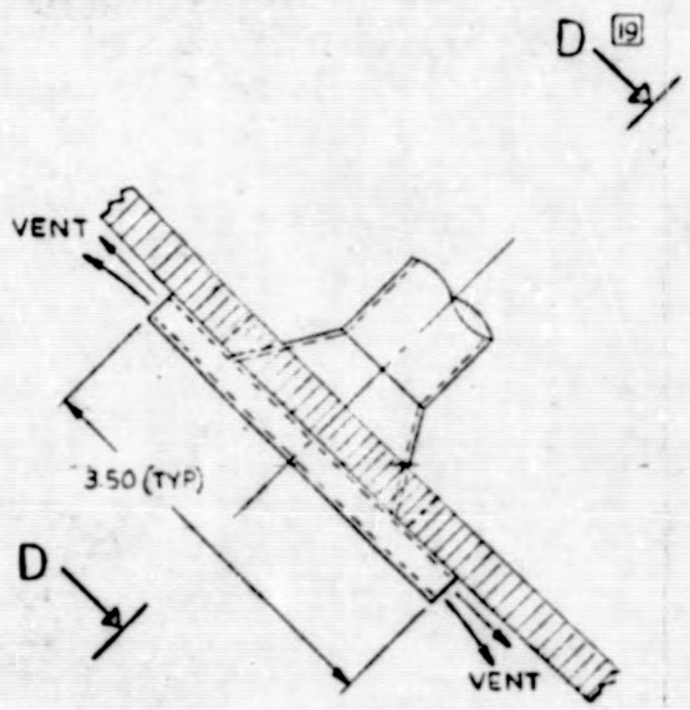
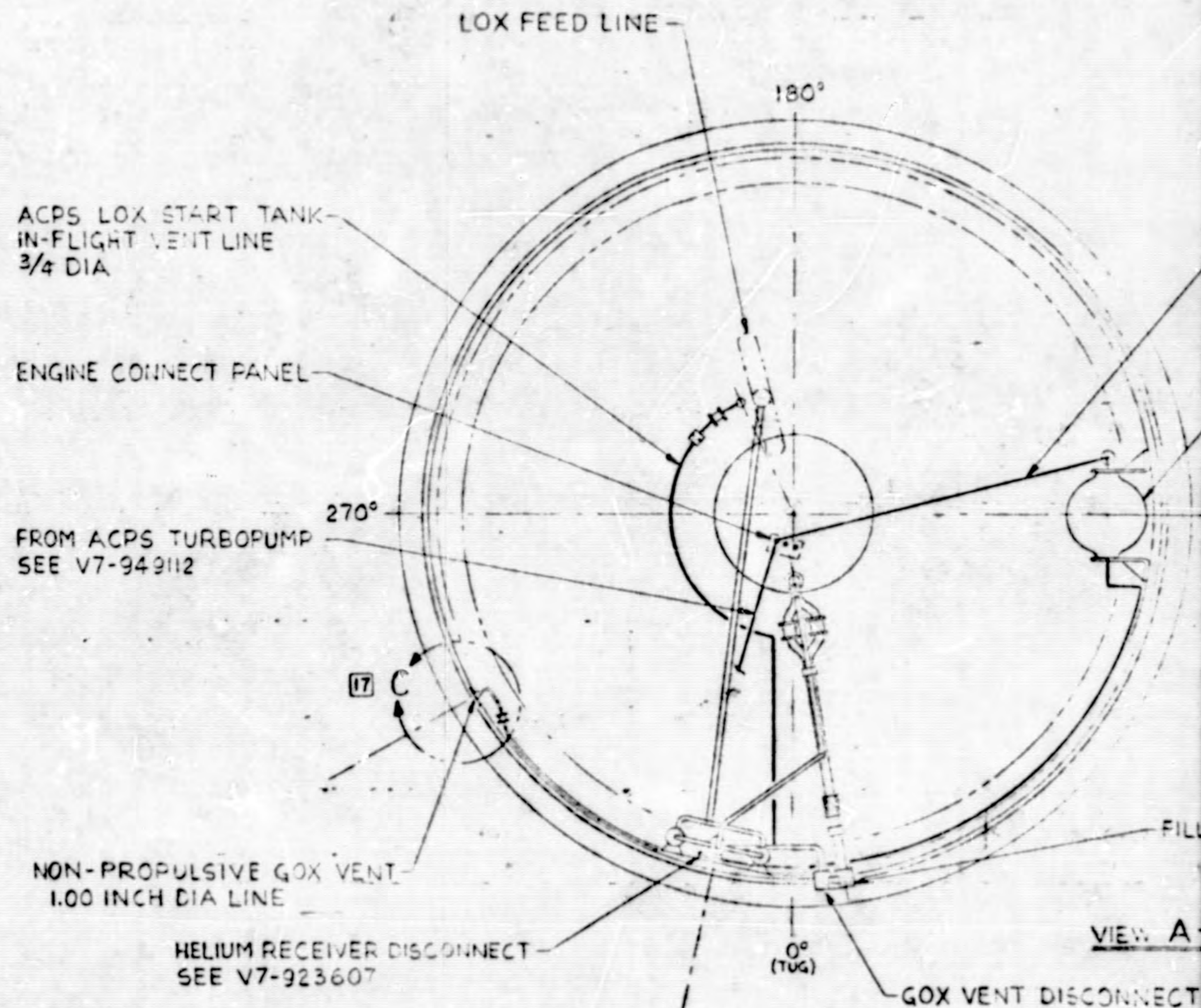
19



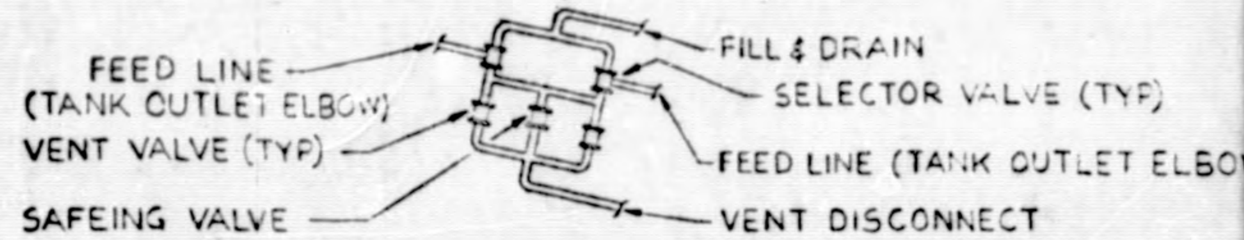
N D-D 16



DETAIL C 14
SCALE: 1/1
TYP NON-PROPULSIVE VENT



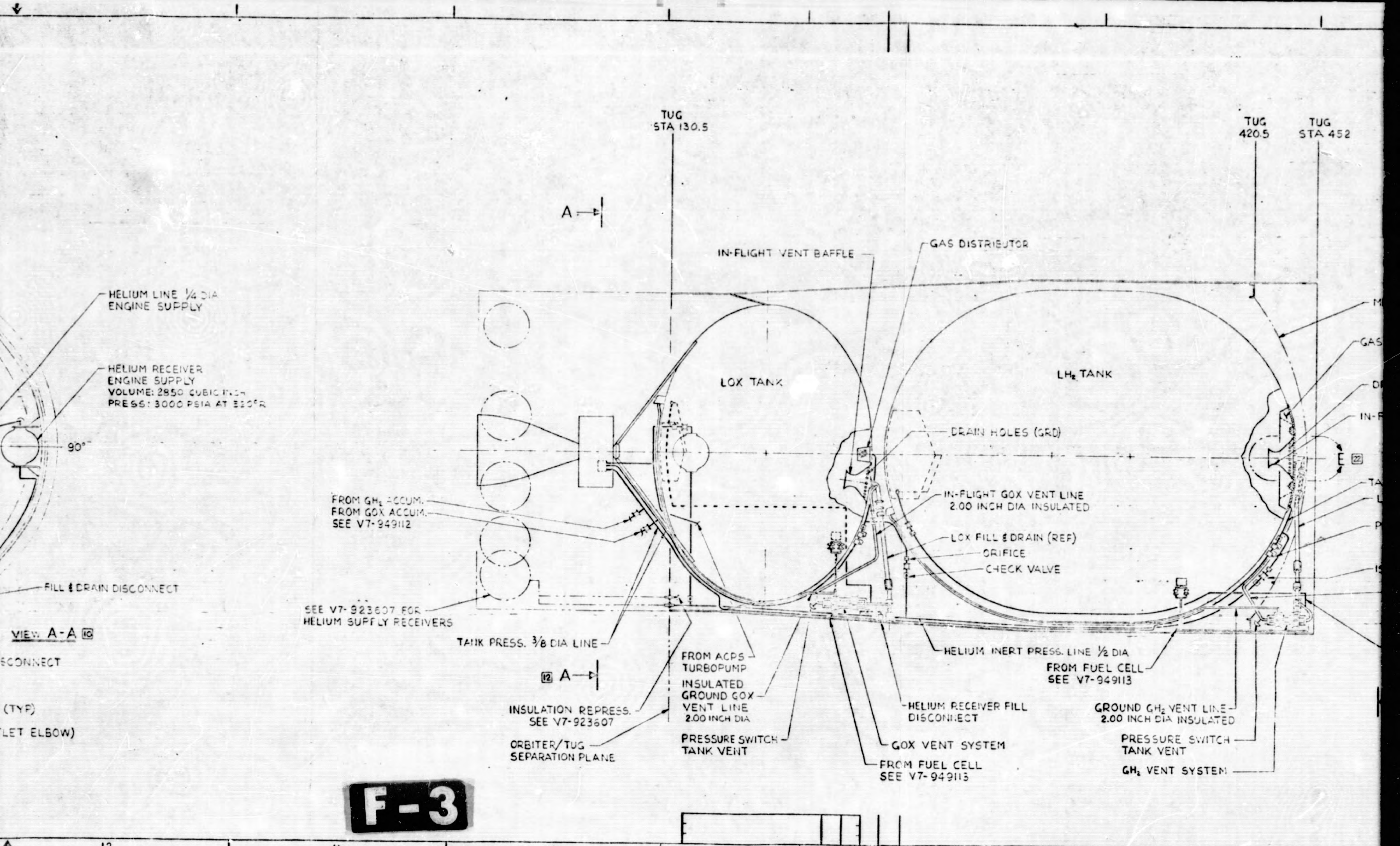
VIEW A



TYP VENT SYSTEM

F-2

18 17 16 15 14 13



F-3

12 11 10 9 8 7

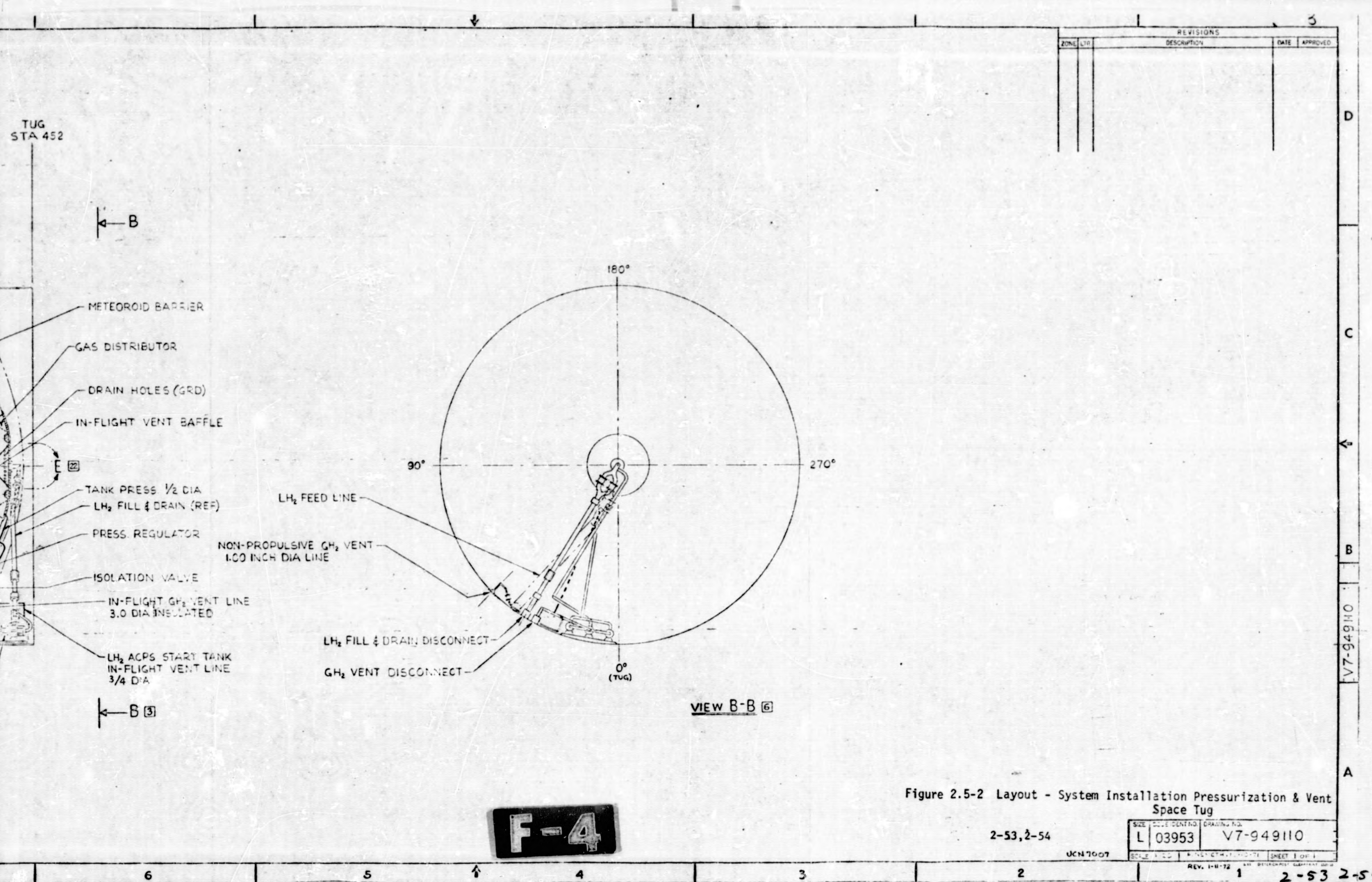


Figure 2.5-2 Layout - System Installation Pressurization & Vent
Space Tug

2-53, 2-54

Mainstage pressurant supply will be available from the main engine. The GOX pressurant will be supplied from the engine heat exchanger in the approximate pressure range of 3000 to 3500 psia at a temperature of 600°R. The GH_2 pressurant will be supplied from the thrust chamber jacket in the pressure range of 2000 to 2500 psia at a temperature of 260°R. This temperature was selected in order to take advantage of a structural weight savings. Again, the pressurant flow will be controlled by the pressurization regulators. Engine isolation check valves are incorporated into the pressurization lines.

The APS turbopumps will have zero NPSH inducers. Therefore, the APS screened tanks will not require pressurization above the existing vapor pressure.

Main tank venting requirements during orbital coast periods will be handled by propellant settling and gas venting. At present, both tanks will be vented just prior to the main engine burn following the 74 hour coast period. This venting will reduce the vapor pressure of both propellants and any further venting is not anticipated. The venting will be accomplished by firing the APS thrusters to settle propellants and switching the relief valves from the high to the low mode operating bands. After the relief valves have reseated, the valves will be switched back to the high mode.

The components required for the pressurization system are presented in Table 2.5-1. The component sizes and weights are also shown in the Table.

Table 2.5-1. Pressurization System Components

Component/Line (No. Required)	Size (in. Dia.)		Weight (ea.) (lb.)	
	LOX	LH_2	LOX	LH_2
Pressurization regulator (1) Inlet -Outlet	3/8 1/2	3/8 1/2	7	7
High pressure pressurization line (1) - S.S.	3/8	3/8	12	16
Low pressure pressurization line (1) - Al.	1/2	1/2	1	1
Prepressurization solenoid valves - latching type (2)	3/8	3/8	2¼	2¼
Engine isolation check valves (1)	3/8	3/8	1/4	1/4
Regulator out check valves (1)	1/2	1/2	1/4	1/4
Pressurization gas distributor (1)	N/A	N/A	2	2
	Total (lb)		27	31

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System operational checkout is limited to both stages of each pressurization regulator and the prepressurization solenoid valves. Both LOX and LH_2 tank two-stage pressurization regulators will be checked out by applying pressure to the checkout lines and monitoring the poppet position indicators. Both stages of each regulator will be checked out in this manner. The prepressurization solenoid valves will be checked out by actuating the valves while monitoring the valve discreets. There are no other active components in the pressurization system that require checkout.

B. System Performance

The selected MPS tank pressure levels are presented on Figure 2.5-3. The margins presented in this figure show that maximum propellant vapor pressures of 19 psia for the LOX and 19.5 psia for the LH_2 are acceptable with the present component bands.

In a low gravity environment, the control of propellant tank pressure and/or bulk liquid temperature is complicated by the lack of retention of the liquid in a known position. Over a prolonged period of orbital coast, the cryogenic propellant tank will absorb quantities of heat such that when protected only by a passive means such as multi-layered insulation, there will result a rise in the *bulk* liquid temperature. For the Tug LH_2 propellant tank, over the entire baseline mission, this temperature rise would be approximately 2.5°R, equivalent to an increase of approximately 7 psi saturation pressure.

On earth, or during powered flight, vapor can be vented to limit the pressure rise, for the liquid will occupy a known position. But at low gravity, when the location of liquid is unknown, direct overboard venting could result in the venting of liquid, or of a mixture of liquid and vapor. Propellant losses could be excessive.

Aside from control of tank pressure and/or bulk liquid temperature by direct overboard venting, several concepts have been considered by various investigators. An appraisal of six systems was performed by General Dynamics/Convair (Ref. 2.5-1). Of those systems appraised, a thermodynamic vent heat exchange system was adjudged to be the most promising. Mechanical separators were also promising, but have the drawback of being ineffective for large fill levels, or whenever the fluid fed to the separator is mostly liquid.

The systems considered for control of internal tank thermodynamics include the thermodynamic vent heat exchange system and direct overboard venting.

Thermodynamic Vent Systems. The thermodynamic vent heat exchanger system includes an expansion valve where LH_2 is throttled to a low pressure and temperature. This provides the necessary temperature differential for extraction of heat from the propellant by an appropriate heat exchanger. The heat of vaporization provides the necessary heat sink. Conservation of propellant is assured in that overboard venting of fluid can be limited to vapor only.

	LOX	LH_2
VAPOR PRESSURE (PSIA)	15.0	16.0
MARGIN (PSI)	4.0	3.5
NPSP (PSI)	1.0	0.5
HEAD (PSI)	~ 0	~ 0
FRICTION (PSI)	0.6	0.4
REG BAND (PSI)	1.0	1.0
DEAD BAND (PSI)	1.0	1.0
RELIEF BAND (PSI)	1.0	1.0
TOTAL (PSIA)	23.6	23.4
TANK MAX OPERATING PRESSURE (PSIA)	24.0	24.0

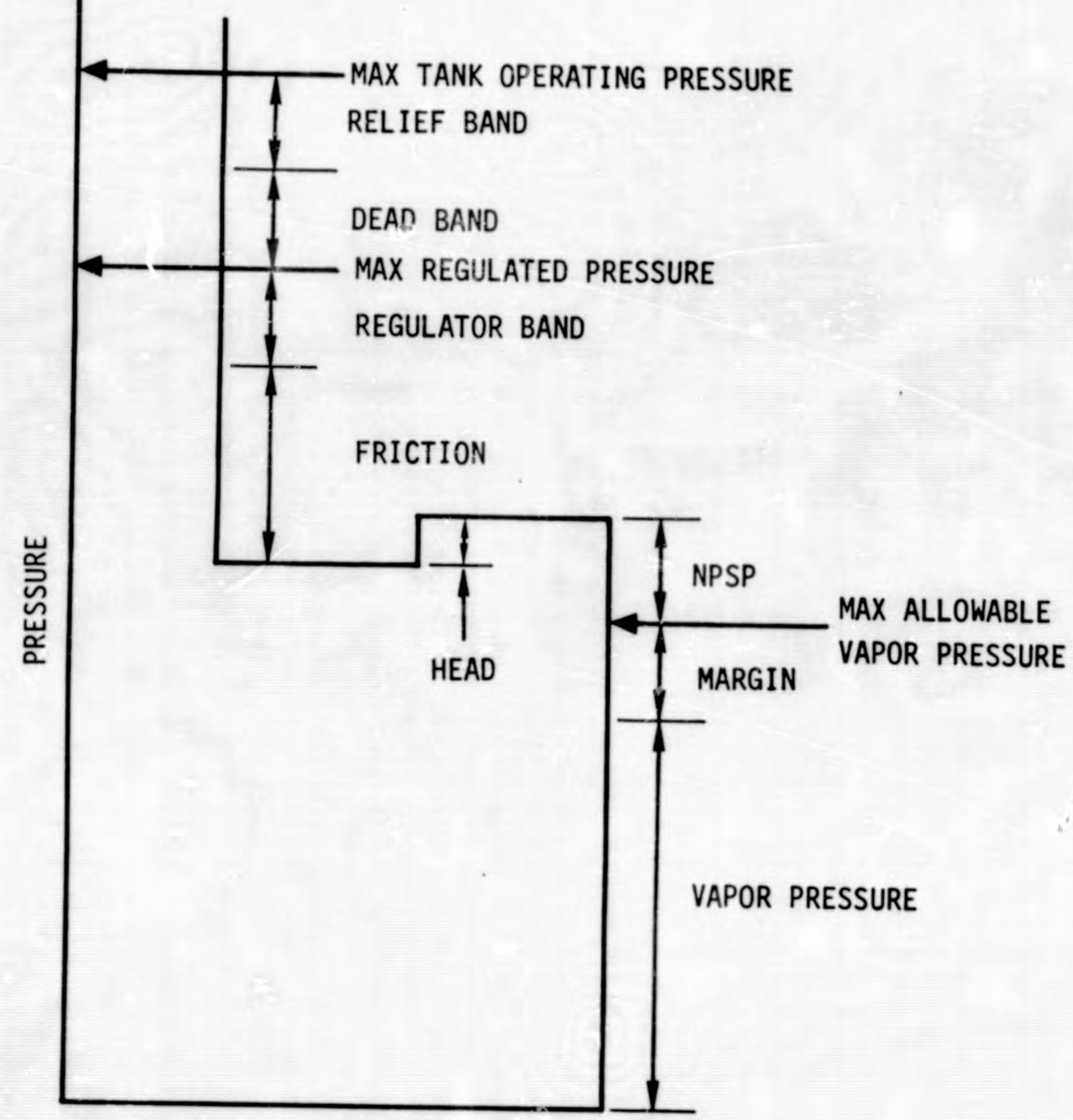


Figure 2.5-3 MPS Tank Pressure Levels

A common concept for mounting such a thermodynamic vent system would involve attachment of the vent tubes to the propellant tank wall. This concept has the advantage that incident energy to the tank can be intercepted at the tank wall, eliminating the need for any mixing system for liquid destratification. Destratification could be desirable under mission conditions whereby the tank pressure would otherwise rise to the relief valve level. However, success for this method of mounting would require that the tank be structurally capable of supporting the system. The thin tank walls designed for the Tug would preclude this design concept for mounting.

A thermodynamic vent system could be installed internal to the cryogenic propellant tank. Heat exchange from the propellant to the vent heat exchanger would be by convection or conduction from the liquid, or by condensation of ullage.

Design for the Tug synchronous orbital mission should be based on zero gravity where convection is non-existent. For the liquid region, reliance on the conduction mechanism would be necessary, in the absence of a mixer system.

There have been several detailed studies concerning the need for a fluid mixing system for control of ullage pressure. One of the most comprehensive studies was conducted by General Dynamics/Ft. Worth (Ref. 2.5-2). The basic reason for consideration of a mixing system was to limit pressure rise by destratification which provides a consequent transfer of heat into the liquid phase. The goal of the present study concerning Tug is diverse from those prior studies, in that control of bulk liquid temperature to insure meeting of NPSP requirements is desired. Transfer of energy from the ullage into the *liquid* is not desired.

Usage of a mixer so that a compact heat exchange system could be employed would introduce the reliability problems attendant to a pump or blower. For this reason, design of a thermodynamic vent system based on conduction alone would be preferable. However, this would create cold regions within the tank, and possibly allow some rise in tank pressure. But the basic criteria should be that a thermodynamic vent system be designed to remove the same quantity of heat as is introduced through the walls and by other leaks. Upon destratification at the next scheduled maneuver, the mixed liquid temperature would equalize to the desired level.

If there is no provision for cooling, one potential problem which should be addressed is that of reaching the relief valve pressure. The maximum relief valve pressure is set at 24 psia, which provides a margin of 7 to 9 psi above the initial vapor pressure. The longest predicted coast period with no destratification maneuvers is 74 hours. For the design heat loads, the LH_2 bulk temperature would rise approximately 1.75°R (Table 2.5-2), equivalent to 3.5 psi. This would appear to be safely below the 7 to 9 psi margin, but thermal stratification or an increase in heat loads above that expected could provide a necessity for a destratification rotation of the vehicle. With the presence of cooling, especially when portions of the cooling coils are exposed to ullage, the risk of a rise in pressure to the vent relief setting

Table 2.5-2. Propellant Temperature Change by Exterior Heat Sources in the Absence of Active Thermal Conditioning

	Time Lapse Hrs	Propellant*** Mass, Lbs.		Capacitance, BTU/°F		**Heat into Propellant, BTU		Temperature Rise, °R	
		LOX	LH_2	LOX	LH_2	LOX	LH_2	LOX	LH_2
1.	1.85	47328	7888	19870	18000				
*2.	13.0	47282	7880	19760	17950	850	1710	0.043	0.095
*3.	2.0	25523	3920	9790	8855	115	230	0.012	0.026
*4.	3.3	23394	3899	9723	8800	187	380	0.019	0.043
5.	0.5	13286	2214	5481	4920				
*6.	0.5	13236	2206	5460	4900	57	115	0.010	0.023
*7.	74.0	13074	2179	5390	4830	4200	8500	0.780	1.76
8.	1.0	12928	2155	5330	4785				
*9.	12.5	12899	2150	5320	4770	770	1565	0.145	0.33
*10.	2.0	6321	1053	2550	2254	115	230	0.045	0.10
*11.	3.3	6265	1044	2530	2231	187	380	0.050	0.17
					$\sum$	6481 BTU	13110 BTU	1.0 °R (1 psi)	2.55 °R (6.5 psi)

*Propulsive Maneuver

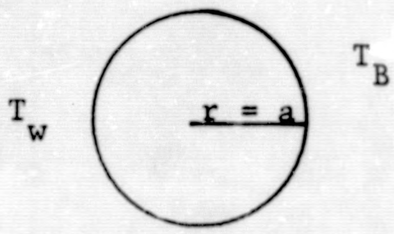
**Tank Heat Load Rates; LH_2 115 BTU/Hr.
LOX 57 BTU/Hr.

***Preliminary estimate for analysis only

could be eliminated. For the LOX system, a maximum rise in vapor pressure of 1 psi during the mission is expected, hence, a pressure rise to the vent relief setting should not occur.

Design of Thermodynamic Vent System.

Conduction in Liquid. This design contemplates no destratification mixing system. For the liquid system, there is an assumption of conduction into an infinite medium, with a driving force, $\Delta T = 7°F$.



$$T_B - T_W = 7°F$$

from Carslaw and Jaeger (Ref. 2.5-3), P336, Eq. 8,

The surface flux, f, is given by

$$f = \frac{K\Delta T}{a}\left[\frac{1}{\frac{\pi\alpha t}{a^2}} + \frac{1}{2} - \frac{1}{4}\left(\frac{\alpha t}{a^2 \pi}\right)^{1/2} + \frac{1}{8}\frac{\alpha t}{a^2}\right]$$

on P338, Figure 42, Ref. 2.5-3, a plot of the flux as a function of time, t, is given in the form $\text{Log}_{10}\,[af/K\Delta T]$ vs $\text{Log}_{10}\,[\alpha t/a^2]$.

For LH_2.

$$\alpha = \frac{K}{\rho C_p} = \frac{0.07}{(4.4)\ (2.3)} = 0.007\ \frac{ft^2}{hr.}$$

let a = 3/16 in = 0.0156 ft.

$$\frac{\alpha}{a^2} = 28.4\ \text{Sec.}^{-1}$$

$$\frac{a}{K\Delta T} = \frac{0.0156}{(.07)\ (7)} = 0.032\ \frac{ft^2\ hr.}{BTU}$$

Figure 2.5-4 shows a plot of f vs. t, and an integration of that curve, tabulated on Figure 2.5-4, provides the total heat transferred for a given time interval. The results for the Tug synchronous orbital mission are presented in Table 2.5-3.

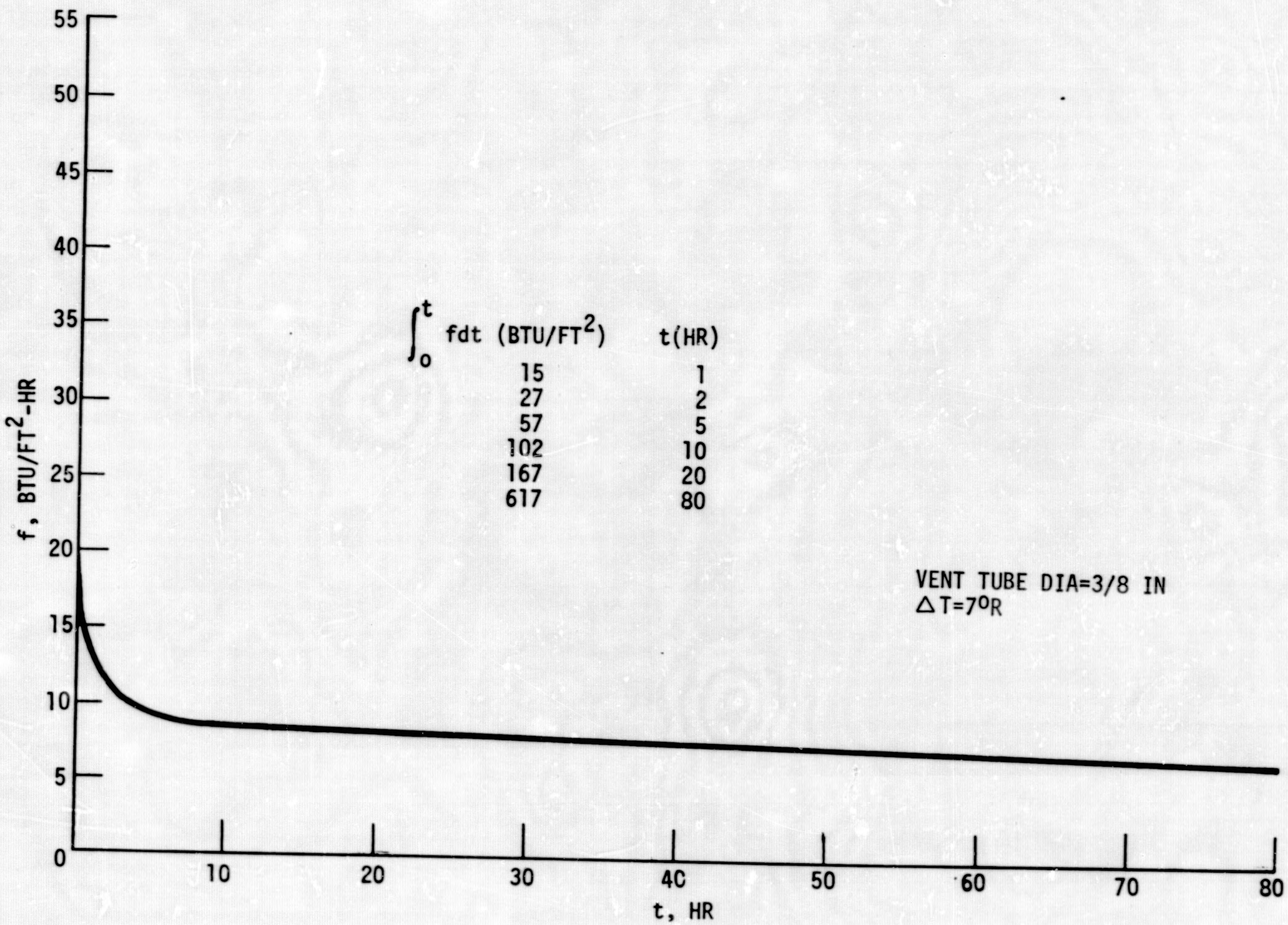


Figure 2.5-4 Conduction from LH_2 Vent Tube into LH_2

Table 2.5-3. Heat Extracted to LH_2 Thermodynamic Vent System; Conduction Mode

Time Lapse, Hrs.	% of Tank Vol. Filled	Heat Extracted, BTU/Ft^2 Region		Heat Extracted Multiplied by % of Fill, BTU/Ft^2		Sum of Liquid and Vapor Heat Extracted, BTU/Ft^2
		Liquid	Vapor	Liquid	Vapor	
1.85	0.93					
13.0	0.93	140	165	130	11	141
2.0	0.465	27	62	13	33	46
3.3	0.465	40	78	19	42	61
0.5	0.263					
0.5	0.263	15	43	4	31	35
74.0	0.26	575	370	149	273	422
1.0	0.255					
12.5	0.255	140	160	36	120	156
2.0	0.125	27	62	3	54	57
3.3	0.125	40	78	5	68	73
		Σ 1004				991

Similarly for the LOX system, where

$$\alpha = \frac{K}{\rho C_p} = \frac{0.083}{(73)\ (.43)} = 0.0026\ \text{Ft}^2/\text{Hr}.$$

$$a = 3/16\ \text{inch} = 0.0156\ \text{Ft}.$$

and

$$\frac{\alpha}{a^2} = 10.4\ \text{Sec.}^{-1}$$

$$\frac{a}{K\,\Delta T} = \frac{0.0156}{(0.083)\ (18)} = 0.01\ \frac{\text{Ft}^2\ \text{Hr}.}{\text{BTU}}$$

Figure 2.5-5 shows a plot of f vs. t for LOX, with an integration of that curve tabulated on Figure 2.5-5. The results for the Tug synchronous orbital mission are presented in Table 2.5-4.

Condensation of Ullage. The other possibility is that the vent coils will be in the ullage. With the assumption of condensation in the ullage, for LH_2.

For the ullage temperature at the saturation temperature with conduction only (zero gravity), a condensation parameter, β, is defined by:

$$\frac{e^{-\beta^2}}{\text{erf}\ \beta} = \frac{\beta\ h_{fg}\ \sqrt{\pi}}{C_{PL}\ \Delta T} \qquad \text{(Reference 4)}$$

where

$$\beta = \frac{X(t)}{2\sqrt{\alpha_L t}}$$

for

$$\Delta T = 7°\text{F}, \qquad \beta \approx 0.33$$

This provides a measure of the condensation rate, hence

$$X(t) = 2\beta\sqrt{\alpha_L t}$$

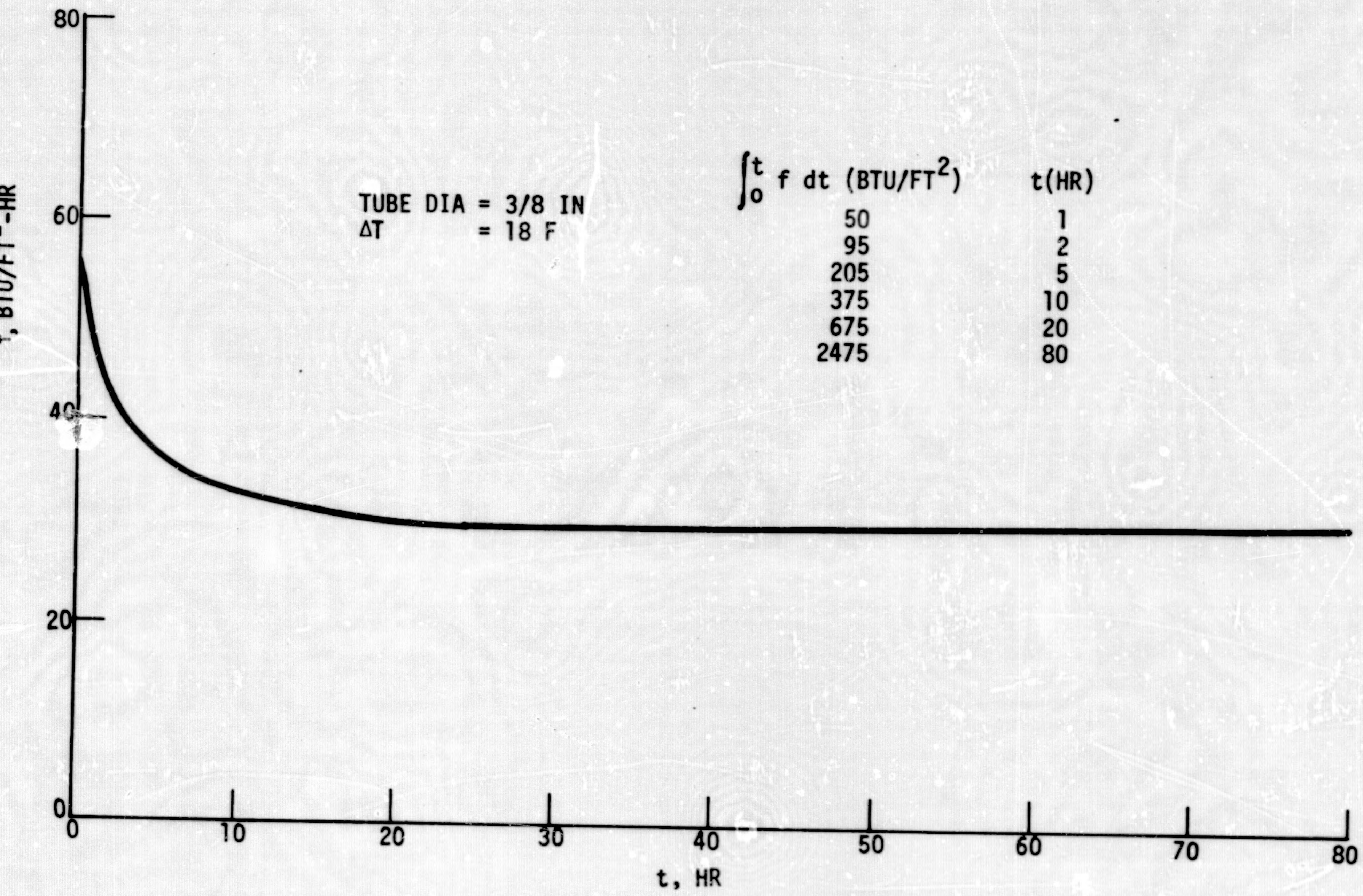


Figure 2.5-5 Conduction from LOX Vent Tube into LOX

Table 2.5-4. Heat Extracted to LOX Thermodynamic Vent System; Conduction Mode

Time Lapse, Hrs.	% of Tank Vol. Filled	Heat Extracted, BTU/Ft^2 Region		Heat Extracted Multiplied by % of Fill, BTU/Ft^2		Sum of Liquid and Vapor Heat Extracted, BTU/Ft^2
		Liquid	Vapor	Liquid	Vapor	
1.85	0.91					
13.0	0.91	525	730	480	65	545
2.0	0.45	95	270	43	150	193
3.3	0.45	150	345	67	190	257
0.5	0.255					
0.5	0.255	50	190	13	140	153
74.0	0.25	2275	1630	570	1220	1790
1.0	0.25					
12.5	0.25	500	700	125	520	645
2.0	0.12	95	270	11	235	246
3.3	0.12	150	345	18	300	318
		Σ 3840				4147

Space Division
North American Rockwell

for

$$\alpha_L = 0.007 \text{ Ft}^2/\text{Hr.}$$

$$X(t) = 0.055 \sqrt{t} \text{ Ft.}$$

The results would not be significantly different for ullage temperatures of 30 or 40°R greater than saturation (Reference 2.5-4).

The mass condensed = $\rho V = \rho A\ X(t)$.

$$\frac{m}{A} = 0.055\ \rho \sqrt{t}\ \frac{\text{Lb}}{\text{ft}^2}$$

$$\frac{m}{A} = 0.24 \sqrt{t}\ \frac{\text{Lb}}{\text{ft}^2}$$

$$\frac{Q}{A} = \frac{m}{A} h_{f_g} = 43 \sqrt{t}\ \frac{\text{BTU}}{\text{ft}^2}$$

This assumes that the liquid does not drain off.

Similarly for condensation of LOX,

$$\frac{e^{-\beta^2}}{\text{erf }\beta} = \frac{\beta\, h_{f_g} \sqrt{\pi}}{C_{P_L} \Delta T} = \frac{\beta\ (90)\ (\sqrt{3.14})}{(.43)\ (18)} = 20.6\ \beta$$

$$\beta = 0.29$$

$$X(t) = 2\beta \sqrt{\alpha_L t} \text{ ft.}$$

$$X(t) = (2)\ (.29)\ (\sqrt{.0026t}) = 0.0296 \sqrt{t} \text{ ft.}$$

$$\frac{Q}{A} = \rho\ X(t)\ h_{f_g}$$

$$\frac{Q}{A} = (71.3)\ (.0296\sqrt{t})\ (90)$$

$$\frac{Q}{A} = 190 \sqrt{t}\ \frac{\text{BTU}}{\text{Ft}^2}$$

The required tubing surface area for heat exchanger is determined as follows:

The required surface area is:

$$S = \pi DL = \frac{Q}{\Sigma\, Q/A}$$

For LH_2

$$S = \frac{13110}{991} = 13.3 \text{ Ft}^2$$

$$L = \frac{13.3}{\pi D} = \frac{13.3}{\pi\ (.375/12)} = 135 \text{ Ft.}$$

With assumption of 0.010 inch S.S. walls

$$m = A_c L\ \rho_m$$

where

ρ_m = density of metal

A_c = cross section area

$$m = \frac{\pi}{4}\ \frac{[.375^2 - .355^2]}{144}\ (135)\ (480)$$

$$m = 5.2 \text{ Lbs.}$$

Since the total heat transferred, $\Sigma\, Q/A$, is only slightly different for the two cases of total immersion in the liquid or for proportional immersion in the liquid and vapor, these values *should* be satisfactory regardless of the liquid orientation.

For the LOX system

$$S = \frac{Q}{\Sigma\, Q/A} = \frac{6481}{4197} = 1.6 \text{ Ft}^2$$

$$L = \frac{S}{\pi D} = \frac{1.6}{\pi\ (.375/12)} = 23 \text{ lineal feet for LOX tank.}$$

Overboard Venting. Depressurization or vent down of a cryogenic tank in orbit is a realistic method of reducing bulk liquid temperature. This was experimentally confirmed during tests of an orbiting Saturn S-IVB LH_2 tank, (References 2.5-5 and 2.5-6), under settling accelerations of 2×10^{-5} to 3×10^{-4} g. Venting of a propellant tank in a low g environment introduces the potential problem of overboard venting of liquid, either through boilover caused by a rise in the liquid level, or by entrainment within the vapor stream. Consequently, venting must be initiated only after the propellant has been settled, and must be terminated before either of these two phenomena can cause venting of significant quantities of liquid.

A pressure decrease can cause the bulk liquid to become superheated. A small superheat can be sustained before boiling will occur at various nucleation sites, or cavities along the wall and other hardware. This superheat has been estimated to be of the order of 0.1°R for LH_2 (Reference 2.5-6). For the insulated Tug propellant tanks, the incident heat leaks should be sufficiently low that the heat available from the superheated liquid should predominate in the ensuing boiling. Results from the S-IVB test showed this to be an effective method of reducing the liquid temperature, (Ref. 2.5-5). The bulk liquid lagged the saturation temperature by about 0.1°R.

Overboard venting is contemplated after settling, but prior to pre-pressurization for propellant feedout. During this venting, which occurs after a quiescent period coast, the liquid is likely to be superheated relative to the design bulk liquid temperature in the absence of a thermodynamic vent system. The following discussion concerning boiling applies whenever the liquid is superheated.

Once boiling has been initiated along the walls, the bulk liquid level will begin to rise due to entrainment of the vapor within the liquid under a low g environment. Analytical methods have been developed by Convair (Reference 2.5-6) for dimensionless liquid level rise as a function of pressure reduction for various assumed degrees of vapor entrainment.

The worst assumptions would be 100% vapor entrainment and all of the superheat entering into vaporization. The actual degree of vapor entrainment will depend in large measure on the rate of depressurization. Analysis of the S-IVB tests showed these assumptions to be overly conservative (Ref. 2.5-6), but they should suffice for a design basis. The analysis would indicate that for a LH_2 pressure drop of 7 psi, the initial fill level would need to be less than 1/2 full.

The displacement of liquid by the boiling and entrainment of vapor would require that the venting be terminated with sufficient time for re-settling prior to initiation of feedout to the engines.

The S-IVB depressurization tests were accompanied by Television pictures within the hydrogen tank (Reference 2.5-5). Upon opening of the vent, a fog was observed to begin rising above the liquid surface. Approximately 1-1/4 minutes were required for the fog to rise to the top of the

tank, which contained LH_2 sufficient to fill approximately 40% of the tank. During this period, the depressurization was approximately 2.4 psi. A subsequent depressurization from 17 to 13 psia within 90 seconds showed the fog approached the tank dome shortly after the 90 seconds.

Liquid globules much larger than that which could be entrained within the stream of fog were observed moving toward the dome, but within the envelope of the advancing fog stream. These were presumed to have been formed by either coalescence or by interfacial disturbance during the boiling process. In a positive acceleration field, these globules should begin to fall, if their upward momentum can be arrested. For this purpose deflective baffles to route any globules away from the vent and toward the bottom of the tank will be installed.

Available time for propellant settling, venting, and re-settling of propellant should be compared to that which is necessary for employment of the depressurization method of cooling the stored propellant. Reduction of bulk liquid temperature at rates significantly above these evaluated in the S-IVB venting tests should be considered to be beyond the present state-of-the-art. If vent-down at an average of 0.02°R/sec. were selected as a design limit, which is just beyond the upper limit of experimental data (Ref. 2.5-5), this would provide a vapor pressure decay of approximately 3 psi in one minute. If the liquid were greatly stratified, the actual pressure drop could be much greater than 3 psi, to be equivalent to this maximum vapor pressure reduction of 3 psi.

The worst case for the predicated baseline mission would be after the 74-hour coast period where the saturation pressure is anticipated to have increased by nearly 5 psi (See Table 2.5-2) for which 100 seconds should be allowed for depressurization. Additional time should be allowed for settling and resettling of the liquid.

Based upon the propellant temperature rises presented in Table 2.5-2, the LH_2 tank will require venting only once during the mission. The LOX tank will not require venting. Therefore, both propellant tanks will be vented after the 74-hour coast period. Venting of the LOX tank will be performed to enhance mission flexibility and to be compatible with the LH_2 tank. This method of propellant vapor pressure control presents the simplest solution combined with the lightest weight system.

The pressurant mass requirements and propellant vapor pressure histories throughout the mission are shown on Tables 2.5-5 for LOX and 2.5-6 for LH_2. The boiloff quantities were established from internal enthalpy requirements based on saturation conditions existing after orbital coast periods. This assumption represents a worst case condition when establishing pressurant requirements. The equations used in this analysis are shown below. The equations require an iterative solution in their present form.

Table 2.5-5. LOX Tank Pressurant Mass Requirements

Time (hr)	Propellant Vapor Pressure (psia)	Boiloff Mass (lb)	Vented Mass (lb)	Prepress Mass (lb)	Expulsion Mass (lb)	Ullage Mass (lb)
0	15.0	0	0	0	0	7.3
0.1	15.4	0.2	0	0	0	7.5
14.85	15.5	0	0	0	0	7.5
14.85	15.5	0	0	1.3	63.2	72.0
20.15	15.4	33.8	0	0	0	105.8
20.15	15.4	0	0	18.4	26.9	151.1
108.65	15.9	11.0	0	0	0	162.1
108.65	15.2	0	20.1	0	0	142.0
108.65	15.9	0	0	17.5	17.5	177.0
113.95	15.6	-0.7	0	0	0	176.3
113.95	15.6	0	0	14.4	14.9	205.6
135.95	17.7	6.2	0	0	0	211.8
135.95	17.7	0	0	21.6	0.4	233.8
136.71	17.7	0	0	0	0	233.8
136.71	17.7	0	0	2.0	0.4	236.2
	Totals (lb)	50.5	20.1	75.2	123.3	236.2

Table 2.5-6. LH_2 Tank Pressurant Mass Requirements

Time (hr)	Propellant Vapor Pressure (psia)	Boiloff Mass (lb)	Vented Mass (lb)	Prepress Mass (lb)	Expulsion Mass (lb)	Ullage Mass (lb)
0	16.0	0	0	0	0	6.0
0.1	17.4	0.5	0	0	0	6.5
14.85	17.6	0.2	0	0	0	6.7
14.85	17.6	0	0	0.2	13.4	20.3
20.15	16.1	69.7	0	0	0	90.0
20.15	16.1	0	0	2.8	5.7	98.5
108.65	18.2	78.8	0	0	0	177.3
108.65	16.2	0	63.2	0	0	114.1
108.65	16.2	0	0	3.9	3.7	121.7
113.95	15.5	23.0	0	0	0	144.7
113.95	15.5	0	0	5.5	3.2	153.4
135.95	15.7	26.6	0	0	0	180.0
135.95	15.7	0	0	7.1	0.1	187.2
136.71	15.7	0	0	0	0	187.2
136.71	15.7	0	0	0	0.1	187.3
	Totals (lb)	198.8	63.2	19.5	26.2	187.3

$$\frac{144V}{J}(P_2 - P_1) = m_{v_2}(A+BP_2) - m_{v_1} h_{v_1} + (m_{v_1} + m_{L_1} - m_{v_2})(C+DP_2) - m_{L_1} h_{L_1} - Q$$

$$\frac{m_{v_2}}{E + FP_2} + \frac{(m_{v_1} + M_{L_1} - m_{v_2})}{\rho L_2} = V$$

where:

V = Tank volume (ft^3)

J = 778.2 ft-lb/Btu

P = ullage pressure (psia)

m_v = ullage mass (lb)

h_v = ullage enthalpy (Btu/lb)

m_L = liquid mass (lb)

h_L = liquid enthalpy (Btu/lb)

Q = heat added (Btu)

ρ_L = liquid density (lb/ft^3)

A = empirical constant (Btu/lb) (31.678, LOX; 75.21, LH_2)

B = empirical constant (Btu/lb-psi) (0.169, LOX; 0.448, LH_2)

C = empirical constant (Btu/lb) (-63.519, LOX; -123.3, LH_2)

D = empirical constant (Btu/lb-psi) (0.4192, LOX; 0.912, LH_2)

E = empirical constant (lb/ft^3) (.02382, LOX; .0091, LH_2)

F = empirical constant (lb/ft^3 -psi) (.017358, LOX; .00508, LH_2)

Subscripts,

1 = initial conditions

2 = final conditions

The prepress and expulsion quantities were calculated from the following equations, which will be derived from the first law of thermodynamics.

$$\Delta m_p = \frac{F_{cp}}{C_p T}\left[\frac{144\,\Delta PV}{J\,(\gamma-1)}\right]_u$$

$$\Delta m_E = \frac{F_{c_E}}{C_p T}\left[\frac{144\,P\,\Delta V}{J}\left(\frac{\gamma}{\gamma-1}\right)\right]_u$$

$$F_c = 1 + C\left(\frac{T}{T_{sat}}\right) - 1$$

where:

Δm = pressurant mass (lb)

F_c = Collapse factor

C_p = constant pressure specific heat of pressurant (Btu/lb °R)

T = pressurant temperature (°R)
Prepress: 400, LOX; 200, LH_2
Expulsion: 600, LOX; 260, LH_2

ΔP = required pressure change for prepress (psi)

V = volume pressurized (ft^3)

J = 778.2 ft-lb/Btu

γ = ratio of specific heats

P = regulated pressure level (psia)

ΔV = volume change during expulsion (ft^3)

C = empirical constant (.3, LOX; .05, LH_2)

T_{SAT} = liquid temperature (°R)

Subscripts,

P - prepress

E = expulsion

The ullage pressure and temperature histories for the LOX and LH_2 tanks are presented on Figures 2.5-6 and 2.5-7, respectively. The purge cycles required for inerting the tanks are also shown on these figures.

2.5.3 Subsystem Selection and Options Considered

The options considered are shown schematically in Figure 2.5-8. Option 1 is the selected option and appears on Figure 2.5-1. Prepressurization gases are available from the APS accumulators and pressurization gases are tapped off of the main engine. Option 2 utilizes stored, heated helium for prepressurization. Pressurization of the LOX tank is done with stage helium heated in the main engine heat exchanger. LH_2 tank pressurization gas is tapped off of the main engine.

Option 3 entails prepressurization with stored, heated helium and pressurization with gases tapped off of the main engine. Option 4 utilizes cold, stored helium prepressurization with cold helium pressurization for the LOX tank. LH_2 tank pressurization gas is tapped off of the main engine. Option 5 has stored, cold helium prepressurization with pressurization gases tapped off of the main engine. In all the options, the helium was assumed to be stored in the LH_2 tank.

The selected option represents both the lightest weight system and the system with the least complex operation. The relative weights of all the options are presented on the Options Summary Matrix in Table 2.5-7. Also presented in this table are some of the basic advantages and disadvantages associated with each concept. The selected option has a disadvantage in that the engine NPSP requirements cannot be met instantaneously, that is propellant slosh can collapse the ullage pressure below the regulator level and a specific period of time will have to be used to prepressurize the tanks. Since the main engine burns will be planned, this prepressurization period can be incorporated into the overall mission timeline.

The APS propellant conditioning system must be able to start instantaneously. The turbopumps are supplied propellants from the APS screened tanks. Pressurization for this tank comes from the main tank, less the screen pressure drop. Therefore, the ullage pressure decay problem also exists for the screened tank. This problem has been eliminated by incorporating inducers on both propellant pumps. The inducers operate with a zero NPSP requirement. Therefore, the APS screened tanks have no pressurization requirements. Additionally, the thermodynamic vent system used for maintaining liquid in the feedlines will also have coils around the outside of the screened tank to subcool the APS propellants relative to the main tank propellants.

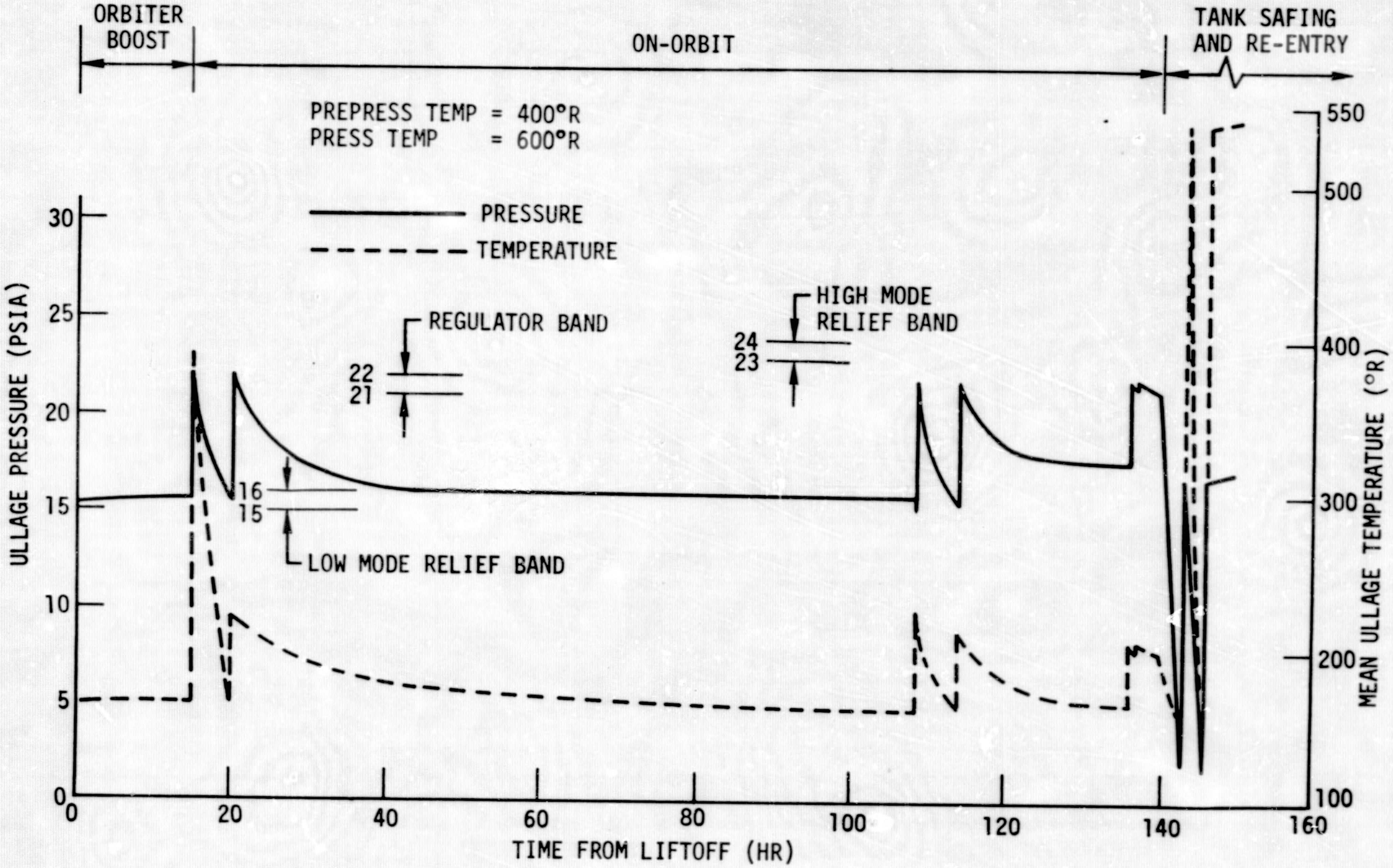


Figure 2.5-6 Main LOX Tank Temperature/Pressure History

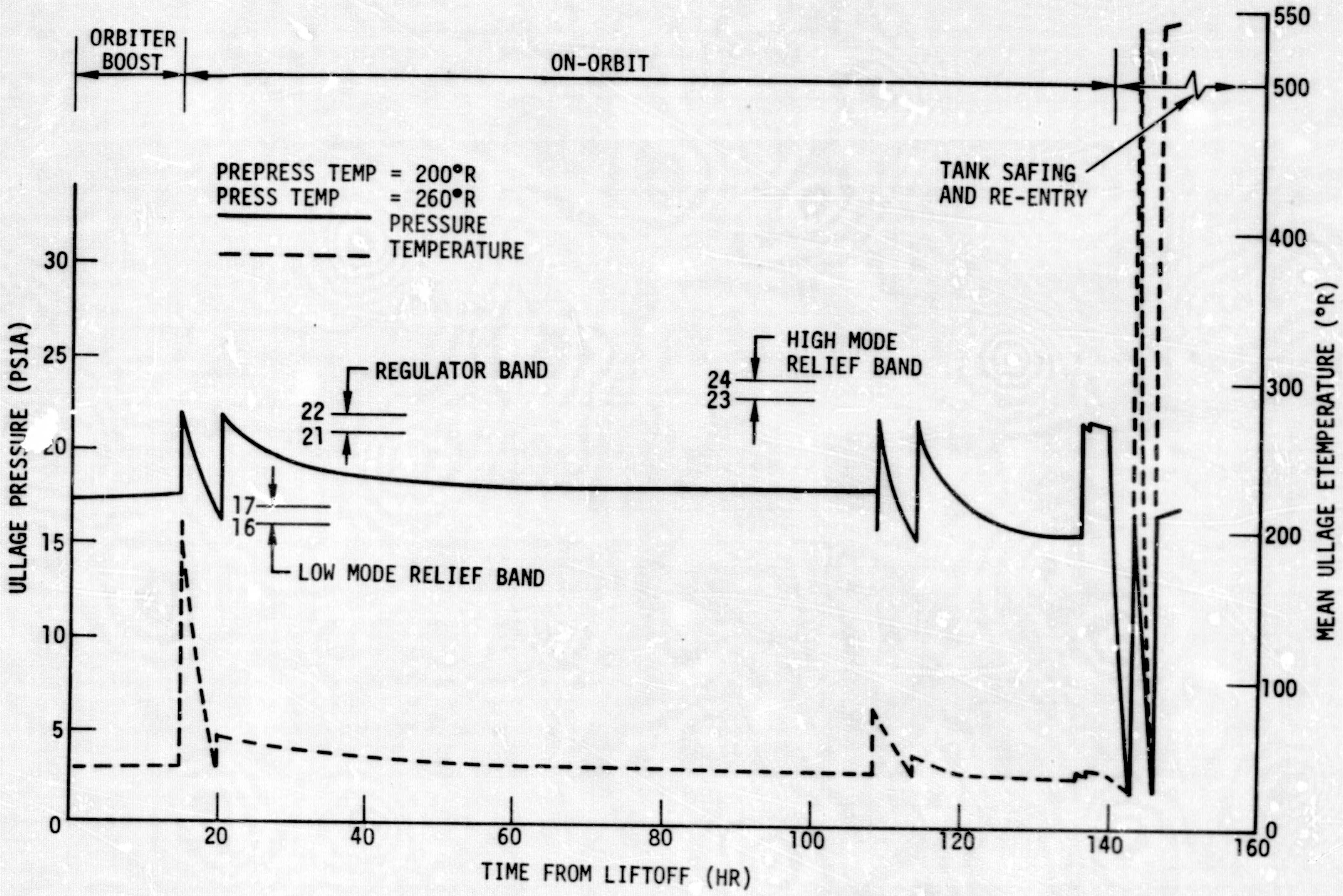


Figure 2.5-7 Main LH_2 Tank Temperature/Pressure History

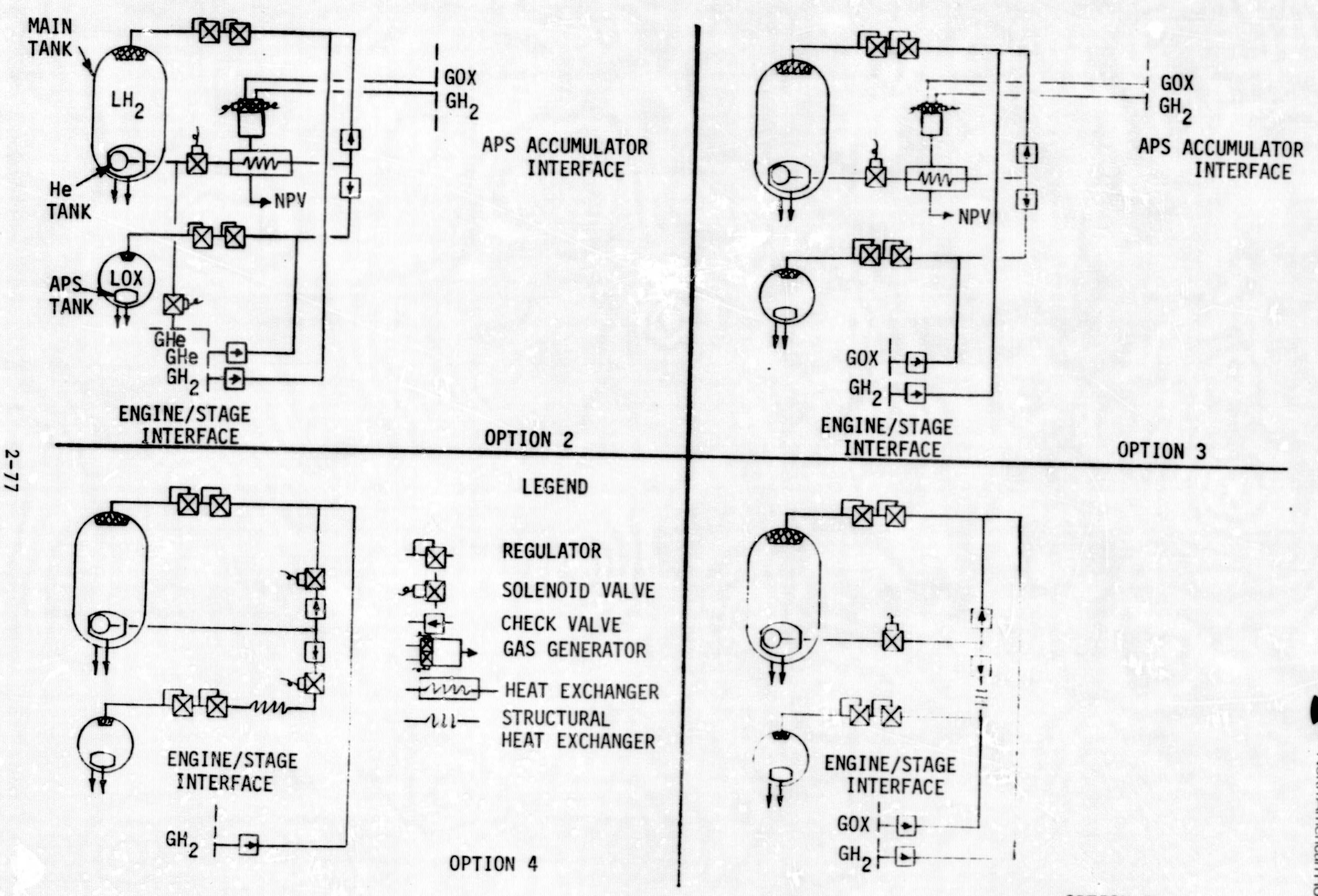


Figure 2.5-8 Optional Systems Schematics

Table 2.5-7. Option Summary Matrix

Option	Relative Weight (lb)	Operational Characteristics	
		Advantages	Disadvantages
1	0	APS gas availability, ME heat exchanger availability.	NPSP not available instantaneously.
2	+115	NPSP requirements met by partial pressure of stored gas, ME heat exchanger availability.	Requires stage inert gas storage system, separate gas generator and heat exchanger. Requires additional engine interface.
3	+165	NPSP requirements met by partial pressure of stored gas, ME heat exchanger availability.	Requires stage inert gas storage system, separate gas generator and heat exchanger.
4	+685	NPSP requirements met by partial pressure of stored gas, ME heat exchanger availability.	Requires stage inert gas storage system. Requires additional engine interface.
5	+730	NPSP requirements met by partial pressure of stored gas, ME heat exchanger availability.	Requires stage inert gas storage system.

2.6 PROPELLANT ACQUISITION SYSTEM

2.6.1 Summary

This section presents the results of a design study of propellant acquisition and feedout for the Tug main propulsion (MPS) and auxiliary propulsion systems (APS).

For the MPS, baffles to hasten propellant settling, prevent sloshing at low fill levels, and prevent vortex formation, are defined.

Six candidate concepts for APS propellant acquisition and feedout were evaluated, using as selection criteria, weight, reliability, operational complexity and mission flexibility. The selected concept utilizes an auxiliary container with an unvalved screened port into the main tank. The container has a 17 ft^3 capacity for LH_2 and a 3.3 ft^3 capacity for LOX, and is refillable in orbit. The MPS and APS propellant supplies are integrated giving unlimited mission flexibility. The APS container can be ground filled, launched upside down, and does not require helium pressurization. Propellant has direct access to the container via the unvalved screened port. Container propellant is maintained subcooled (compared to main tank pressure) by a Joule-Thompson expansion valve and chill coil heat exchanger. Refilling can be accomplished during MPS or APS delta V maneuvers or APS settling maneuvers, using an overboard vent for vapor purging. A tubular collector system, covered with screen material, is provided within the containers for APS propellant acquisition from any portion of the container.

2.6.2 Design Description

For each of the propellants, hydrogen and oxygen, two propellant feed systems are required. The main propulsion (MPS) feed system, which accounts for the bulk of propellant usage, supplies propellant for main engine burns. The auxiliary propulsion feed system (APS) supplies propellant for the APS thrusters, thermodynamic vent/liquid chill system, fuel cells, and prepressurization. The function, operation, interaction and design of the two feed systems is described in the following:

MPS Propellant Acquisition and Feedout

The design does not utilize bulk propellant control. Instead, a propellant settling maneuver is performed prior to each main engine burn. Propellant settling for spacecraft has been investigated and criteria established by one-g test programs conducted at NR/SD. Settling was found to be a complex phenomenon involving countercurrent liquid and ullage gas movement, liquid rebound and slosh damping, and bubble formation and rise. Based upon these criteria, MPS settling requirements were established. Briefly stated, settling is accomplished in two steps, with a slight overlap. First, a 22.5-second APS burn (two engines providing 140 lb thrust) is accomplished using propellant from the APS accumulators. This is followed by 120 seconds of chill idle (35 lb thrust). In total, this thrust level-time profile yields.

4.5 "free fall" periods for the worst case propellant loading. If the propellant is assumed as adversely located at a distance, h, from the bottom of the tank, the time required for one free fall period is

$$t = \left(\frac{2h}{g}\right)^{\frac{1}{2}}$$

where g is the acceleration. It is recognized that from a propellant settling point of view it would have been more efficient to reverse the settling sequence; i.e., low thrust followed by high thrust. However, this is not consistent with the MPS requirement that chill idle be preceded by settling.

To hasten the settling process, decrease rebound, and assure bubble free propellant in the outlet region at main engine feed, four settling baffles have been provided at the bottom of the LH_2 and LOX tanks. These baffles and other design features of the tank bottom are shown in Figure 2.6-1. The baffle layout was based upon the possibility of symmetric or asymmetric settling. For the latter case, propellant may flow downward along any sector of the tank wall. The baffles are placed to dissipate the momentum of the settling flow and guide liquid towards the main engine outlet for settling flow from any direction. The APS container, itself, functions as part of the baffle system; conversely, the container must be capable of withstanding settling loads. As an example of the function of the baffles, the case of flow coming from θ = 90 degrees is considered as follows. Without baffles such a condition represents an adverse, if not "worst case" settling direction, for the initial settling propellant would flow around the APS container and separate; this would leave a vapor pocket behind the container and over the outlet for an extended time. In this design the 60° arc baffle would guide and turn the settling flow, deflecting some of the flow against the outlet baffles and into the outlet region. Settling flow from other directions presents other problems. Flow coming from θ = 30 or 150 degrees is to some extent deflected away from the outlet region by the arc baffles. Therefore, the baffles are specified as containing holes or slots so as to be 40 percent open. This permits some of the flow to pass through the baffles towards the outlet region and hastens energy dissipation.

The four settling baffles also function as slosh baffles for low liquid levels. During the last few main engine burns these baffles inhibit slosh and concommitant uncovering of the main engine outlet. Additionally, the baffles function to facilitate refilling of the APS container.

Investigations were conducted to assess the magnitude of the pull through and vortex formation problem during main engine feed. The pull through or surface dip problem appears to be minor; the high acceleration field during the last engine burn reduces the pull-through height and residual propellant. Test data show that the pull-through height above the tank outlet is 2.5 inches for LOX and 3.5 inches for LH_2. The "trapped" residual is not really lost as some of it can be fed to the APS container following the last main engine burn. Vortex formation due to rotation of propellant during main engine feed is potentially more serious than pull through, as vortex formation with

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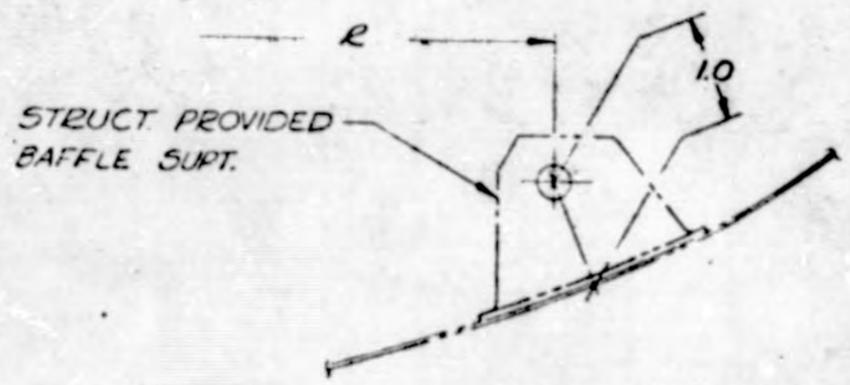
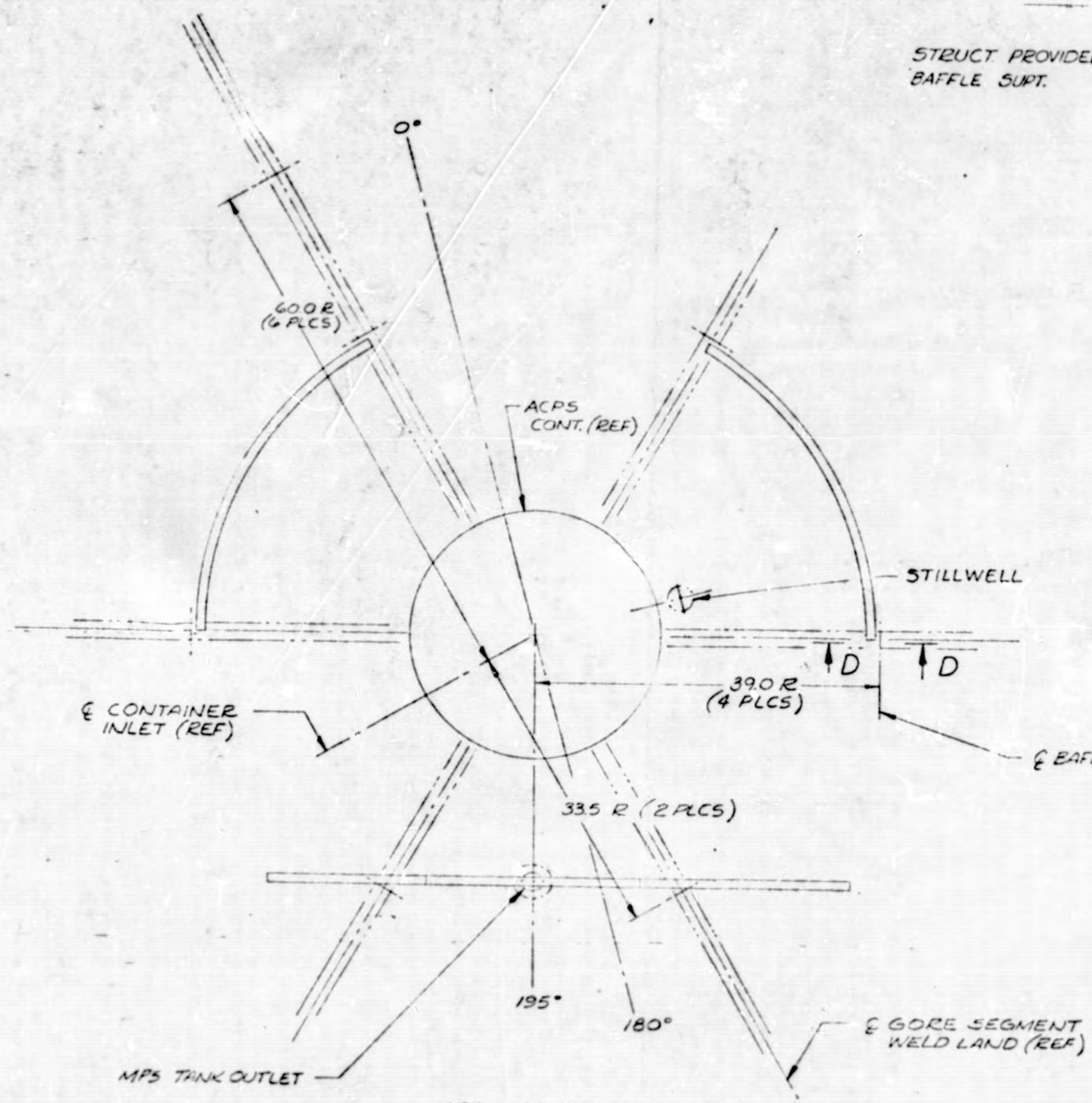
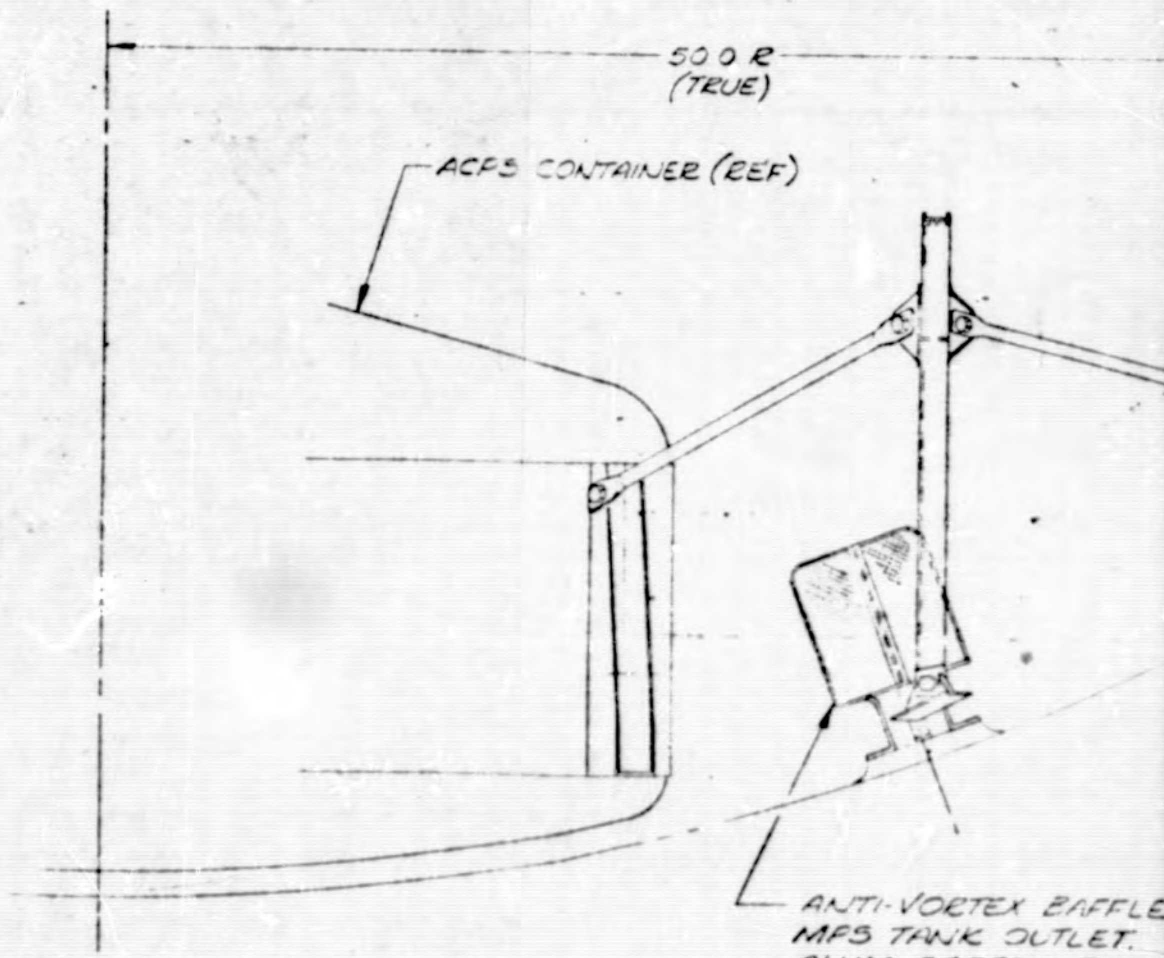


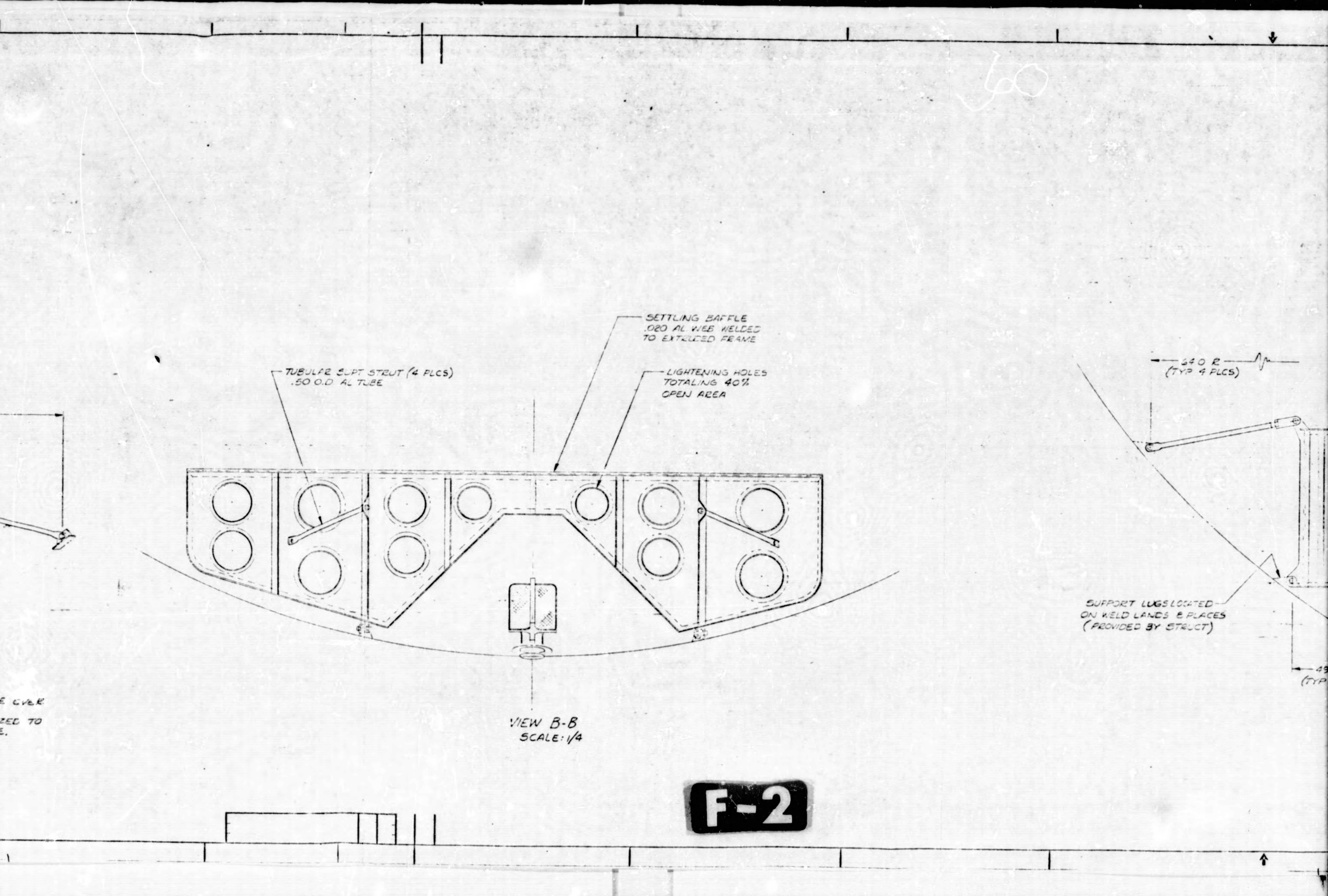
VIEW D-D
SCALE: 1/1
(TYP 12 PLACES)

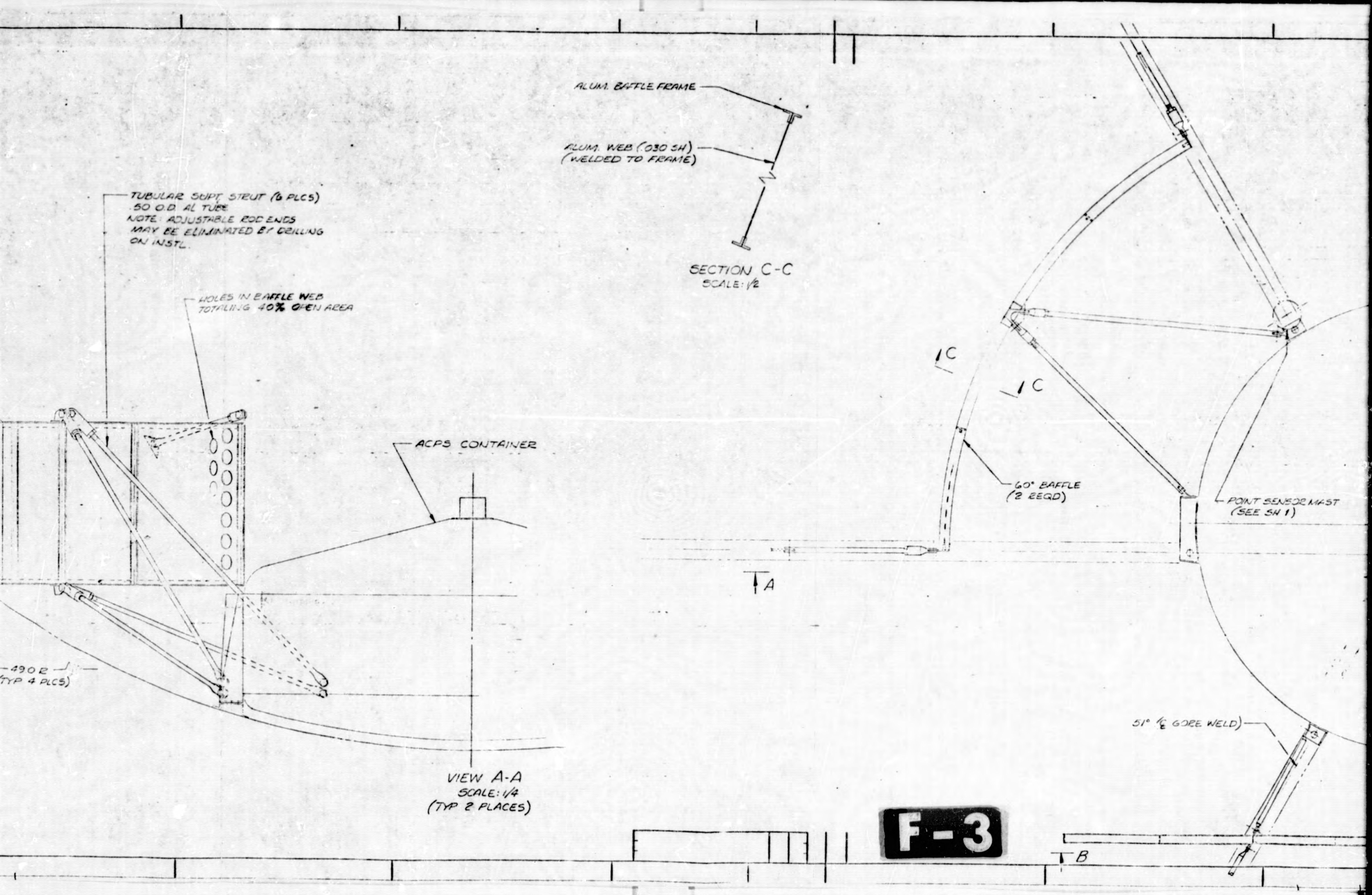


LOOKING AFT IN LOX TANK
SCALE: 1/8
LOX TANK BAFFLES SAME AS
LH_2 TANK BAFFLES EXCEPT
AS SHOWN



F-1


SETTLING BAFFLE
.020 AL WEB WELDED
TO EXTRUDED FRAME
TUBULAR SUPT STRUT (4 PLCS)
.50 O.D AL TUBE
LIGHTENING HOLES
TOTALING 40%
OPEN AREA
64.0 R
(TYP 4 PLCS)
SUPPORT LUGS LOCATED
ON WELD LANDS 8 PLACES
(PROVIDED BY STRUCT)
VIEW B-B
SCALE: 1/4
F-2


ALUM. BAFFLE FRAME
ALUM. WEB (.030 SH)
(WELDED TO FRAME)
SECTION C-C
SCALE: 1/2
TUBULAR SUPT. STRUT (6 PLCS)
.50 O.D. AL TUBE
NOTE: ADJUSTABLE ROD ENDS
MAY BE ELIMINATED BY DRILLING
ON INSTL.
HOLES IN BAFFLE WEB
TOTALING 40% OPEN AREA
ACPS CONTAINER
C
C
60° BAFFLE
(2 REQD)
POINT SENSOR MAST
(SEE SH 1)
A
-490 R
(TYP 4 PLCS)
VIEW A-A
SCALE: 1/4
(TYP 2 PLACES)
51° ℄ GORE WELD)
B

F-3

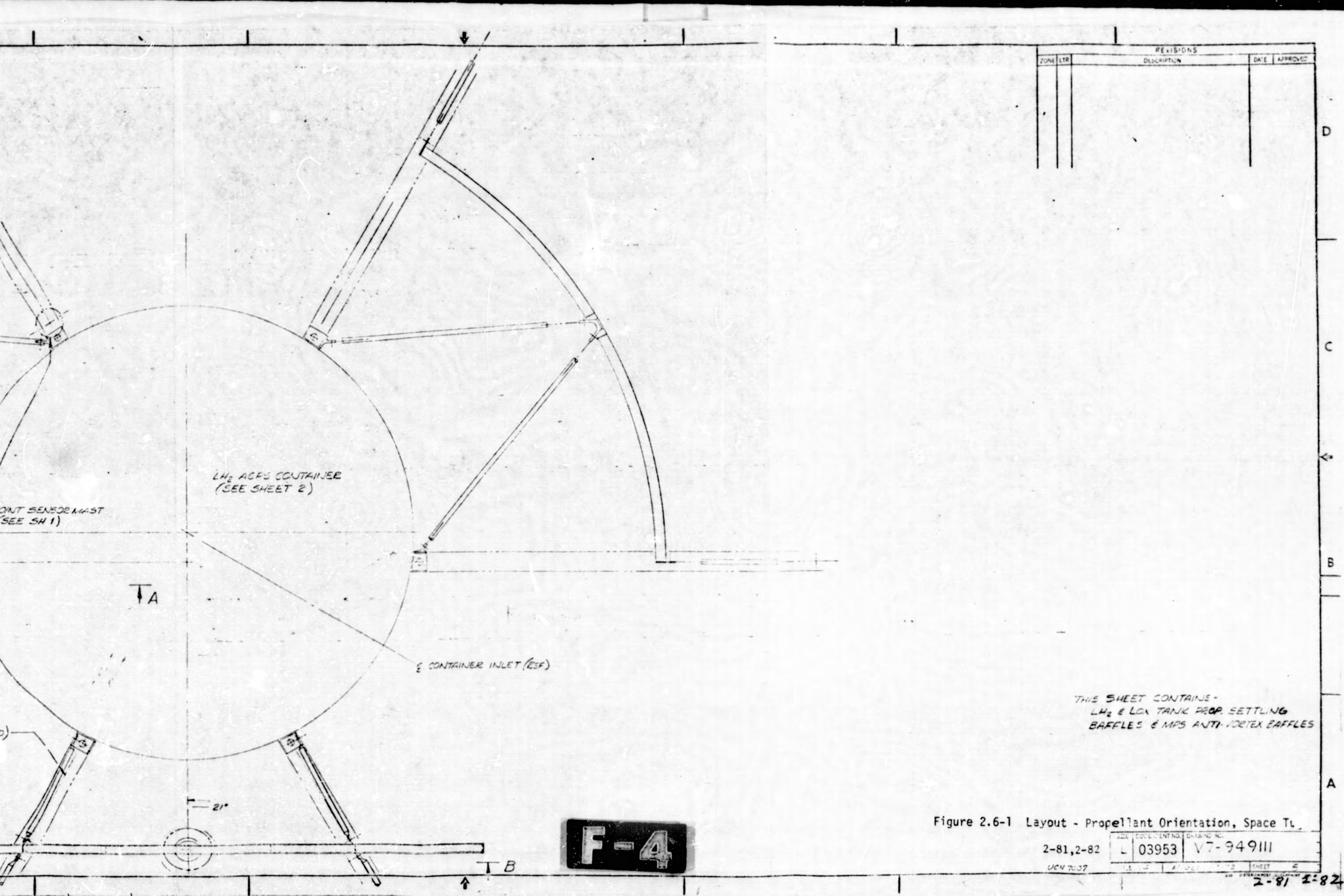


Figure 2.6-1 Layout - Propellant Orientation, Space Tu

concommitant gas injection may occur during any main engine burn. Recent work on vortex formation, has provided information on the flow processes and on free surface deformation. The off-center location of the main tank outlet alleviates the problem somewhat, but a vortex baffle, as shown in Figure 2.6-1 is required.

APS Propellant Acquisition and Feedout

The selected system utilizes an auxilliary container with an unvalved screened part into the main tank (See Figure 2.6-2) that can be ground filled and launched upside down, can be refilled in orbit, and does not require helium pressurization. The system is light-weight, has low residuals and is believed to be highly reliable.

The system requires special operating procedures but these are considered reasonable. The operation and design concepts are believed to be readily achievable within the 1976 time period.

An important design provision is the screened refill port. Pressure isolation and lack of fluid communication, both of which are undesirable, are avoided. The refill port is covered with self wicking 200 x 1400 mesh dutch twill screen. The dimensions of the refill ports are sufficiently small so that propellant retention during launch is achieved with a safety factor of four. The shape of the port (two to three times as wide as it is high) is based upon anticipated launch acceleration which is two to three times as great in the vertical direction as in the lateral direction. Thus, hydrodynamic stability (i.e., the ability to prevent liquid loss) is equally good in all directions. The refill port area is sufficiently large so that flow losses during feedout and refill are only a fraction of a psi. Using NR/SD data for pressure drop flowing across 200 x 1400 mesh dutch twill screen, the pressure loss across the screens is less than 0.2 lb/in^2 for both LH_2 and LOX. This is based upon a flow rate into the containers of 1 lb/sec LH_2 and 3 lb/sec LOX. These are twice the maximum usage rates (0.5 lb/sec LH_2 and 1.5 lb/sec LOX) and, therefore, the minimum refill rates are 0.5 lb/sec LH_2 and 1.5 lb/sec LOX.

Although the mission APS propellant consumption is four times the container volume, it is believed that the refill requirements will not be nearly this great. This is due to the communication between the APS container and the main tank, and the subcooling of the APS container. Subcooling of the APS container lowers the vapor pressure of the liquid within. This leads to cooling and condensation of vapor within the ullage. As the pressure within the container drops below tank pressure, flow ensues from the tank into the container. If the entering fluid is liquid the container will refill rather quickly. If vapor enters, it will condense - resulting in slow but certain refilling of the container during coast periods.

This mode of refilling is a bonus and the design is not dependent upon it. Nominally, refill is accomplished during main engine firing. Shortly

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	DIMENSIONS (INCHES)				
	A	B	C	D	E
LOX	28	14	*4x5.5	0.5	1.0
LH_2	46	24	4x9	1.0	2.0

*4 IN RETURN ON BOTTOM

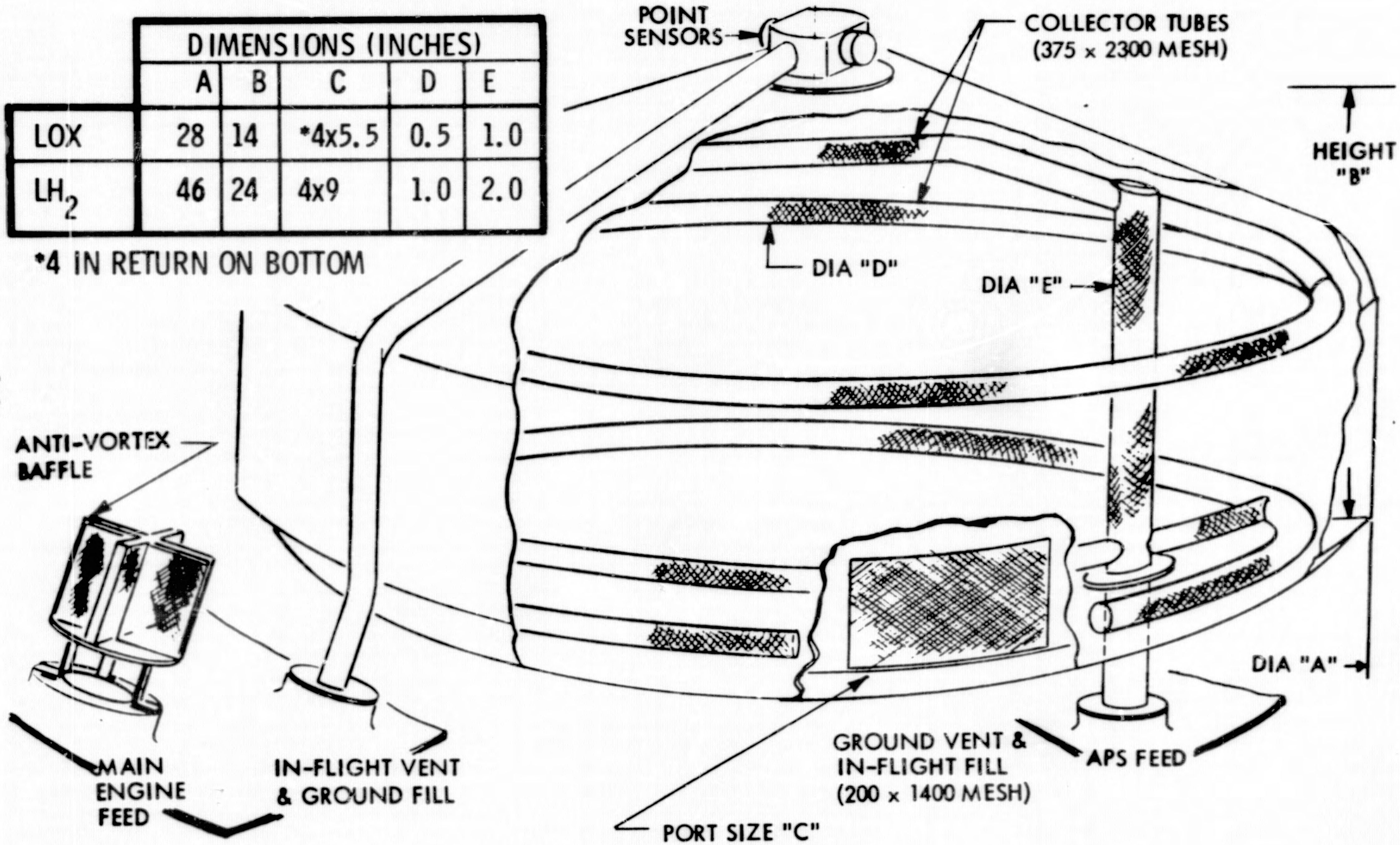

Figure 2.6-2 Auxiliary Propellant Tank

after main engine firing initiation (propellant is fully settled by then), the overboard vent valve is opened and refill continues until liquid is sensed at the vent outlet. As shown above, the pressure drop is sufficiently low (the pressure within the container is sufficiently high) so that local flashing is precluded.

As a backup to this method of refilling, refill can also be accomplished using APS engine +X firing to settle propellant. Once propellant is settled, refill is accomplished while maintaining +X APS thrust. This mode of refilling corresponds to APS ΔV maneuvers. Therefore the APS container can be refilled during APS ΔV burns lasting 32 seconds or longer.

The top of the container is tilted at 15 degrees to facilitate bubble rise to the top of the container. Therefore, the APS container is fully settled by the time the main tank is settled and there is assurance that vapor is properly located within the container when vapor venting is initiated.

It should be noted that refill will generally be required after periods of coast, for there is no guarantee that APS propellant withdrawn from the container will be replaced by liquid, i.e., there is no provision for bulk propellant location control during coast.

Within the container a screened tubular system is provided for feedout propellant acquisition. This system consists of a vertical hub collector, radial spoke collectors at top and bottom of the container, and toroidal collectors near the top and bottom of the container. This system, looking somewhat like a double wheel, is shown schematically in Figure 2.6-2 and in detail in the installation drawings of Figures 2.6-3,4. This collector system design assures that propellant will be acquired for all possible liquid/bubble orientations during coast. The collectors are covered with 375 x 2300 mesh dutch twill screen material. The collector screens, by preferentially passing liquid rather than vapor, assures liquid feed for all APS usage even if the container is nearly full of vapor.

The collector system is composed of one-inch diameter tubes for the LH_2 container and 0.5-inch diameter tubes for the LOX container for all tubes except the hub. The hub collector is two-inches in diameter for the LH_2 container and one-inch in diameter for the LOX container. For the LH2 and LOX containers, the total volume of the collector system is only 1.2 percent of the container volume. Residual liquid within the containers is conservatively estimated at 5 percent of the container volume. Toroidal and radial collectors are located at 0.5 inches from the container wall to avoid heating of the collector; this 0.5-inch offset is for extra reliability, as the container wall is chilled by the subcooling coil. The vertical collector was terminated sufficiently far from the top of the container to avoid interference with bubble purging.

The 375 x 2300 screen which lines the collector has a small micron rating (large bubble pressure and enhanced hydrodynamic stability). This small micron rating is required should the container be nearly empty at main engine burn initiation. In this case, the collector screen must retain propellant

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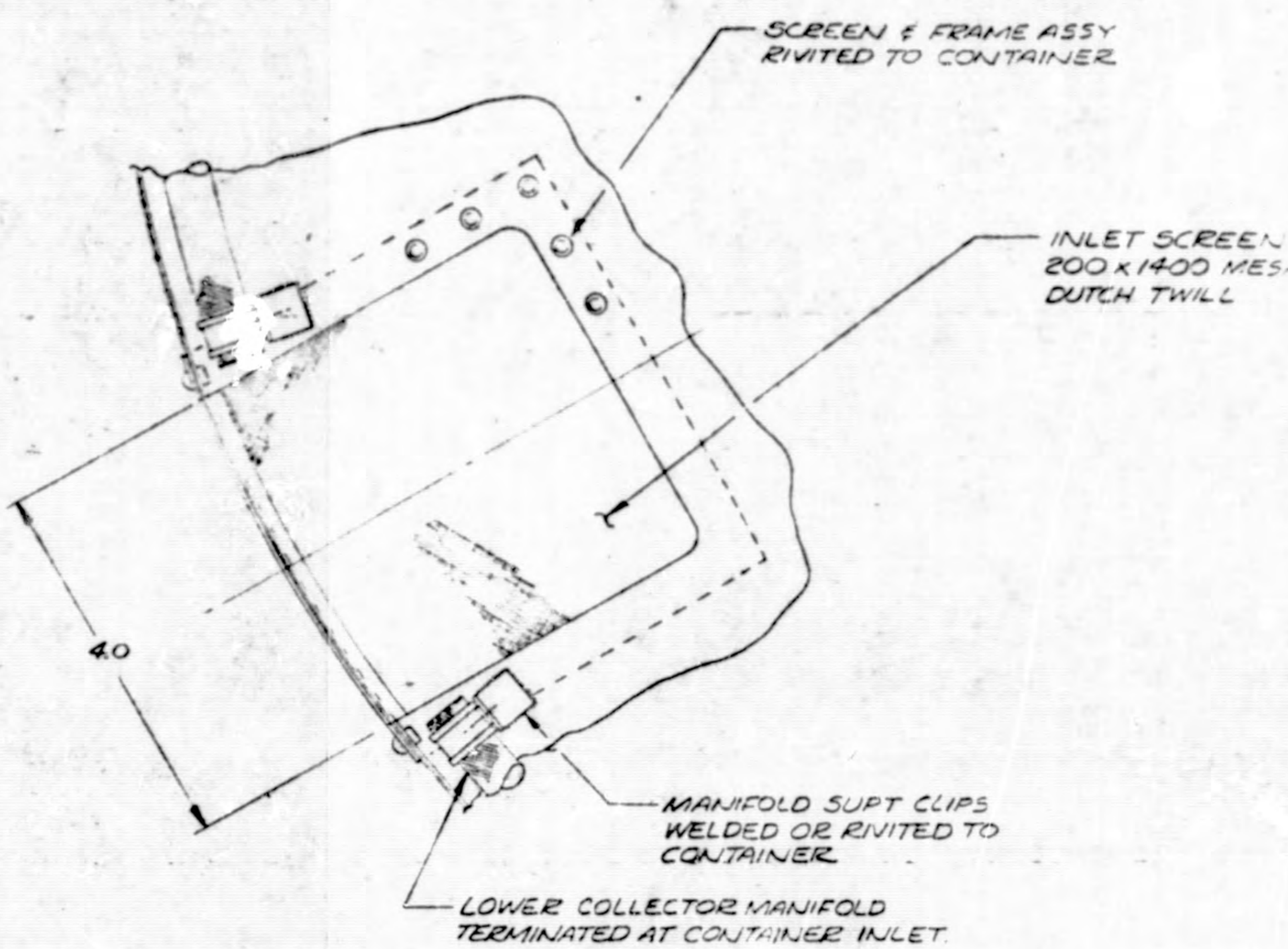
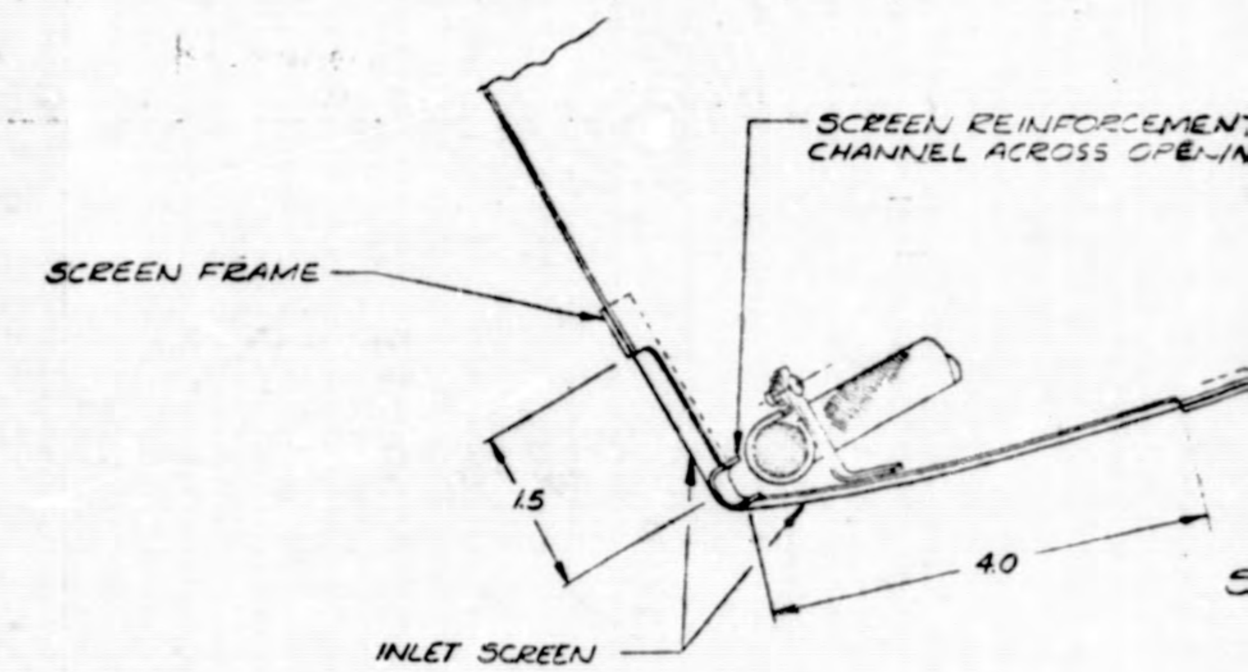
D


SCREEN & FRAME ASSY
RIVITED TO CONTAINER
INLET SCREEN
200 x 1400 MESH
DUTCH TWILL
4.0
MANIFOLD SUPT CLIPS
WELDED OR RIVITED TO
CONTAINER
LOWER COLLECTOR MANIFOLD
TERMINATED AT CONTAINER INLET

C

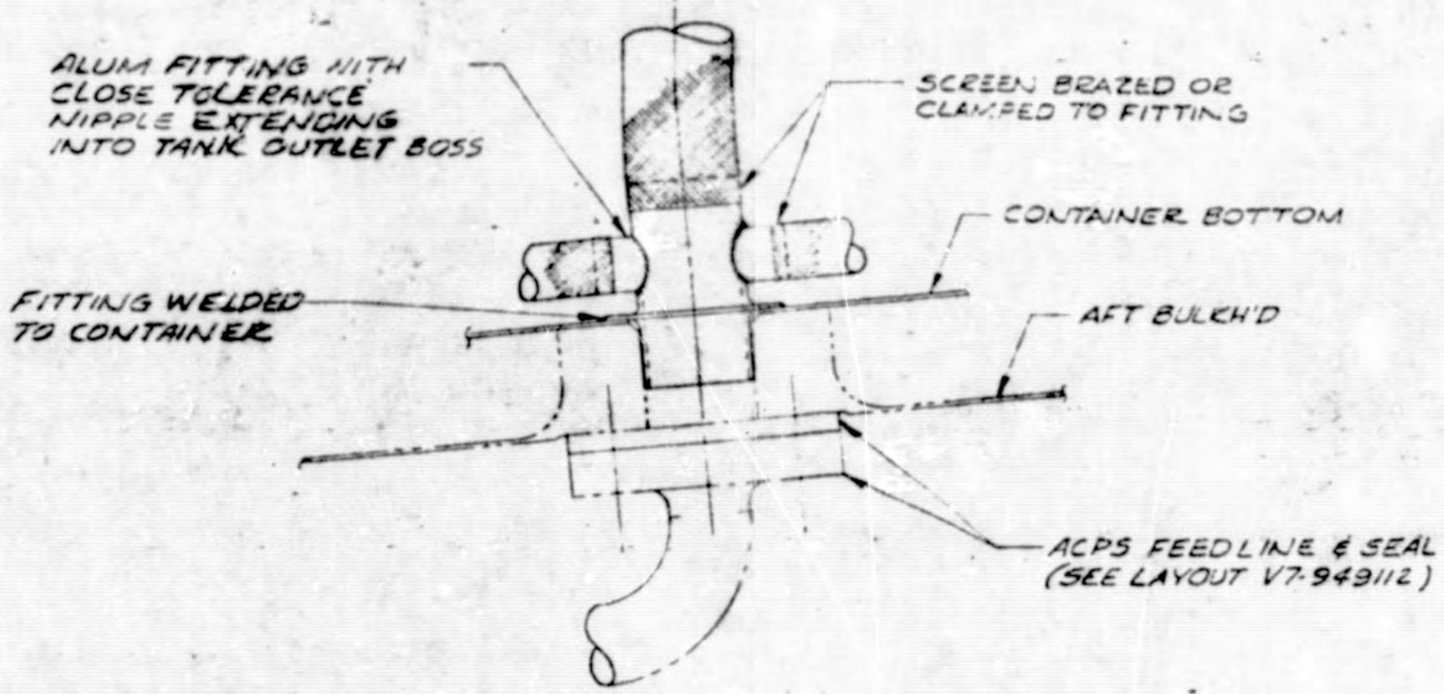
B

A

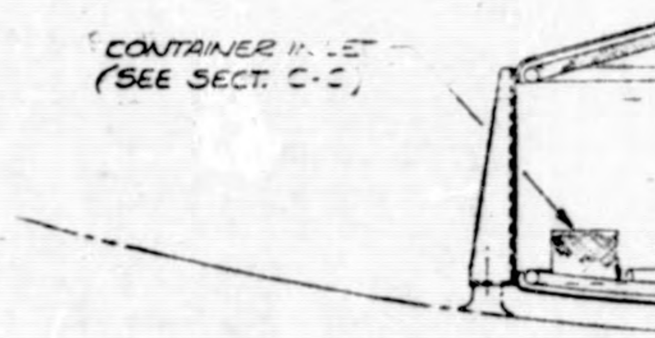

SCREEN REINFORCEMENT
CHANNEL ACROSS
SCREEN FRAME
1.5
4.0
INLET SCREEN

F-1

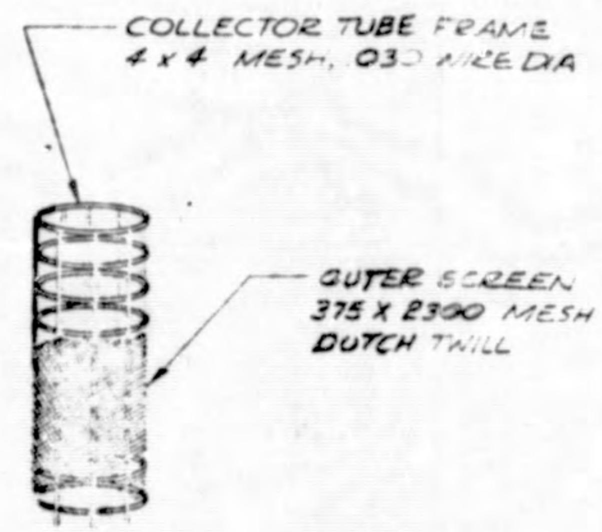
SECTION C-C
SCALE: 1/1



DETAIL D
SCALE: 1/1



F-2



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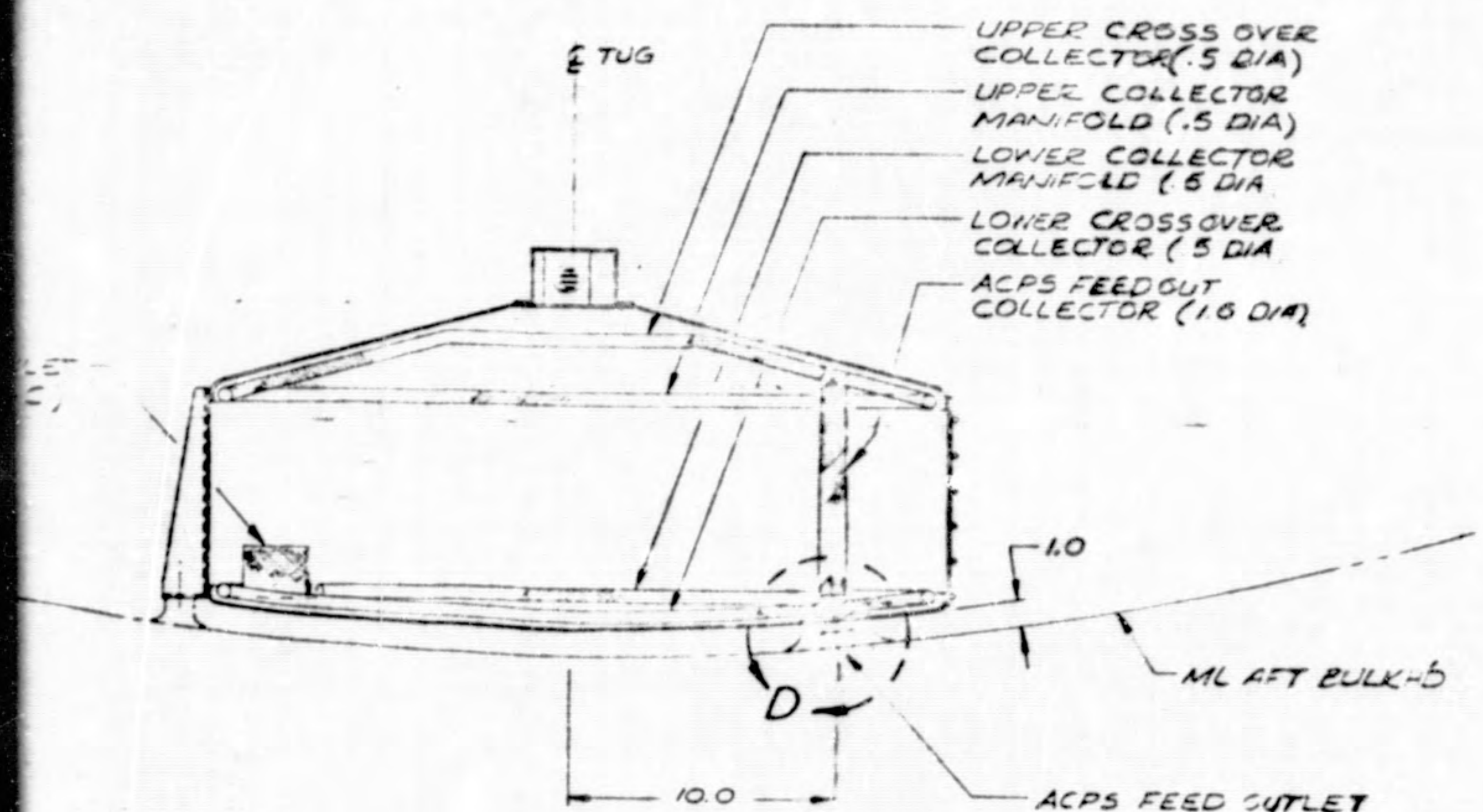

SECTION B-B
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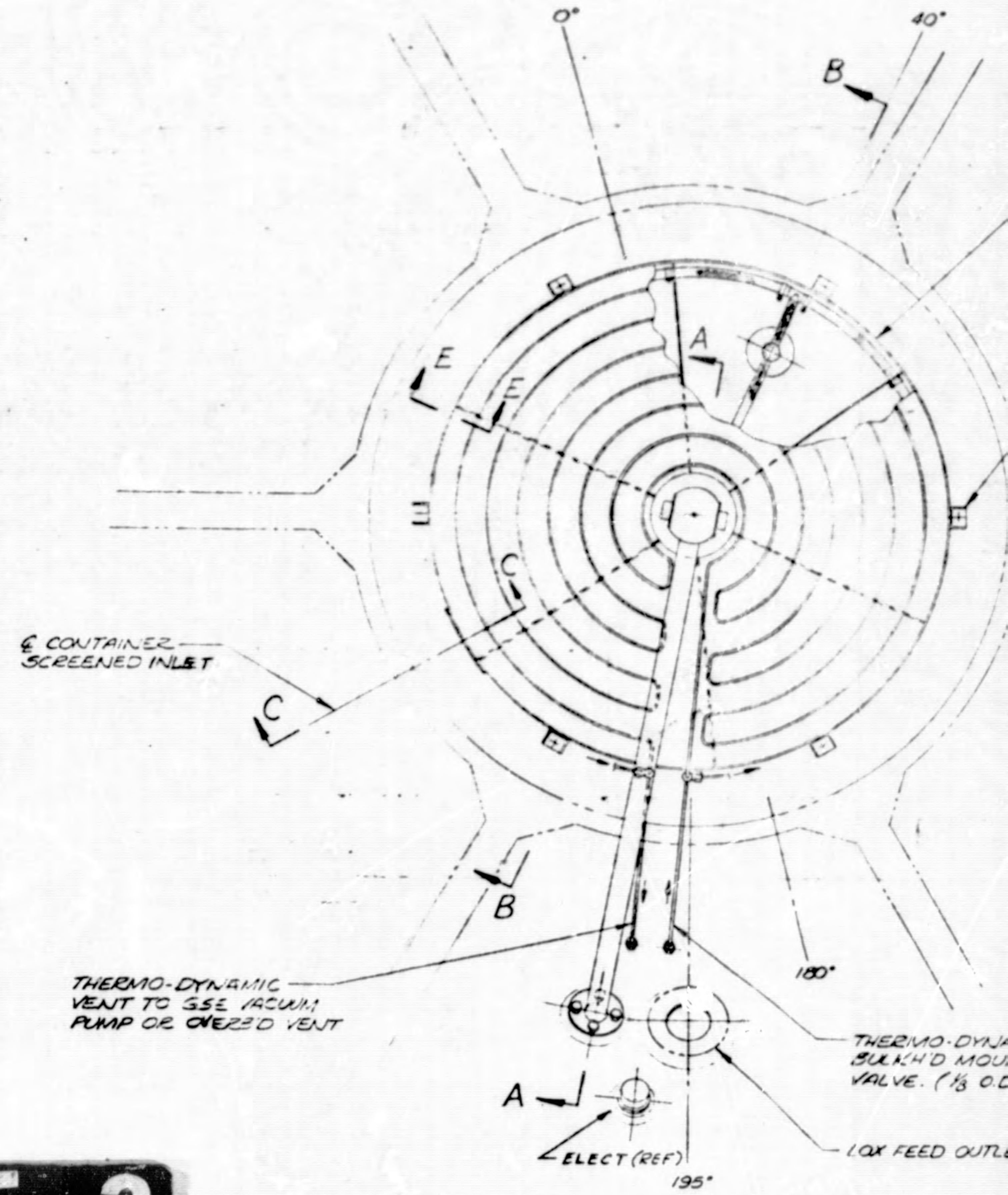


VIEW LOOKING AFT (TUG)
SCALE: 1/4

F-3

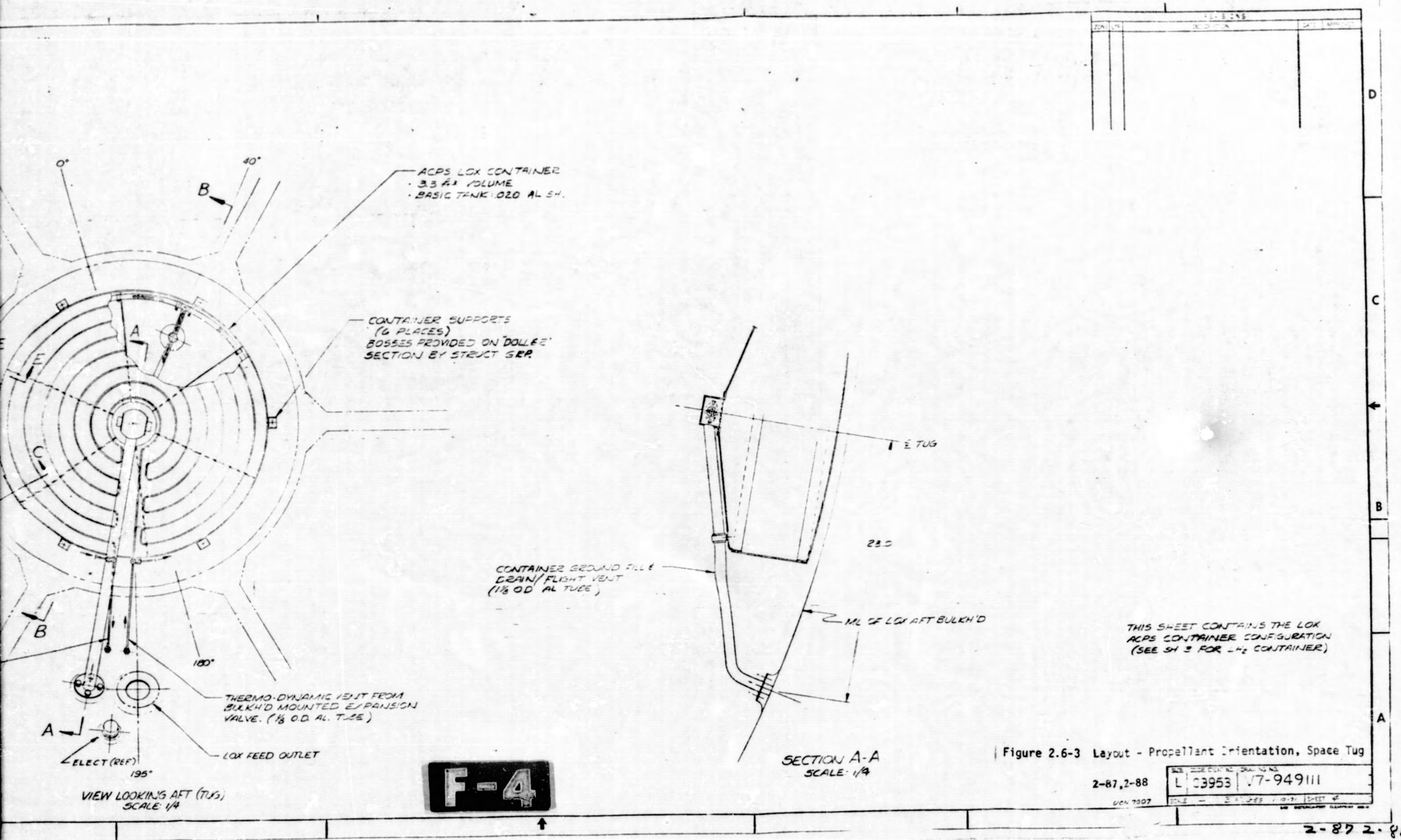
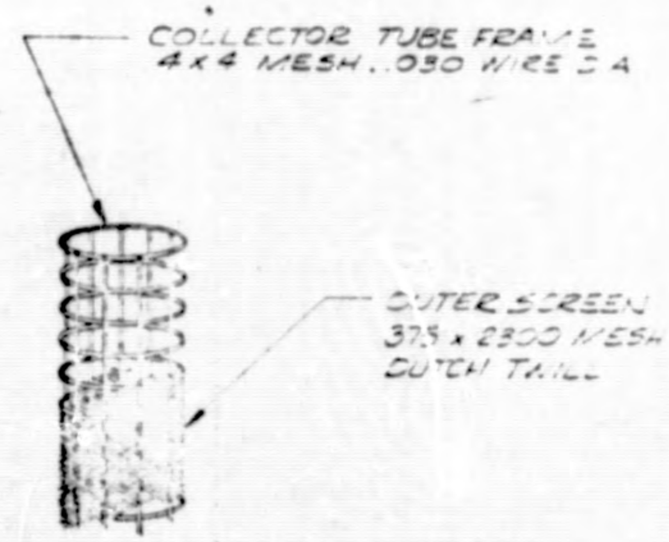


Figure 2.6-3 Layout - Propellant Orientation, Space Tug

2-87 2-88

D

C



TYP COLLECTOR CONFIG.
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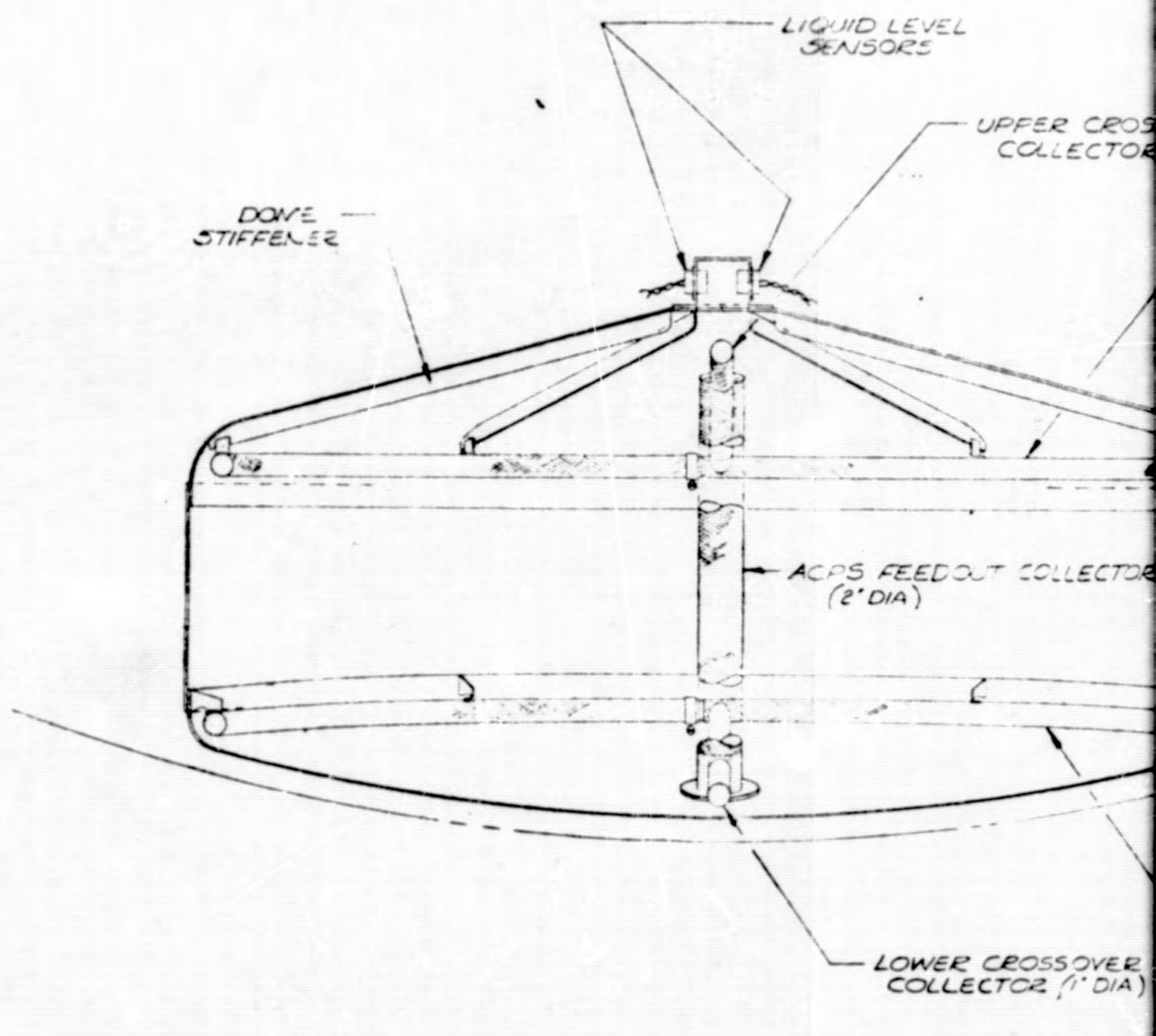
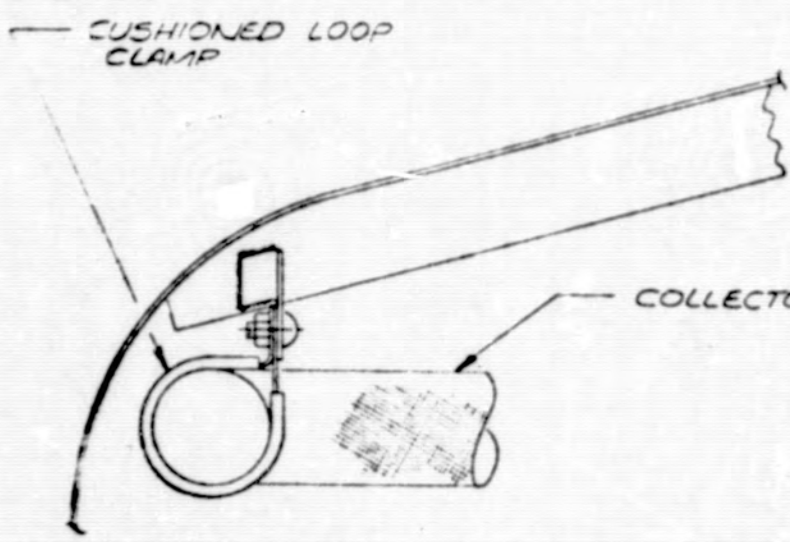


SECTION E-E
SCALE 1/4

B



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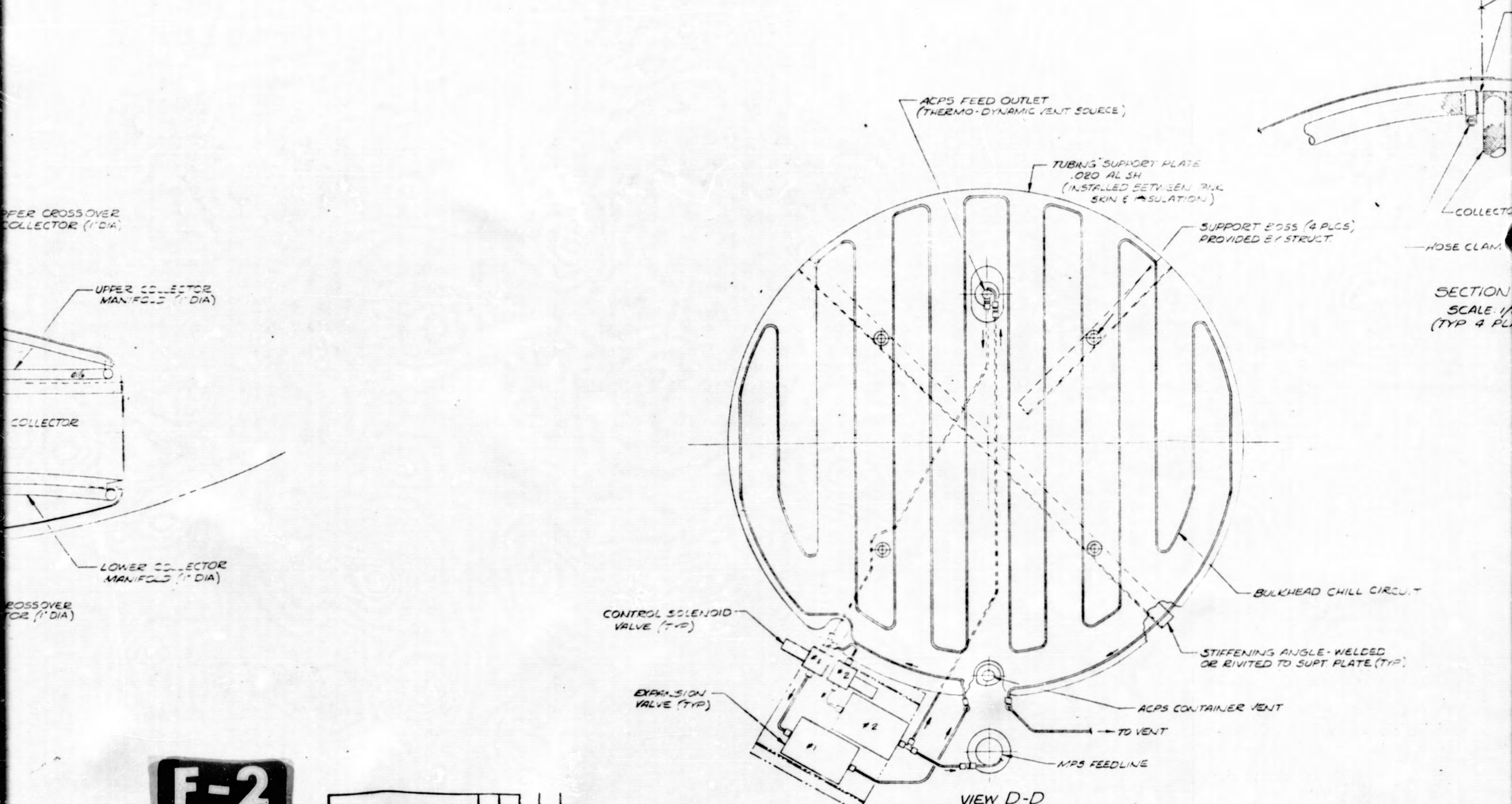
F-1



VIEW D-D
SCALE: 1/4

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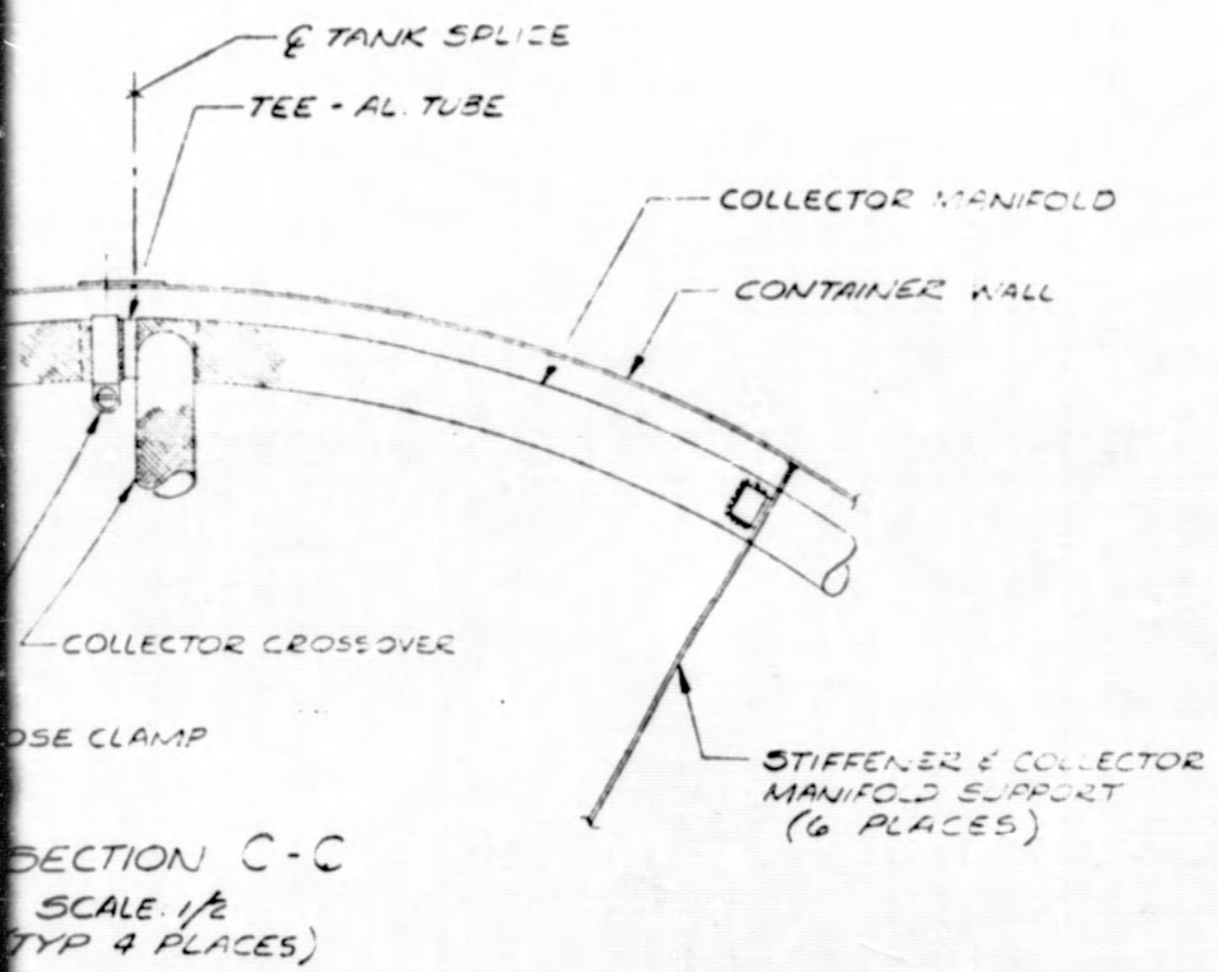
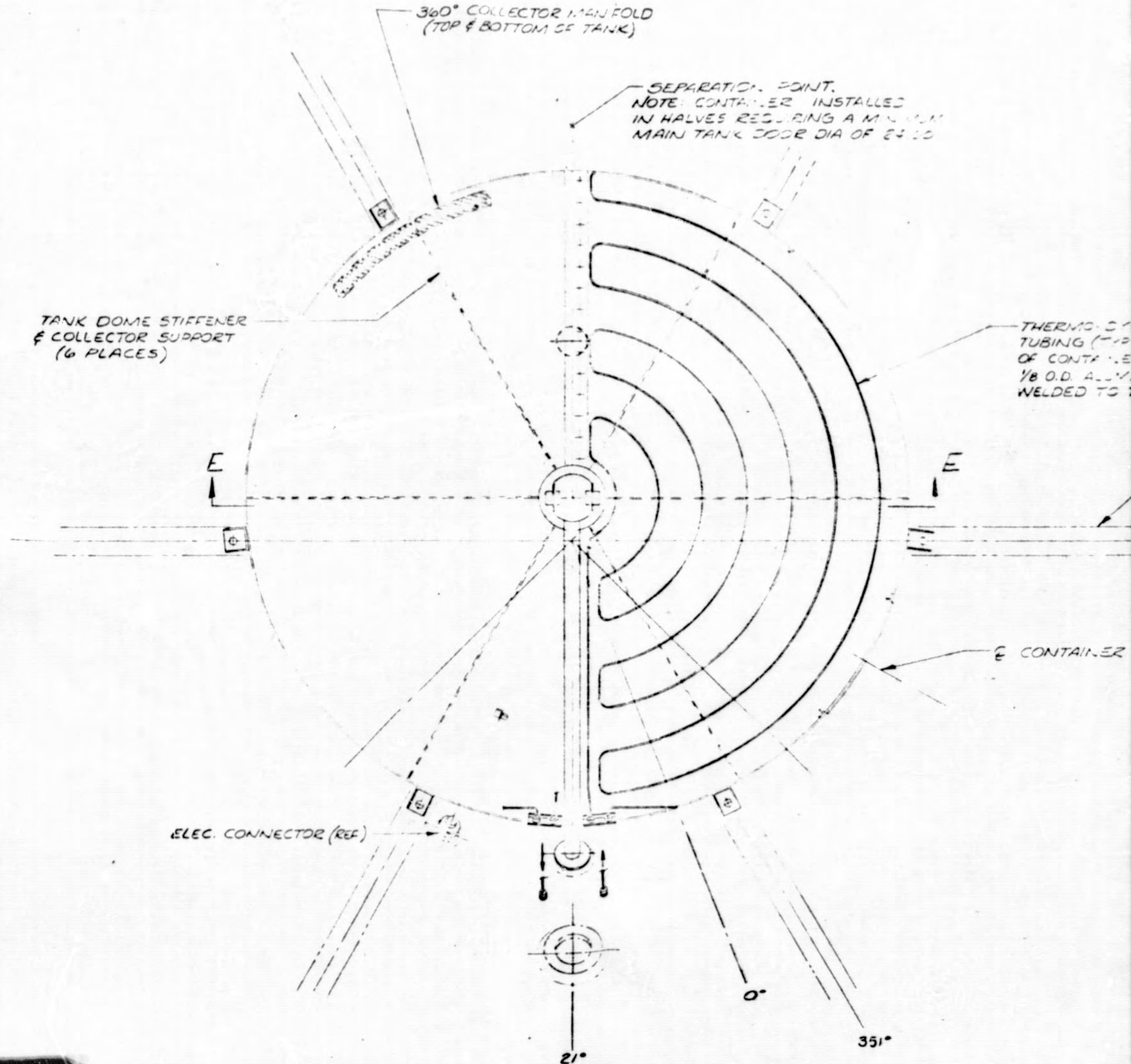


SECTION C-C
SCALE: 1/2
(TYP 4 PLACES)



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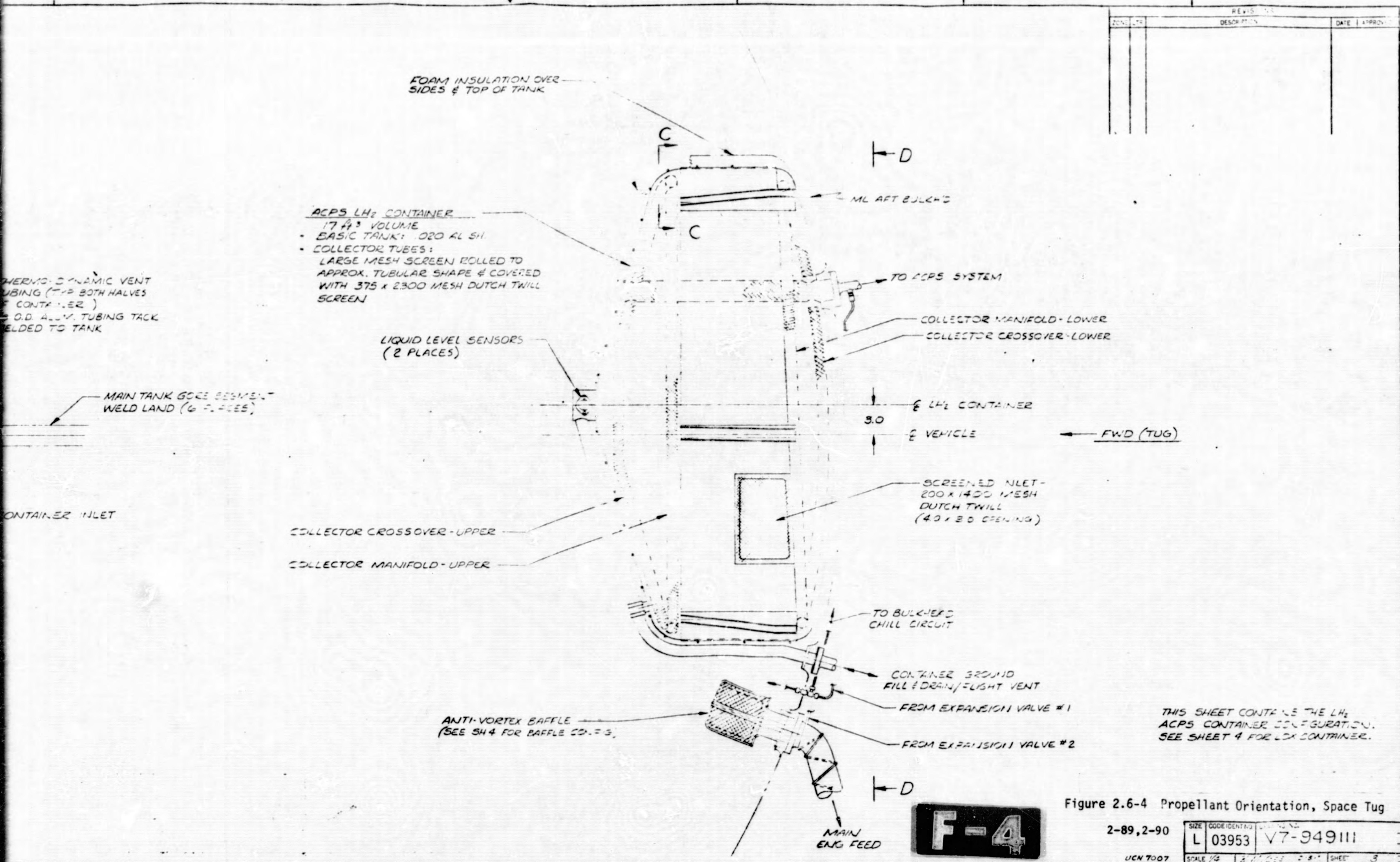

Figure 2.6-4 Propellant Orientation, Space Tug

2-89, 2-90

under the main engine acceleration condition (i.e., 1 to 1.4 g's). As designed, the collector will retain propellant under this condition with a safety factor of 3.5. The need to retain liquid in the collector during main engine burn is one of the factors favoring a short, squat container. Additionally, a short container has lower hydrodynamic loads during settling and slosh. The short squat design also furnishes a false bottom effect reducing main tank residuals.

As pressure difference between inside and outside of the container is quite small, container structural requirements are minimized. The design is based upon a 2 psi differential. In addition to their function of hastening settling, the baffles have design provisions to facilitate refill of the APS container for low main tank propellant levels. The baffles inhibit propellant sloshing which might cause premature uncovering of the refill port. The baffles placement was selected to minimize residual propellant trapped in the main tank.

2.6.3 Concept Selection

Six candidates were evaluated for APS propellant acquisition and feedout. Illustrations of the alternate concepts are shown in Figure 2.6-5. They are: #1, Dedicated Tank; #2, Isolated, Refillable Tank; #3, Integrated Start Tank; #4, Basket, Launched Dry; and #5, Double Basket-Hybrid. All of the concepts require thermal isolation and/or chill sub-cooling, and a screened collector system for propellant acquisition.

The dedicated tank is ground filled with the entire amount of APS propellant required for the mission. A helium pressurization system and an overboard vent for pressure control are required. This system tends to be large and heavy, but may be operationally simpler than some of the other concepts. The isolated, refillable tank is smaller and lighter than the dedicated tank, but introduces the operational complexity of refilling. The integrated start tank combines MPS and APS feed lines. However, as the feed rates requirements of the MPS are about a factor of ten higher than those for the APS, meeting the design requirements for both APS and MPS results in compromised performance and overdesign.

The basket is a screen lined, refillable system. It is light-weight and employs passive vapor purging to the main tank. In addition, the helium pressurization system is no longer required, provided the propellant is maintained subcooled. However, because of the upside down launch the basket is difficult to fill on the ground and even if initially full, will tend to empty during the high launch acceleration. The basket could be launched dry with refill in orbit using accumulator propellants for settling and the first main engine burn for filling. However, even if the basket could be filled in this manner, filling of the screened collector tubes is problematical. The double basket-hybrid is an attempt to get around the problems of the single basket. The small basket and collector tubes could be ground filled. The best (smallest micron rating) available screen and porous materials appear

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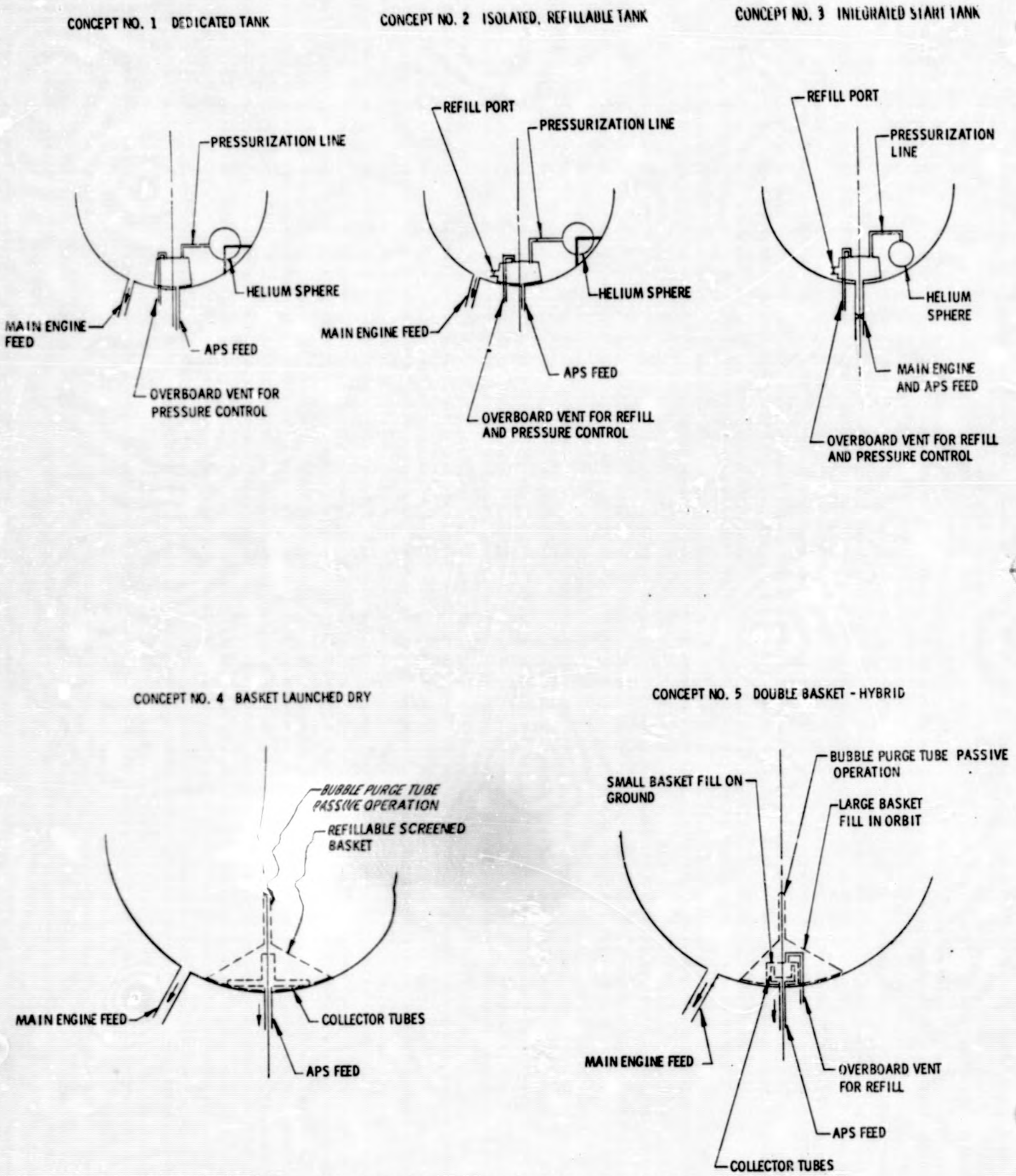


Figure 2.6-5. APS Propellant Acquisition and Feedout Candidates

Space Division
North American Rockwell

capable of retaining propellant in the small basket during launch, although there would be only a small safety factor available. The large basket could readily be filled in orbit during main engine burn. However, overboard venting of the small basket would be required.

The final evaluation involved an in-depth weight comparison between the two most promising concepts, the dedicated tank and the refillable screened container. The remaining candidate concepts were eliminated from further consideration during the initial evaluation for the reasons enumerated above. Figure 2.6-6 illustrates the weight comparison as a function of total APS Δ V propellant. The screened tank is, of course, of much smaller capacity since it is refillable. It is sized to provide 70 percent margin over the required APS ΔV after the last normal refill opportunity, the last MPS burn. (It could be refilled by a special APS settling maneuver in any case). The dedicated tank is sized to allow for 20 percent margin over the reference mission requirements but, as Figure 2.6-6 shows, its weight is not greatly sensitive to capacity in this range. Table 2.6-1 is a breakdown showing the detailed weight comparison. The net difference between the two is 105 pounds in favor of the refillable tank. Together with the mission flexibility, this weight advantage makes the refillable screened container the preferred choice.

2.7 PROPELLANT MANAGEMENT SYSTEM

2.7.1 Function

The function of the propellant management system is two-fold. Primarily, it is to minimize propellant outage (residual usable propellant after the other propellant has been exhausted) by use of a propellant utilization system which measures or infers the propellant remaining in each tank during main engine operation and adjusts the engine mixture ratio (EMR) accordingly. Secondarily, it is to measure the amount of propellants loaded on the ground to ensure that the total is sufficient for mission requirements, and that the mixture ratio is within the limits correctable by the propellant utilization system.

2.7.2 Description and Operation

To implement the requirements of the propellant management system, the point sensor concept of propellant gaging was selected for EMR control (propellant utilization) and capacitance probes were selected for ground loading control (see Figures 2.7-1, 2.7-2).

The point sensor concept uses discrete level sensors located at corresponding levels in each of the propellant tanks. As these pairs of sensors (one in LOX tank and one in LH_2 tank) are uncovered due to propellant consumption, the time between the uncovering of each sensor of the pair is noted by the on-board computer, the amount of remaining propellant in each tank is determined, and the EMR is commanded to the appropriate setting for

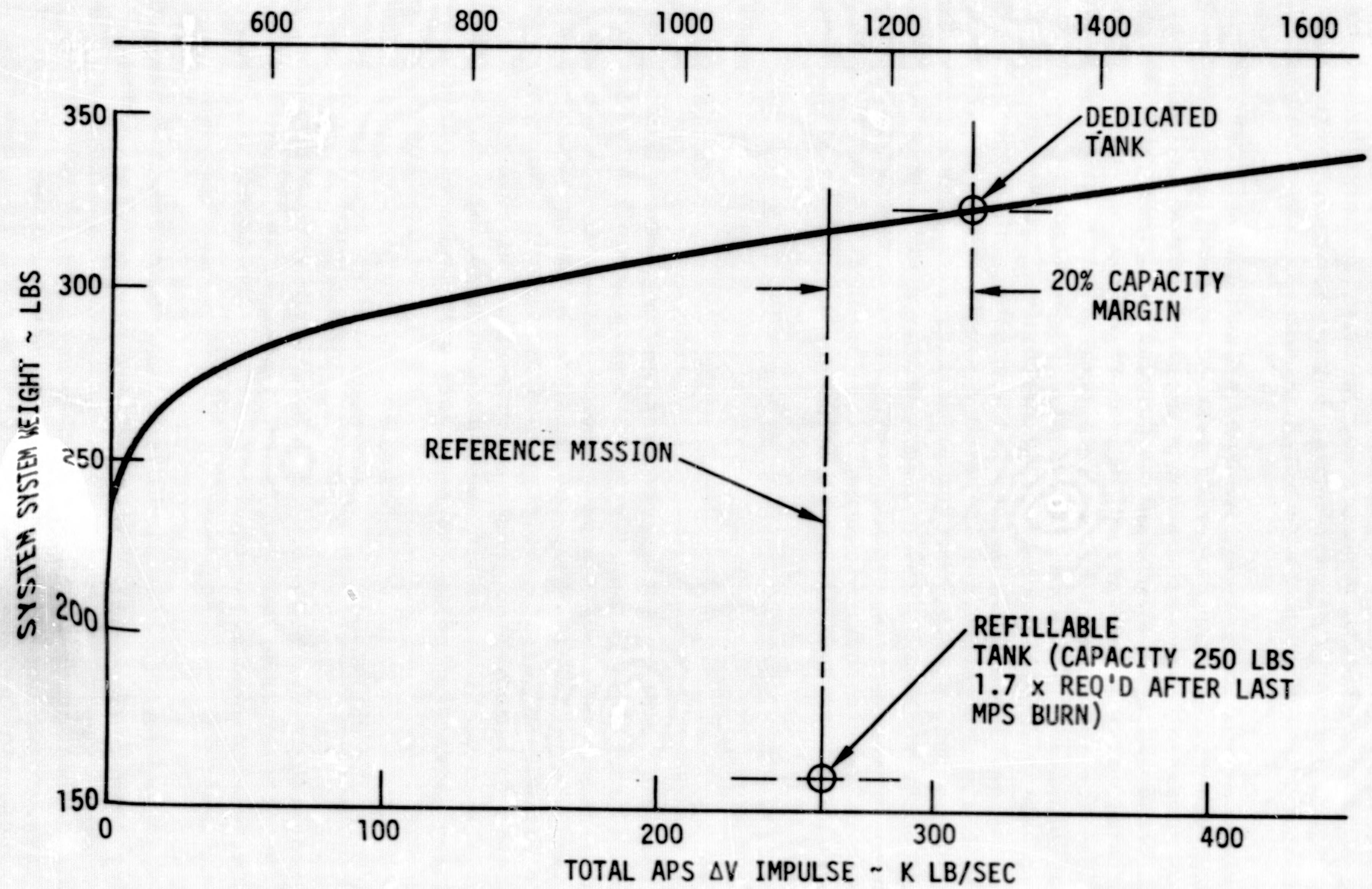
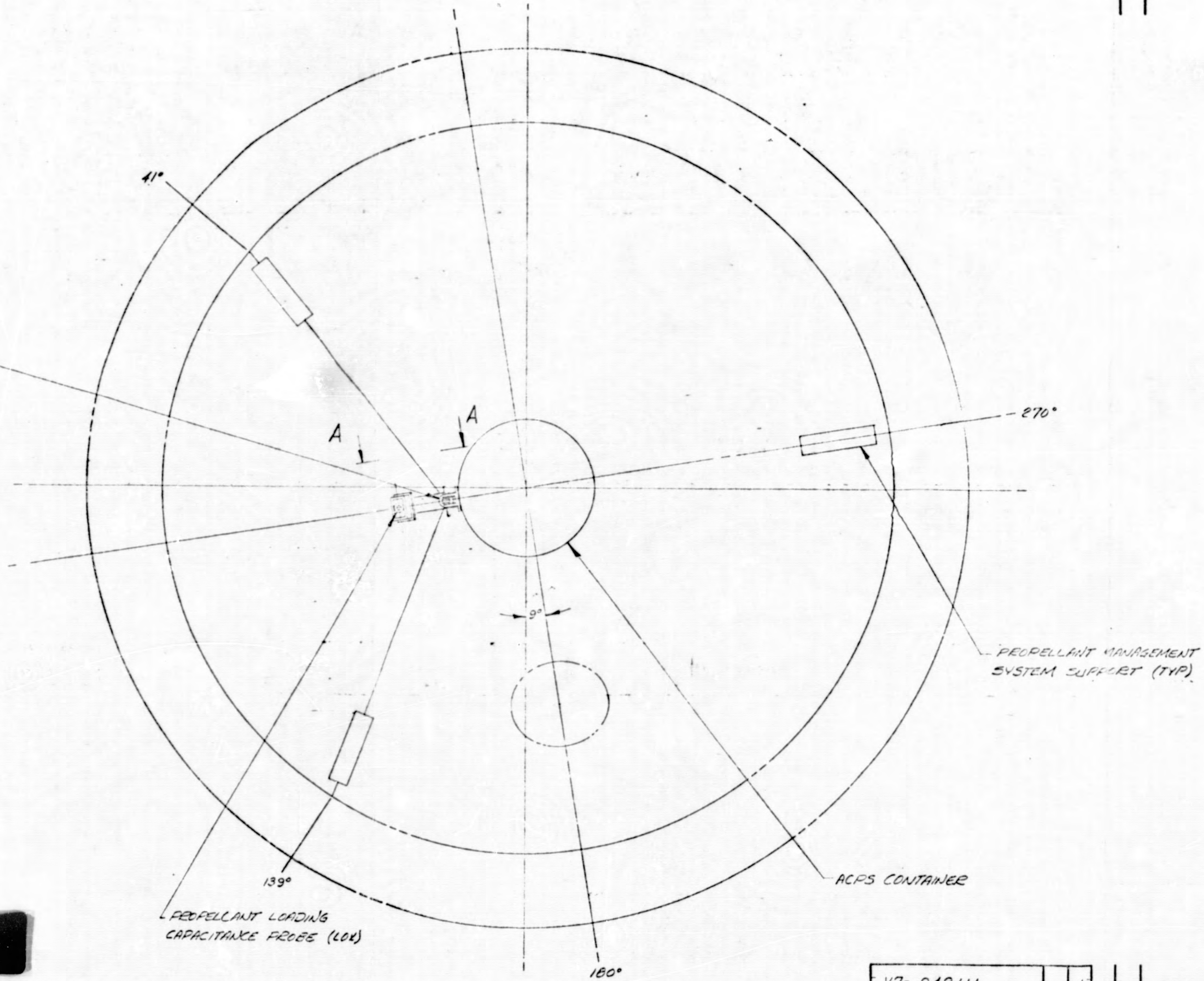


Figure 2.6-6 APS Tank Weight Comparison



41°
POINT SENSOR
STILLWELL (LOX)
A
A
270°
9°
PROPELLANT MANAGEMENT
SYSTEM SUPPORT (TYP)
139°
ACPS CONTAINER
PROPELLANT LOADING
CAPACITANCE PROBE (LOX)
180°

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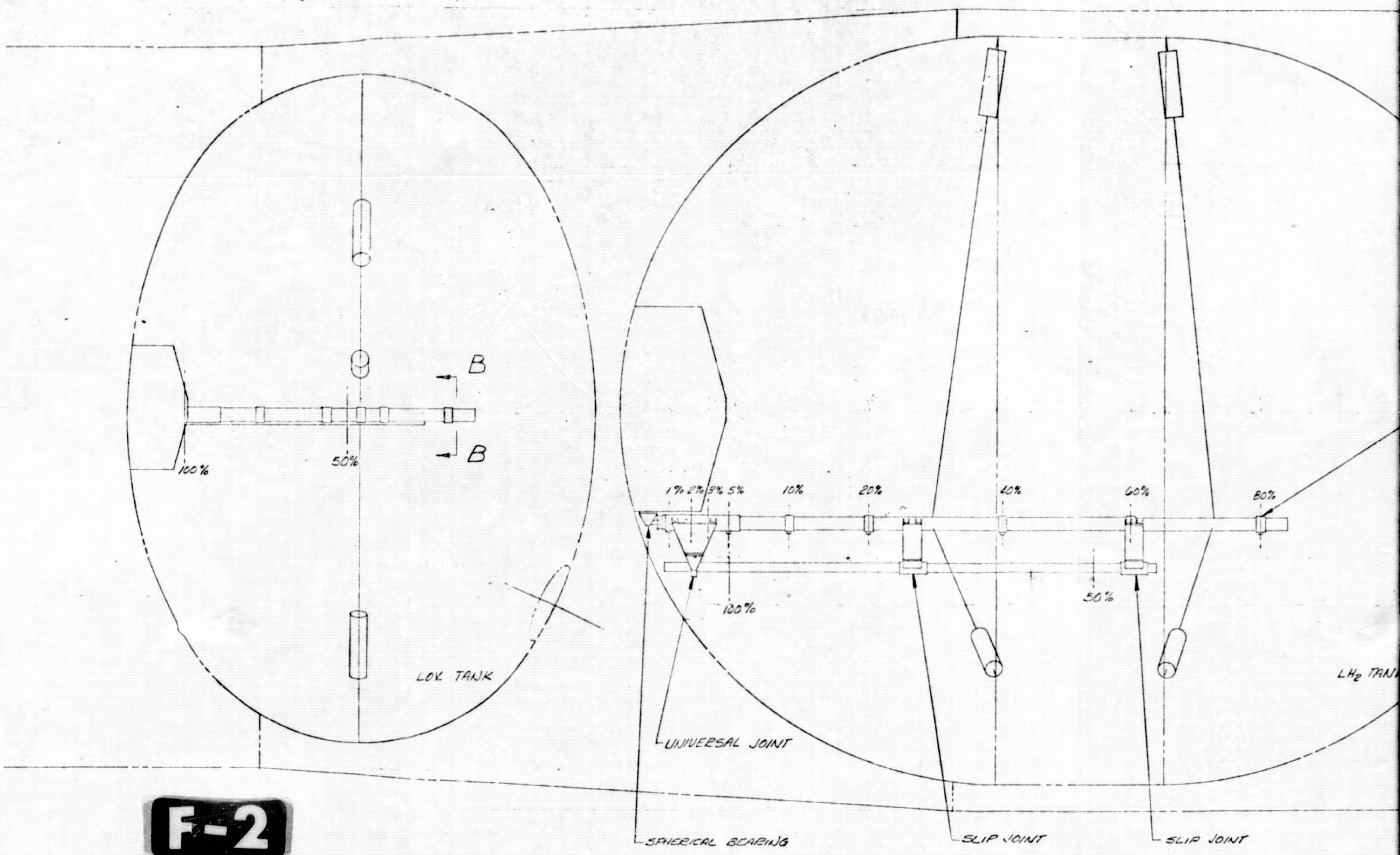
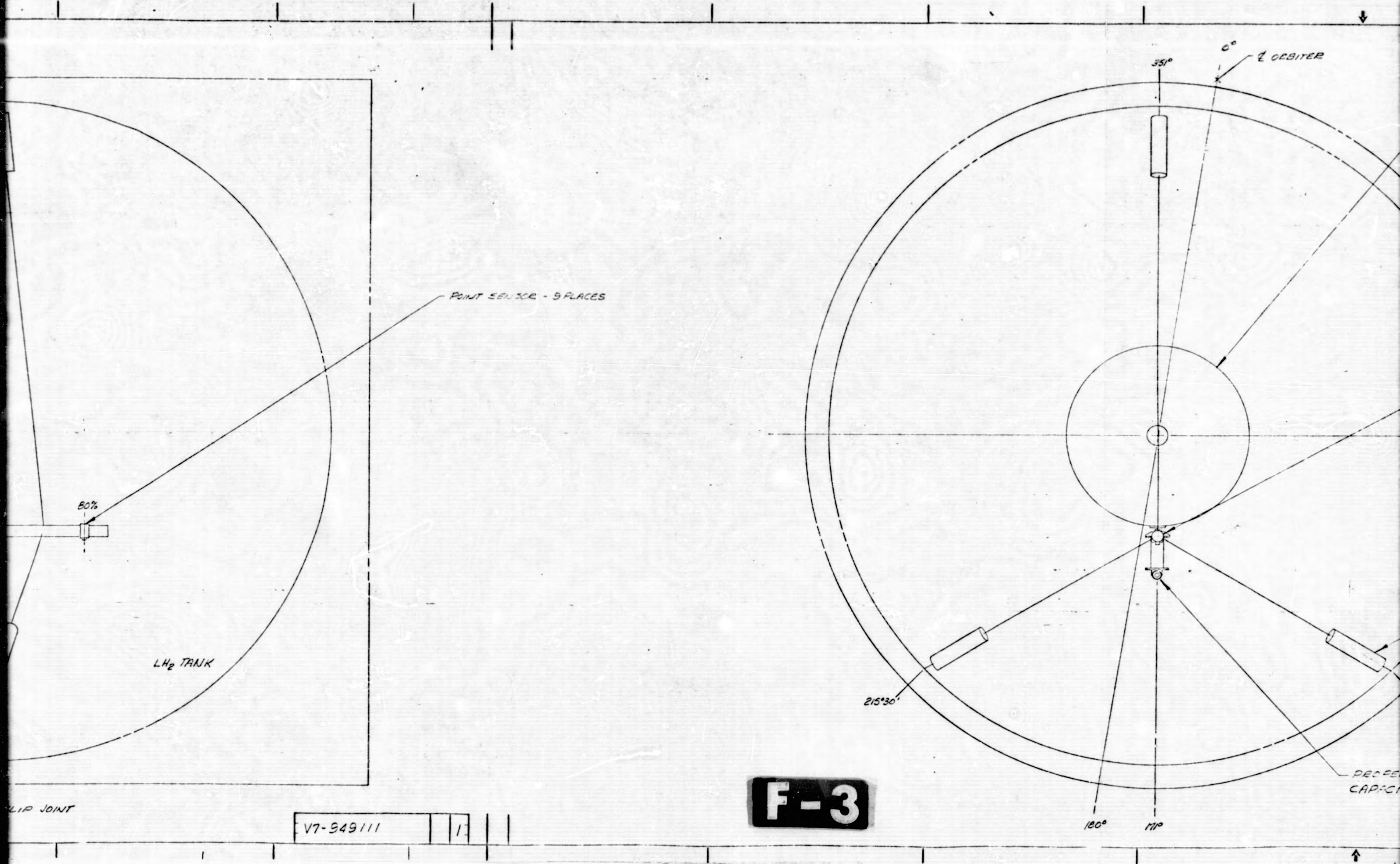


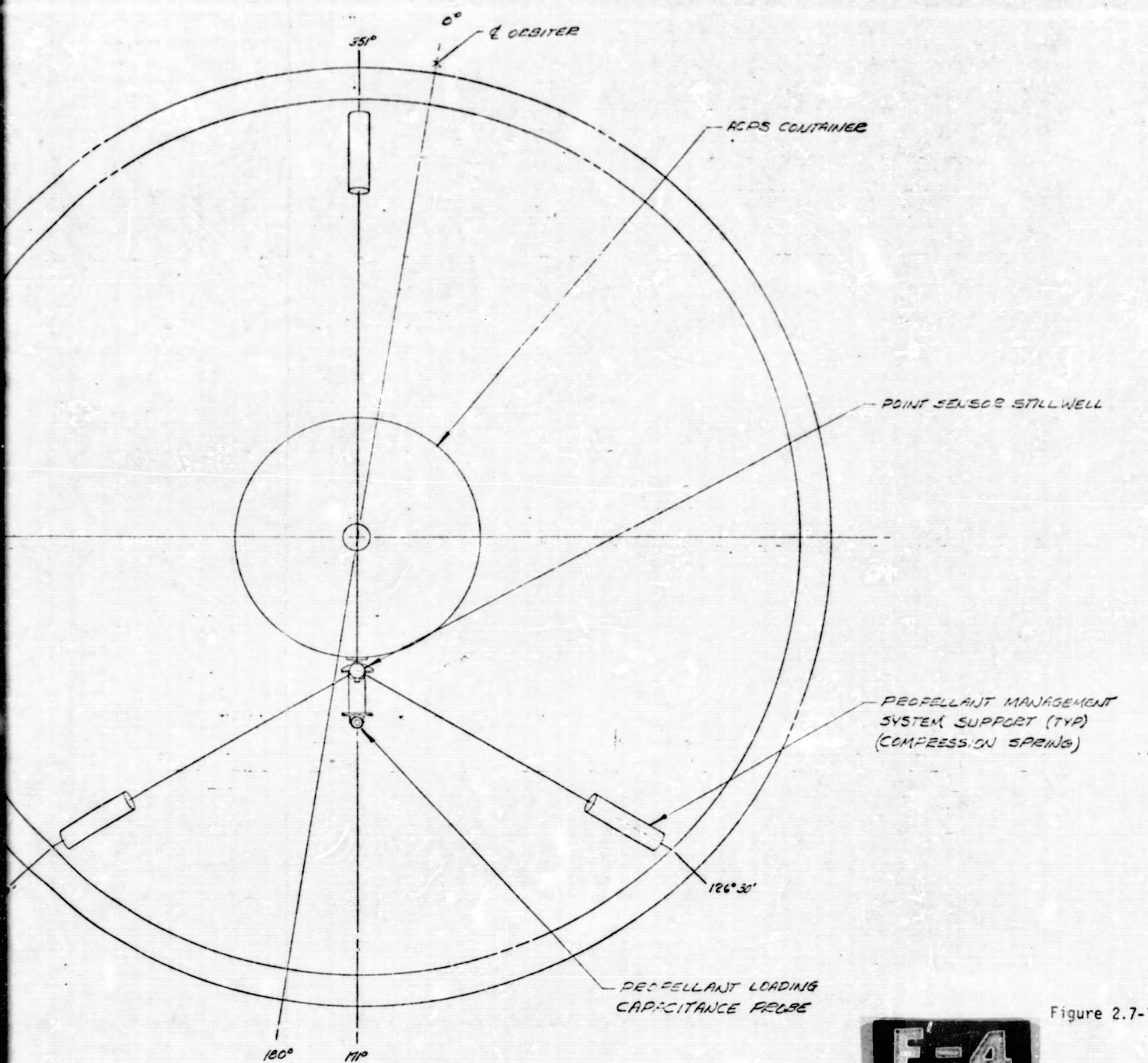
B
B
100%
50%
LOX TANK
1% 2% 3% 5%
10%
20%
40%
60%
80%
100%
50%
LH2 TANK
UNIVERSAL JOINT
SPHERICAL BEARING
SLIP JOINT
SLIP JOINT

F-2



POINT SENSOR - 9 PLACES
80%
LH2 TANK
LIP JOINT
V7-949111
1
0°
℄ ORBITER
351°
215°30'
180°
MP
PROFE
CAPACI

F-3



REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED

3. THIS SHEET CONTAINS THE PROPELLANT MANAGEMENT SYS.
2. MATERIAL - ALUMINUM (EXCEPT AS NOTED)
1. SEE SHEET 3 THRU 5 FOR PROPELLANT ORIENTATION SYSTEM

NOTES:

Figure 2.7-1 Layout - System Installation, Propellant Orientation & Management

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UCN 7007

DWG BY J R MOORE DATE 1-7-72

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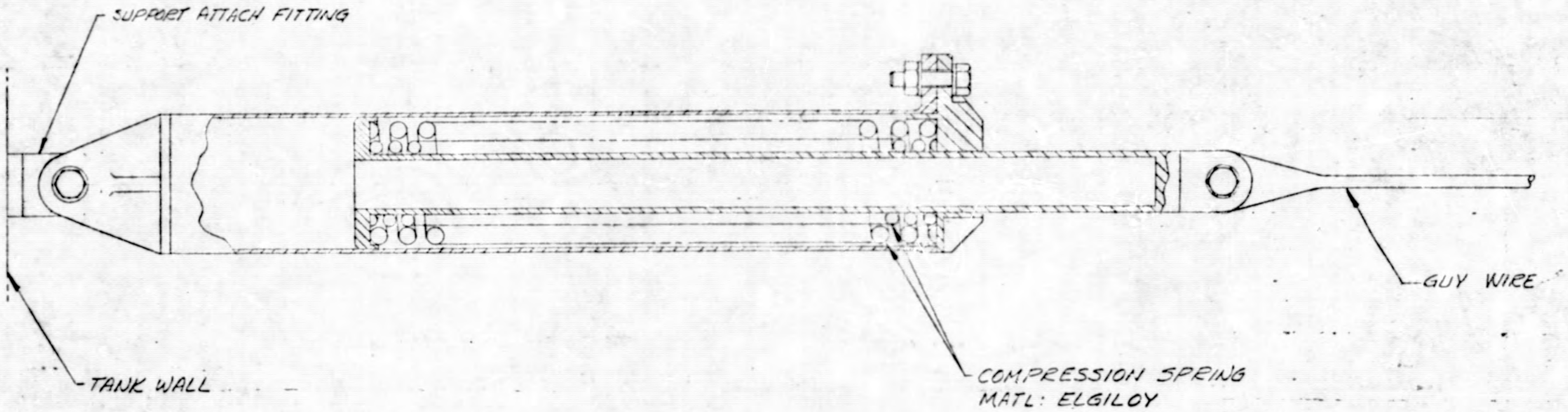

PROPELLANT MANAGEMENT SUPPORT DEVICE

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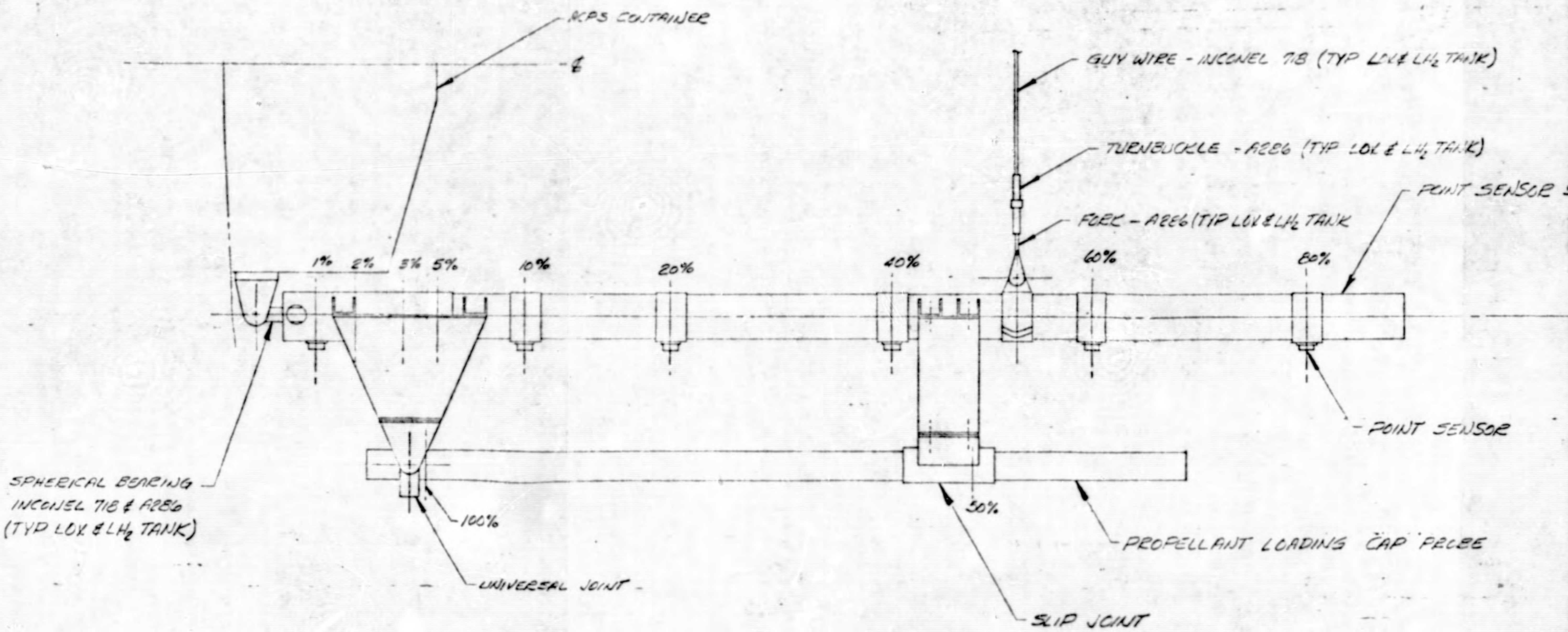

VIEW A-A
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F-2

V7-949111 | 2

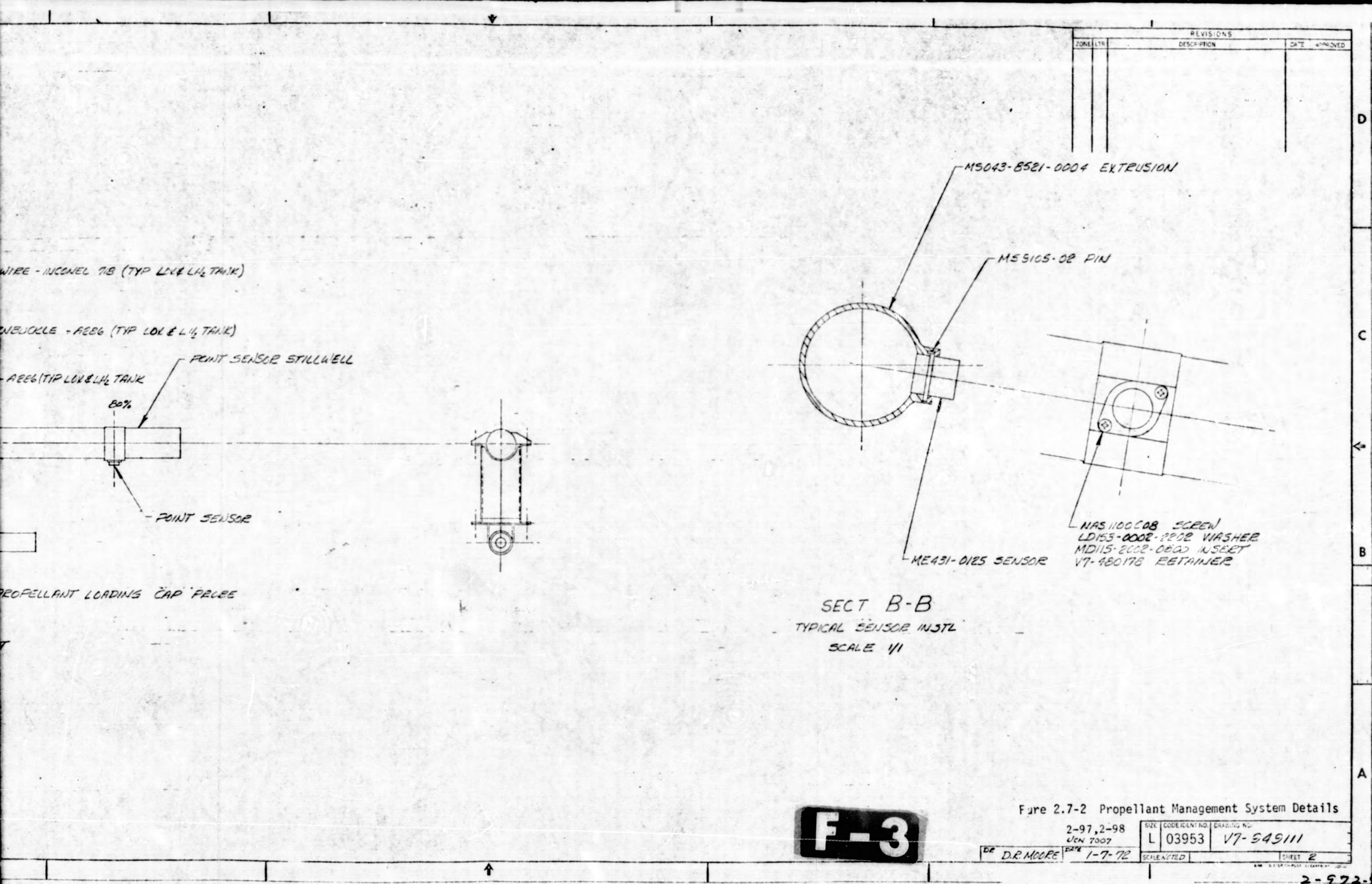

Figure 2.7-2 Propellant Management System Details

Table 2.6-1. APS Propellant Acquisition Weight Comparison

	Dedicated Tank			Refillable Tank		
	LOX	LH_2	Total	LOX	LH_2	Total
Tank Shell	19	51	70	9	19	28
Support Structure	4	10	14	2	3	5
Internals	5	5	10	5	5	10
Vent and Boiloff at .3 effective inert						8
Helium vessel and He	-	-	26	-	-	-
Lines Req and Exchanger	-	-	4	-	-	-
Residuals (Trapped)						
Main Tank	156	5	161	*156	5	*161
Aux Tank (3% - 5%)	29	9	38	12	4	16
Total			323			218
Δ Weight			+105			0
Volume, Cu ft	15.2	73.7	-	3.3	17	

*Potentially Less

the interval computed necessary to correct the mixture ratio consumed to the mixture ratio loaded. For the Tug design, the following criteria were used:

1. The initial tanked mixture ratio is 6.0.
2. The engine will have only three discrete mixture ratio settings; high, low, and nominal. The nominal setting will be 6.00 and the low setting will be 5.50. To reduce the engine performance loss and high operating temperatures resulting from the peak MR setting, the peak setting will be reduced to a value just sufficient to permit an initial tanked mixture ratio of 6.00.
3. The last control sensor will be located at the 10 percent level and will be positioned to result in equally probable outage (this is near optimum) for LOX and LH_2 at an EMR of 6.00.

Based on the above criteria, a system was designed that yielded the parameters of Table 2.7-1 for a representative case.

Table 2.7-1 - Engine Mixture Ratio Range

Nom	6.00 ± 0.12
Low	5.50 ± 0.12
High	6.22 ± 0.12
Initial tanked mixture ratio	6.0
Initial load (Typical)	55,500 lb
Outage	64 lb
Sensor Location (% of Total Load)	
Control Sensors	Instrumentation Sensors
80	5
60	3
40	2
20	1
10	

The operational concept of the system is as follows: The tanks are initially loaded to a mixture ratio of 6.0. The nominal engine operation is at the nominal EMR of 6.0. If the 80 percent sensor pair uncovers within the specified time lag, the engine will continue to run at an EMR of 6.0. If the 80 percent sensor pair does not uncover within the specified time lag, the computer commands the engine to either high (6.22) or low (5.5) EMR depending on which sensor uncovers first. It will remain at this EMR for the length of time determined by the computer or until the next sensor pair picks up within the specified time lag. This sequence is repeated as each sensor pair is uncovered until the 10 percent level. From this point on, the system is open loop. If this sensor pair is uncovered within the specified time span, the remaining operation will be at an EMR of 6.0. If not, the remaining operation will be either at high or low EMR, as required. Those sensors below the 10 percent level are presently planned only for instrumentation to improve the calibration and accuracy of the 10 percent sensor. These were not used in the analysis model since not all mission timelines would permit long enough burns below this level for adequate control response.

The use of mixture ratio changes to minimize residuals will not appreciably affect overall Tug performance as the operating time at the off-nominal mixture ratio is expected to be short and the Isp changes with changes in mixture ratio are small.

The capacitance probe loading system will consist of a probe in each tank and ground electronics. The range of the capacitance probes will permit varying the mission loads in each tank from 50 percent to 100 percent. The estimated accuracy of this system is 1.25 percent for LH_2 and 0.62 percent for LOX.

2.7.3 System Selection and Options

Since the purpose of the propellant utilization system is to minimize the propellant outage, the system that best accomplishes this with a minimum of complexity should be used. Looking first at the utmost in simplicity, which is no control system at all, the outage can be reduced to 0.6 percent of the total load by fuel biasing (assuming loading errors of 0.62 percent for LOX and 1.25 percent for LH_2 and an EMR error of 2.0 percent). Considering next a closed-loop point sensor EMR control and propellant utilization system, the outage can be reduced to 0.15 percent of the total load with the same loading and EMR errors. Although implementation of such a system requires the addition of approximately 30 lbs hardware weight, the overall stage weight savings is approximately 250 lbs. Thus a closed-loop control system is beneficial.

Although the preceding rationale shows that a closed-loop system is preferable to an open-loop system, it does not prove that the point sensor gaging system is the optimum closed-loop system. A capacitance probe gaging

system is also capable of reducing the propellant outage to less than 0.25 percent of the total load with approximately the same additional hardware weight. However, the following items must be considered:

1. Use of the point sensor system involves only simple electronics. It provides a good quality (high signal to noise ratio) discrete or digital signal to the computer as opposed to a low level analog signal of the capacitance probe. It is not susceptible to system drift.

2. Since the capacitance probe is based on the difference in dielectric constants of the propellant liquid and vapor and the point sensor is primarily on the difference in density, the physical sensitivity (liquid to gas property ratio) of the point sensor system is much higher:

	Point Sensor	Cap Probe
LOX	350	1.5
LH_2	300	1.2

3. From the reliability standpoint, the point sensor system is preferable since the loss of a single sensor does not render the system useless as does the loss of a capacitance probe.

Based on these factors, the point sensor was selected for the gaging element of the propellant utilization system.

To control the amount of propellants loaded on the ground, point sensors and a capacitance probe were again compared. In order to permit loading of the vehicle to any level between 50 and 100 percent, the point sensor loading cluster would have to be adjustable whereas a single capacitance probe could extend over this entire length. Although both systems provide the required accuracy (0.62 percent for LOX and 1.25 percent fo LH_2) for approximately the same hardware weight (30 lbs), the capacitance probe was selected to avoid the operational problems involved with the readjusting of the point sensor loading cluster. On-board electronics to permit the probe to supplement gaging for propellant utilization is not advisable since the added weight would be nearly equal to the gain.

2.7.4 Loading Tolerance Analysis

An analysis was conducted to provide an estimate of the propellant loading error for the Tug vehicle. The results of this study are presented in Table 2.7-2 for a representative case. The analysis indicates that a mean excess load of 300 lbs should be included in the gross weight and performance calculations for the shuttle and Tug vehicles together with any other weight and performance variations. It is noted that this loading tolerance does not affect the Tug stage performance, since the excess propellant if present all or in part, may be consumed to achieve the programmed ΔV. However, since the

Table 2.7-2. Representative Propellant Loading Error for the Tug

	OX	Fuel	Total
3σ loading error - %	±0.6	±1.25	--
3σ loading allowance - %	+1.2 -0.0	+2.5 -0.0	--
Propellant - lbs Required Usable			55,200
Nominal Load	47,571	7929	55,500
3σ loading error	283	99	*300
3σ maximum load	47,854	8028	55,800
Maximum capacity (with 4% ullage)			56,394

*RSS 3σ probability

primary source of the variations involves the hardware dimensional tolerances, the tanks are designed for adequate ullage volume using the maximum 3σ load.

As a point of comparison, it is noted that the current SII loading error is 0.55 percent and 0.65 percent for oxidizer and fuel, respectively. However, the Tug insulation (MLI) is not effective on the ground therefore contributes to bubble formation and increases loading density error. The estimates reflect this consideration.

It is concluded that the propellant loaded weight tolerances are small compared to other unidentified system weight variations and this factor should not be introduced into the studies until later in the shuttle/tug system design evolution.

2.7.5 Recommendations

The use of an open loop system for propellant utilization should be reconsidered in the event that loading and EMR errors can be significantly reduced from the values used in this study. Depending on the extent of off-loading and the performance criticallity of missions requiring it, the capacitance probe loading function could be fulfilled by the outage control point sensors (or a few added ones). This would result in a weight saving of 20 lb.

2.8 MAIN ENGINE

2.8.1 Introduction

The engine design used in this study is as specified (Ref. 2.8-1) as the baseline for the contract effort. In some areas of detail it was necessary

to supplement the baseline data. Additional sources of information are described in the following paragraphs, and included the engine candidate contractors (Ref. 2.8-2,4,5) as well as data previously developed by the USAF Space and Missile Systems organization (SAMSO) for the 1971 orbit-to-orbit shuttle (OOS) tug-related study.

Figure 2.8-1 is the configuration definition furnished by Ref. 2.8-1. Figure 2.8-2 is a representative schematic largely taken from Ref. 2.8-4 for a staged combustion engine with pressure-fed idle capability.

2.8.2 Engine Design Description

The propulsion system incorporates one head mounted, gimballed, bell engine utilizing staged combustion with parallel turbines. Low speed boost pumps are utilized to meet the required main pump inlet conditions. The engine assembly incorporates a preburner to supply gas to drive the turbines and necessary valves, controls, and instrumentation. The propulsion components are mounted from the injector-thrust chamber structure. Internal plumbing details have not been specified by NASA but it is assumed that internally constrained flexural sections located below the engine interface will be used. The two inlet lines are to be symmetrically located about the gimbal point to avoid a continuous pressure-induced force on the gimbal actuators. The engine system will utilize helium for purge gas and incorporates a helium distribution manifold with necessary valves and controls. The engine assembly fluid schematic is presented in Figure 2.8-2.

Interfaces

Interfaces between the engine assembly and the Tug stage involve the structural pivot point mounting, the pitch and yaw gimbal actuator attachments, the fuel and oxidizer propellant lines, the LOX and LH_2 tank pressurization lines, a helium supply line and the various engine thrust, mixture ratio, thermal control, and instrumentation electrical wiring. The engine installation arrangement is shown in Figure 2.8-3. Dimensions assumed for the engine are presented in Figure 2.8-4. The propellant line interface is based on the use of flexures located below the interface as part of the engine.

The engine uses electrical power from the Tug vehicle systems for the control system electronic control unit, solenoid valves, electrical valve actuators, and the ignition system. All electrical components except the ignition system require 28 volts dc; the ignition system requires 115 volts, 400 Hz ac, but an inverter provided as part of the engine eliminates them as an interface supply requirement. The power requirements are shown in the following table taken from reference 2.8-3.

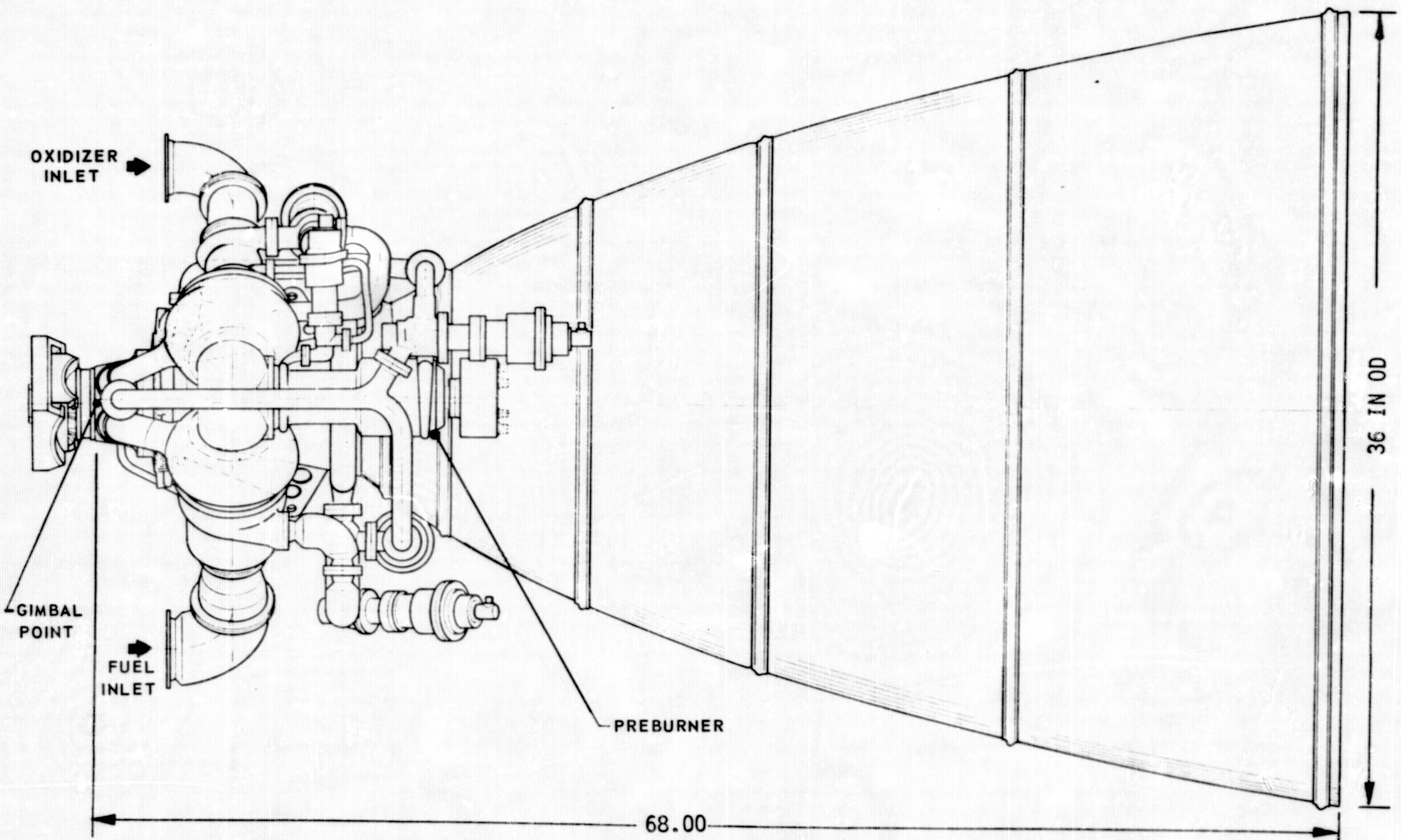

Figure 2.8-1 10K Baseline Engine Configuration

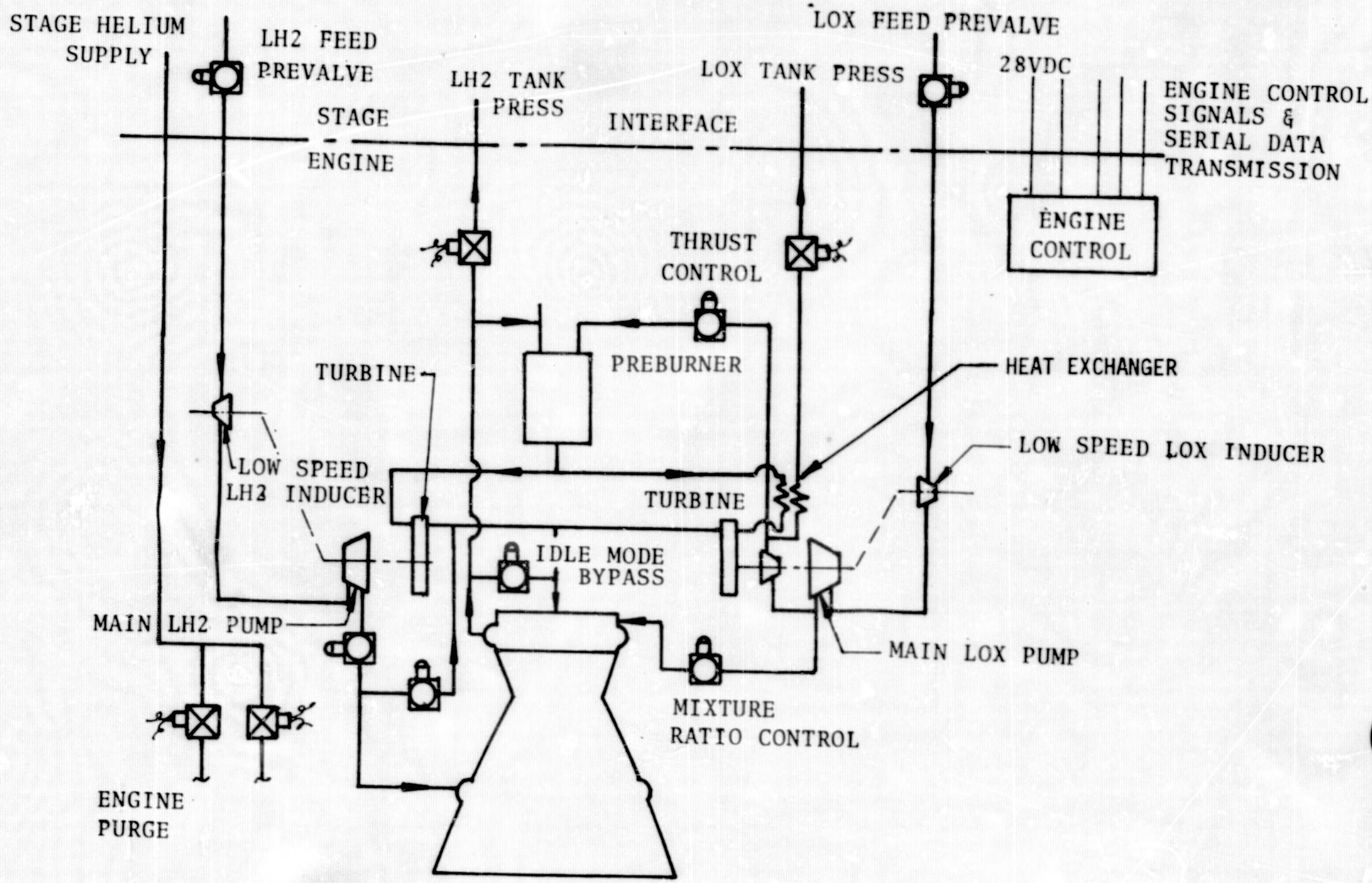

Figure 2.8-2 Tug Main Engine Schematic

D

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B

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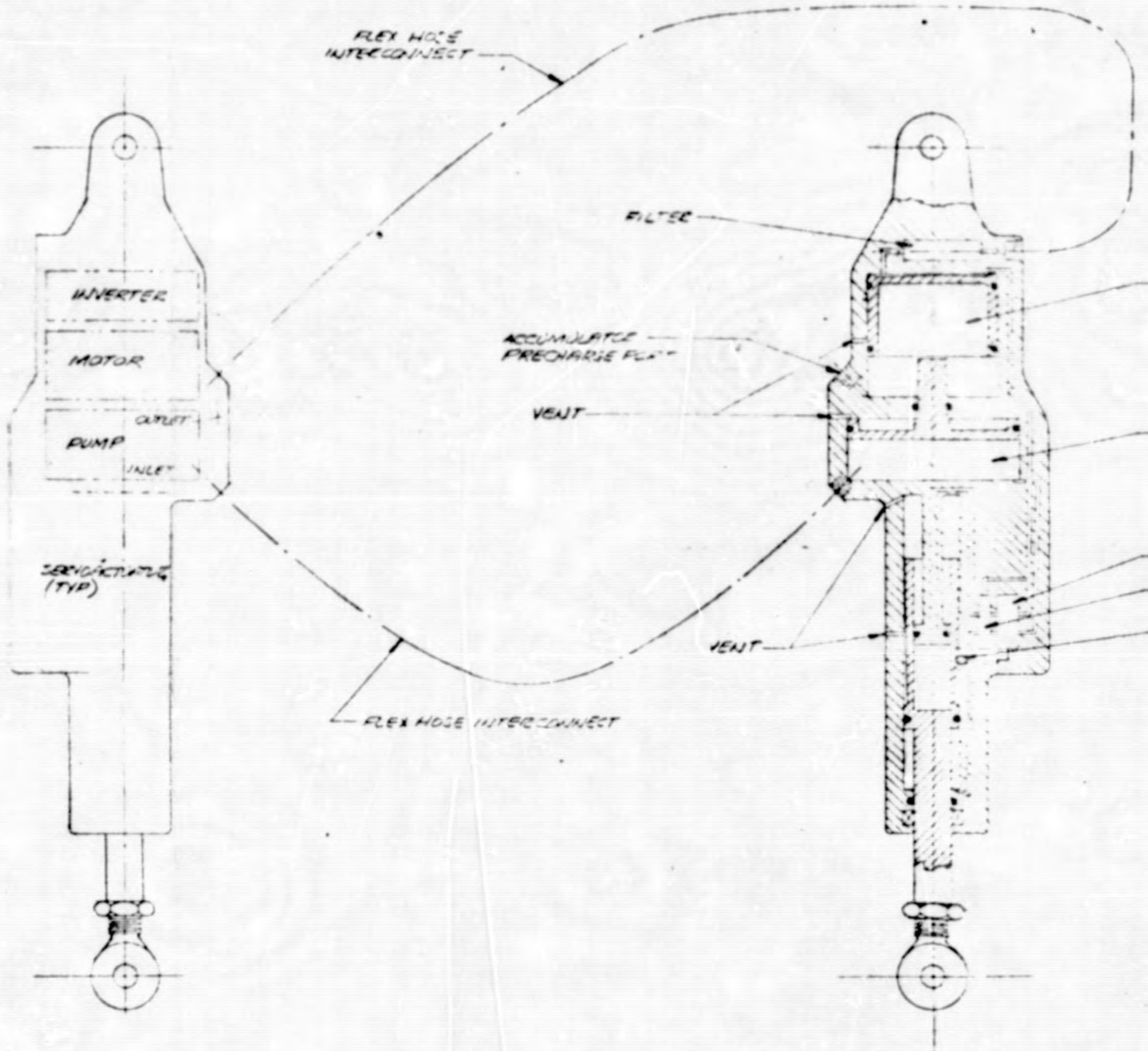

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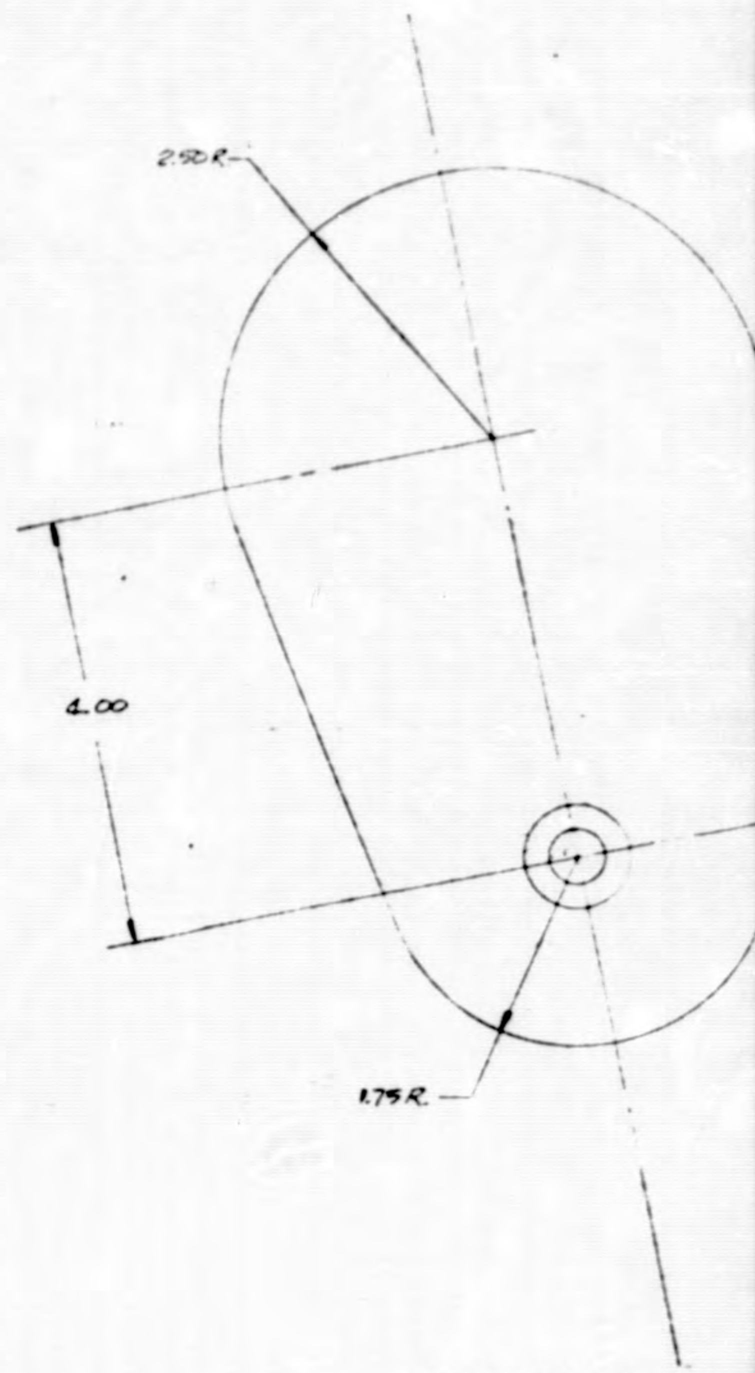

ACTUATOR SECTIO
OUTLINE INCLUDI
INSULATION
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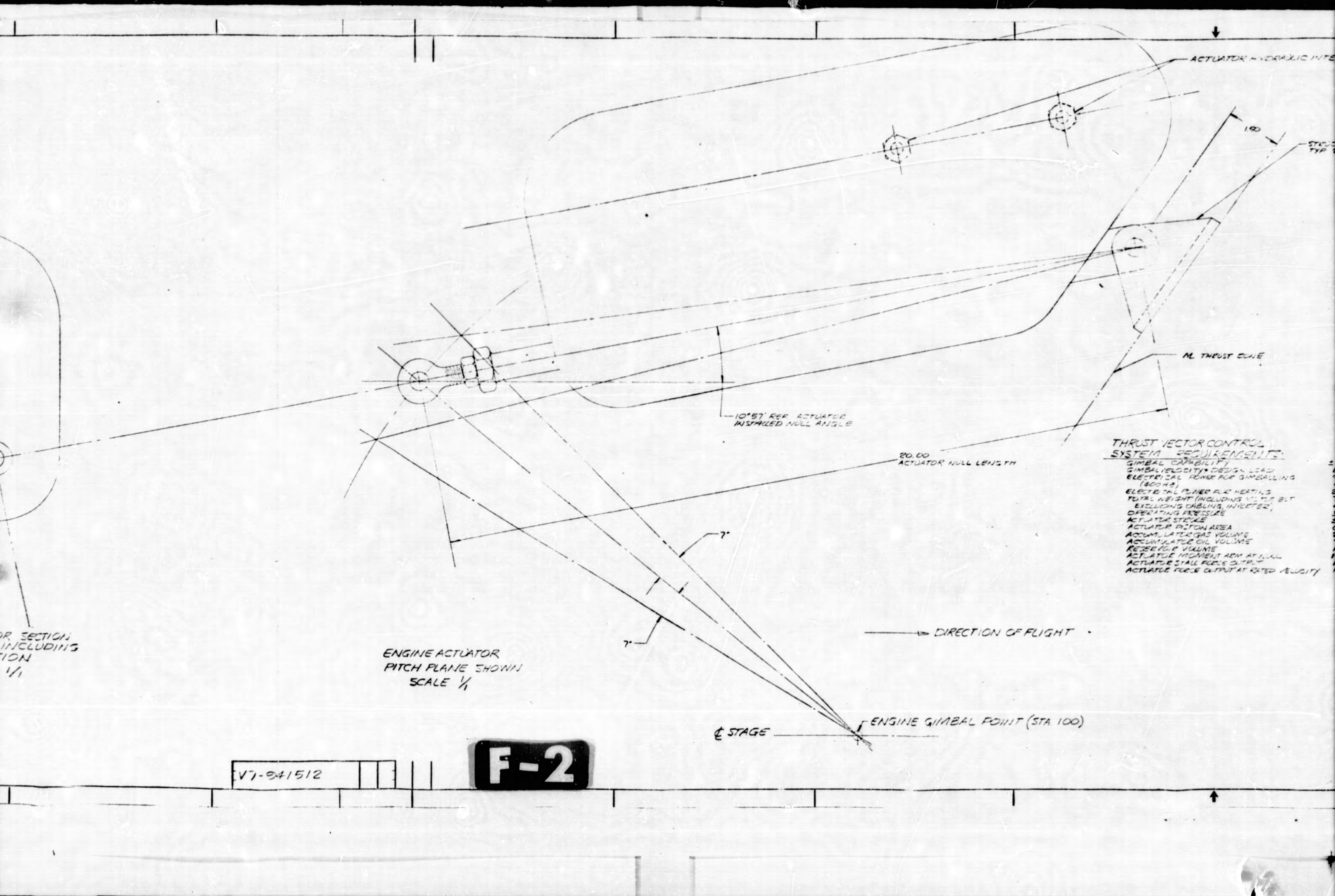


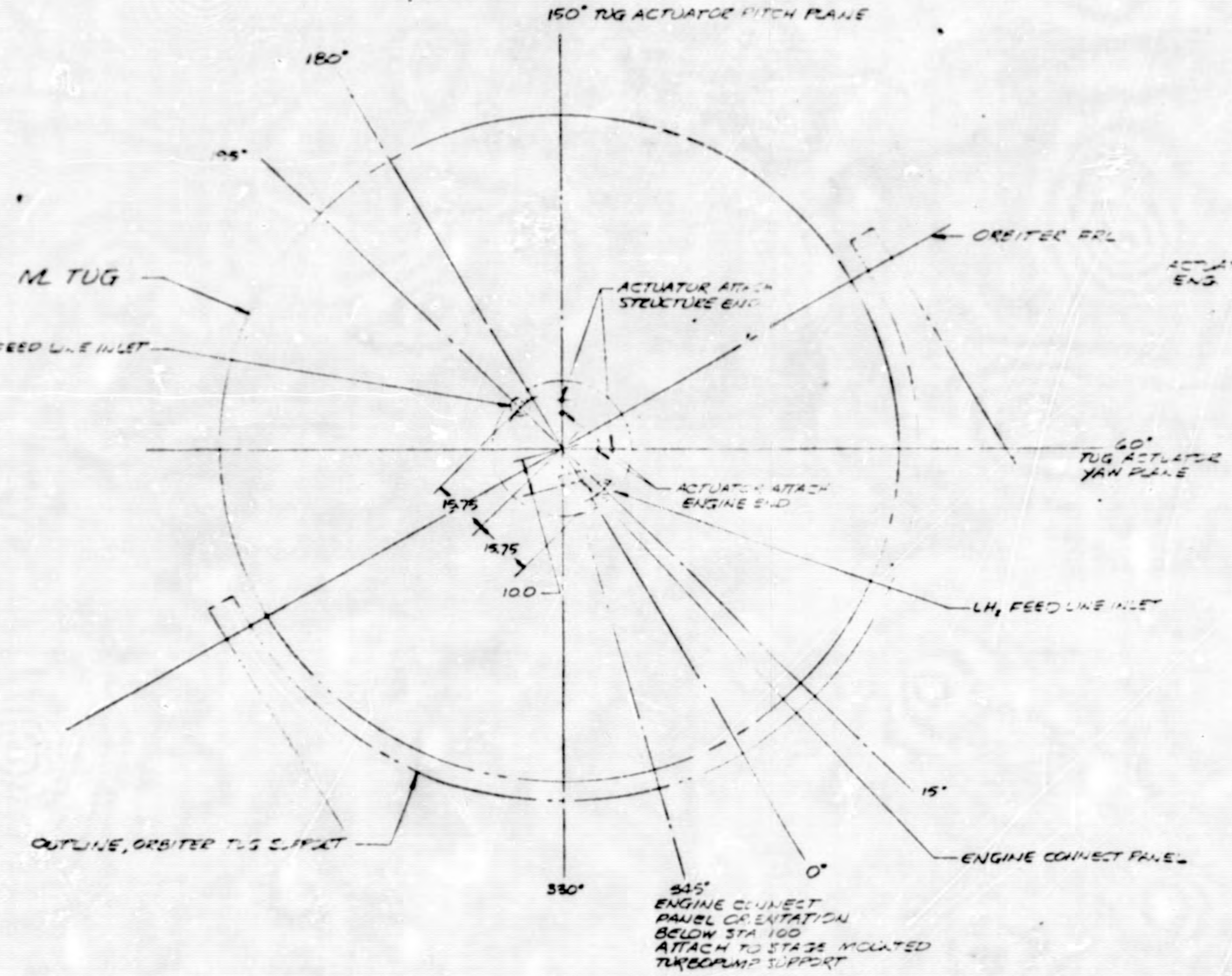
ACTUATOR HYDRAULIC INTE
1.90
STRUC TYP
M. THRUST CONE
10°57' REF. ACTUATOR INSTALLED NULL ANGLE
20.00
ACTUATOR NULL LENGTH
THRUST VECTOR CONTROL SYSTEM REQUIREMENTS:
GIMBAL CAPABILITY
GIMBAL VELOCITY @ DESIGN LOAD
ELECTRICAL POWER FOR GIMBALLING (400 HZ)
ELECTRICAL POWER FOR HEATING
TOTAL WEIGHT (INCLUDING MOTOR BUT EXCLUDING CABLING, INVERTER)
OPERATING PRESSURE
ACTUATOR STROKE
ACTUATOR PISTON AREA
ACCUMULATOR GAS VOLUME
ACCUMULATOR OIL VOLUME
RESERVOIR VOLUME
ACTUATOR MOMENT ARM AT NULL
ACTUATOR STALL FORCE OUTPUT
ACTUATOR FORCE OUTPUT AT RATED VELOCITY
7°
7°
DIRECTION OF FLIGHT
OR SECTION
INCLUDING
TION
1/1
ENGINE ACTUATOR
PITCH PLANE SHOWN
SCALE 1/1
ENGINE GIMBAL POINT (STA. 100)
℄ STAGE
V7-941512
F-2

...DRAULIC INTERCONNECT

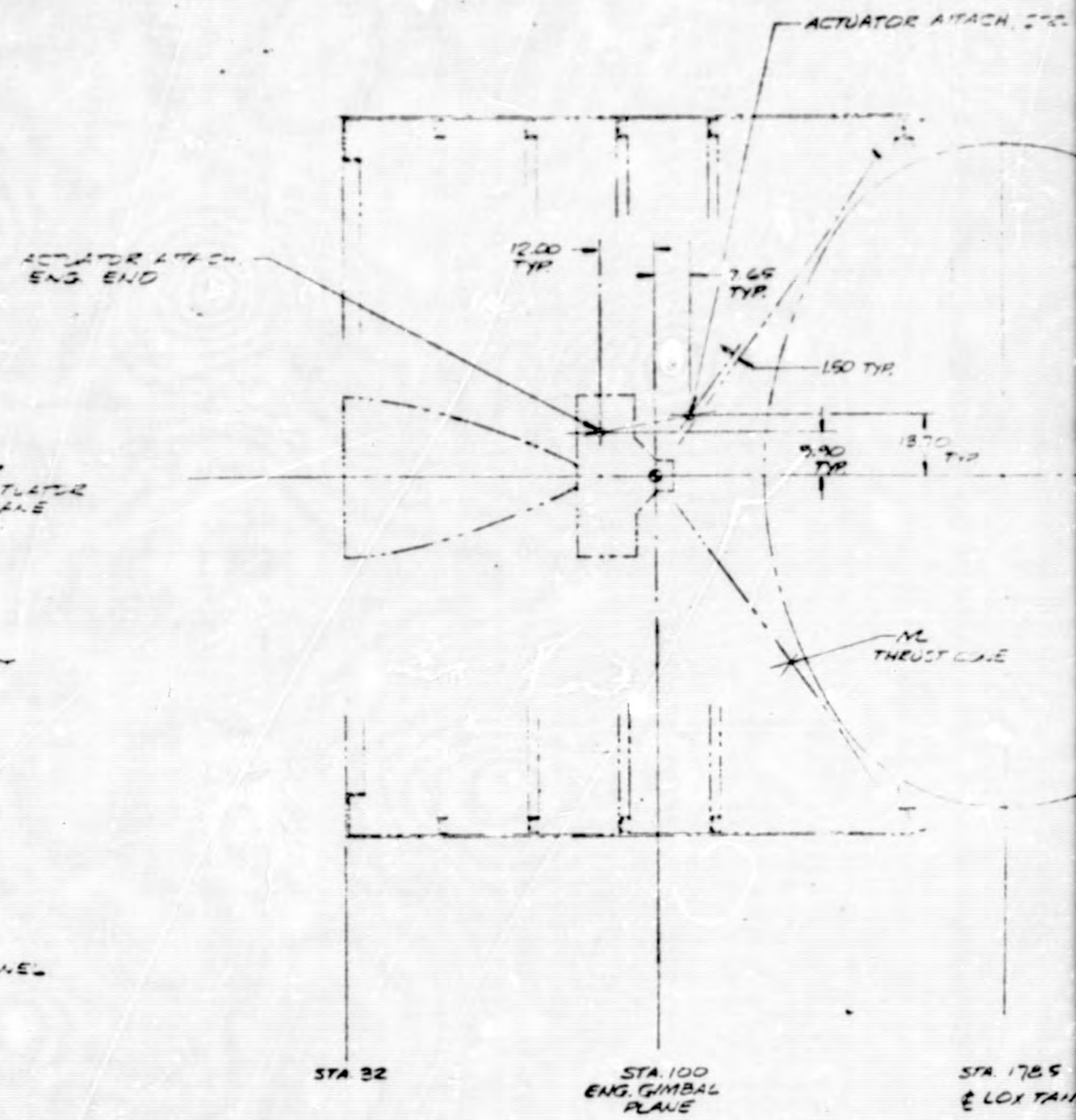
STRUCTURE ATTACH FITTING
TYP 2 PLACES ON THRUST CONE

...TS:

	±7 DEGREES
...AD	8 DEG./SEC.
...ALLING	900 WATTS (MAX.)
	540 WATTS (AVERAGE)
...S	200 WATTS (MAX.)
...E BLT	70 WATTS (AVERAGE)
...Z)	40 LBS.
	3000 PSI
	±1.50 INCHES
	0.6 SQ. IN.
	20 CU. IN
	7 CU. IN.
	15 CU. IN.
...LL	12 IN.
	1800 LBS.
...TED VELOCITY	1200 LBS.



VIEW LOOKING FWD IN TUG
AFT IN ORBITER
SCALE 1/20



ACTUATOR ATTACH LOCATIONS
TYPICAL - PITCH & YAW PLANES
SCALE 1/20

V7-941512

F-3

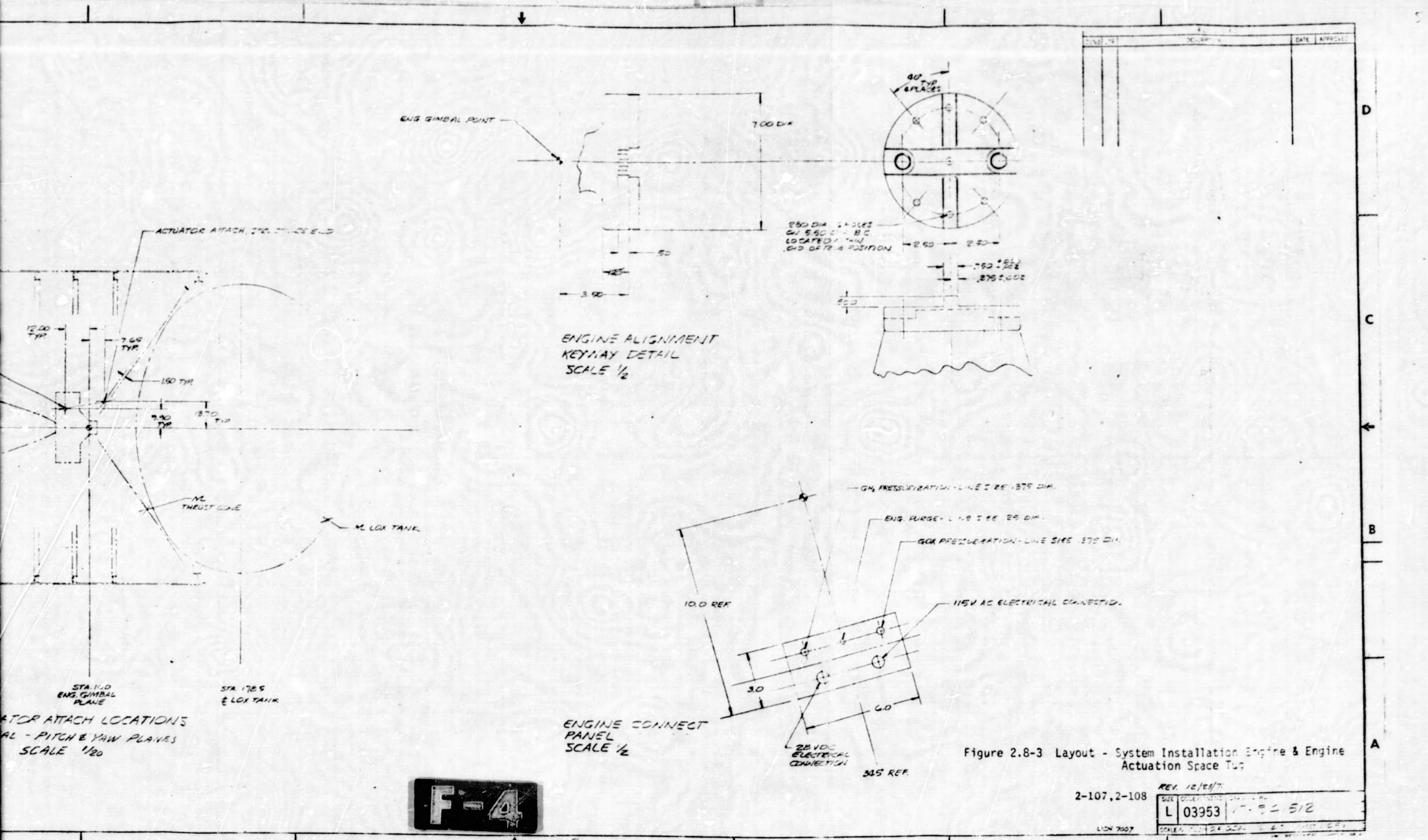
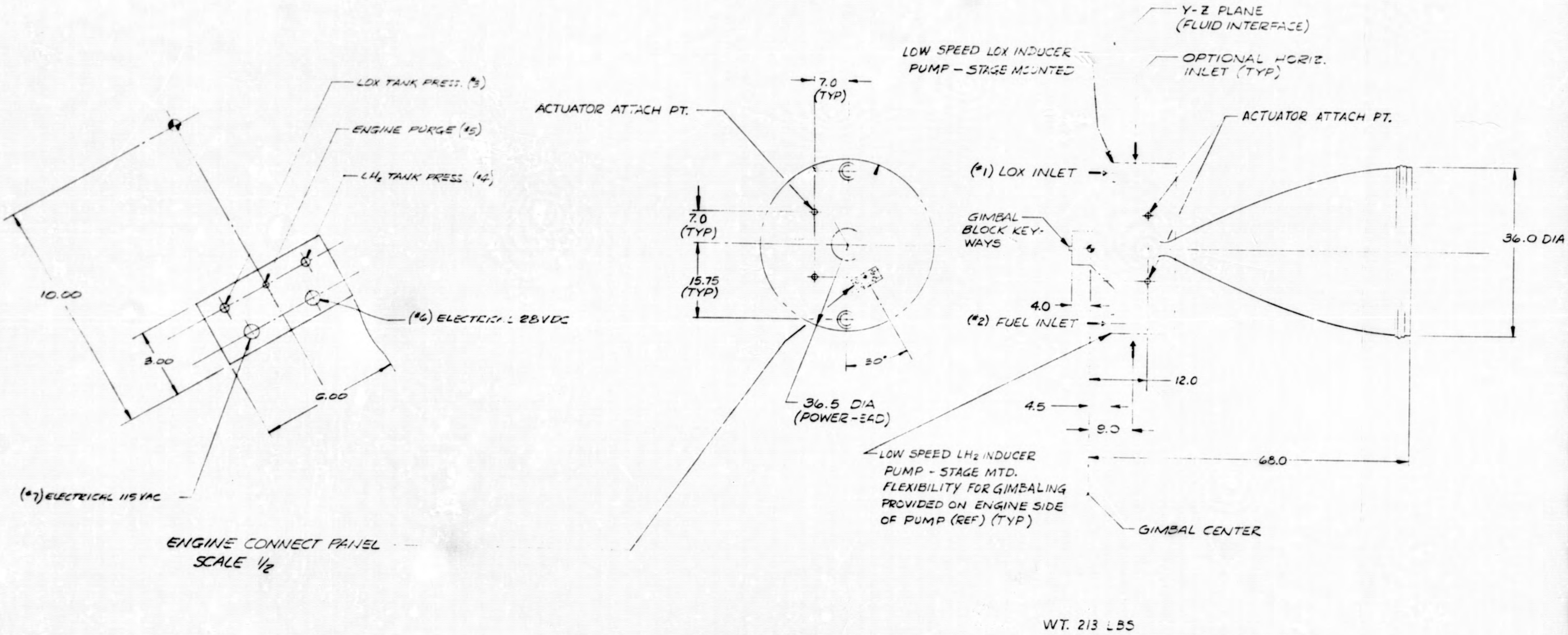


Figure 2.8-3 Layout - System Installation Engine & Engine Actuation Space Tug

2-107, 2-108

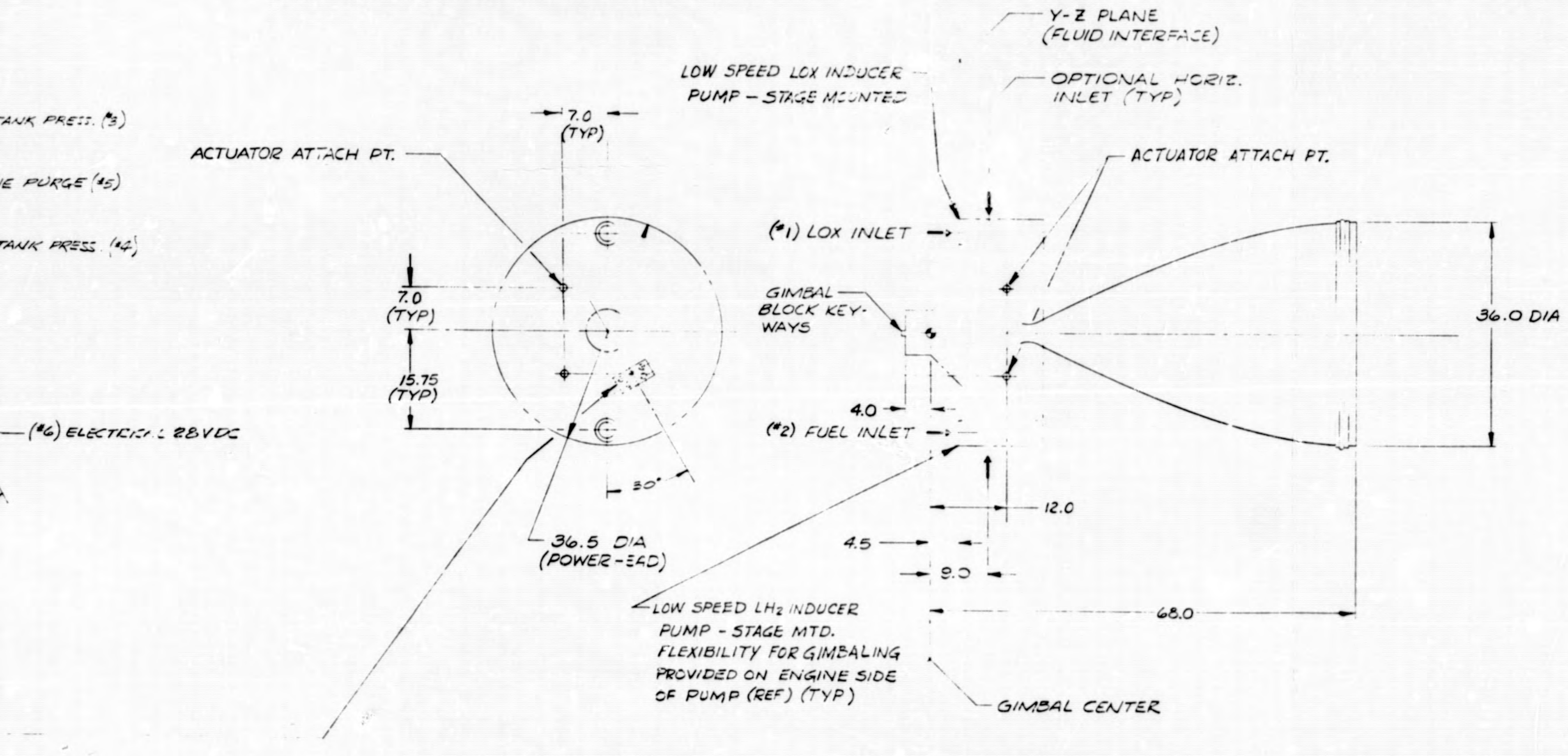


LOX TANK PRESS. (#3)
ENGINE PURGE (#5)
LH_2 TANK PRESS (#4)
10.00
(#6) ELECTRICAL 28VDC
3.00
6.00
(#7) ELECTRICAL 115VAC
ENGINE CONNECT PANEL
SCALE 1/2
ACTUATOR ATTACH PT.
7.0 (TYP)
7.0 (TYP)
15.75 (TYP)
30°
36.5 DIA (POWER-HEAD)
Y-Z PLANE (FLUID INTERFACE)
LOW SPEED LOX INDUCER PUMP - STAGE MOUNTED
OPTIONAL HORIZ. INLET (TYP)
ACTUATOR ATTACH PT.
(#1) LOX INLET
GIMBAL BLOCK KEY-WAYS
36.0 DIA
4.0
(#2) FUEL INLET
12.0
4.5
9.0
68.0
LOW SPEED LH_2 INDUCER PUMP - STAGE MTD. FLEXIBILITY FOR GIMBALING PROVIDED ON ENGINE SIDE OF PUMP (REF) (TYP)
GIMBAL CENTER
WT. 213 LBS

F-1

(REF PD-

ZONE	LTR	REVISIONS DESCRIPTION	DATE	APPROVED



— STAGE/ENGINE INTERFACE —

Nº	CONNECTION	SIZE (NOM)
1	LOX INLET	2.5 DIA
2	LH_2 INLET	2.5 DIA
3	LOX TANK PRESS	0.375 DIA
4	LH_2 TANK PRESS	0.375 DIA
5	ENGINE PURGE	0.25 DIA
6	ELECTRICAL 28 VDC	—
7	ELECTRICAL 115 VAC	—

WT. 213 LBS

F-2

Figure 2.8-4 Layout - Engine Assy, Space Tug

2-109, 2-110

(REF PD-DO-MP-11-9-71)

UCN 7007

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	VT-941511

Component	Peak	Continuous Operating	Shutdown Purge 5 sec	Coast
Electronic Control (1)	200 w	200 w	200 w	10 w
Ignition (4)	360 w	-	-	-
Solenoid Valves (4)	750 w	100 w	100 w	-
Motor Operated (3)	291 w	291 w	-	-

Helium is required for both the Tug main engine and the APS. A common stage mounted storage vessel has been selected to supply both systems. Total mission usable helium requirements, are as shown in Table 2.8-1.

The main engine helium requirement is at 750 psia and above $400^{o}R$. The storage pressure selected is 3000 psia to be common with propellant dump pressurization storage. These requirements are satisfied by a 6AL-4V Titanium pressure vessel of 2850 in^3 capacity and 21 inches in diameter with a fill pressure of 3000 psia at $520^{o}R$.

Table 2.8-1 Stage Helium Requirements

Main Engine (Reference 2.8-2)		Usable Weight
Oxidizer Pump Seal Purge Continuous during firing at 0.375 lb/hr		3.1
Oxidizer System Purge (start and shutdown) - 6 starts		0.2
APS		
Oxidizer Pump Seal Purge		0.3
	Mission total	3.6

Control

Engine control system concept analyses have been conducted by the engine candidate manufacturers with various alternates identified. The currently selected control system features for the Tug design study are summarized below.

Engine operational control is provided for each operation mode by electronic interpretation of the engine conditions by sensing the temperature and pressure parameters shown in Table 2.8-2. Four modulating control valves plus five shutoff valves are energized through the electronic engine control to operate in accordance with predetermined characteristic functions for each command signal.

Table 2.8-2. Engine Control Parameters (Pratt Whitney Data, Ref. 2.8-5)

	Baseline Control No. of Sensors	Range	Control Trim Loop	Permissive Performance	Limiting Loop	Flight	Available Precision	Required Response	Weight/Wt/Unit	Housing Wt/Unit	Weight
Main Chamber Pressure	2	0-2000	X	X		X	±1.1%FS	100 Hz	8 oz	+16 oz	48 oz
Helium System Supply Pressure	1	0-1000		X		X	±1.1%FS	100 Hz	8 oz		8 oz
Oxidizer High Pressure Turbopump Inlet Pressure	1	0-200	X		X	X	±1.0%FS	100 Hz	8 oz		8 oz
Fuel High Pressure Turbopump Inlet Pressure	1	0-100	X		X	X	±1.1%FS	100 Hz	8 oz		8 oz
Oxidizer High Pressure Turbopump Inlet Temperature	1	160-180°R	X		X	X	±0.4°R	5 Hz	12 oz		12 oz
Fuel High Pressure Turbopump Inlet Temperature	1	36-45°R	X		X	X	±0.4°R	5 Hz	12 oz		12 oz
Oxidizer High Pressure Turbopump Housing Temperature	1	160-590°R		X		X	±8.0°R	5 Hz	8 oz		8 oz
Fuel High Pressure Turbopump Housing Temperature	1	35-590°R		X		X	±8.0°R	5 Hz	8 oz		8 oz
Turbopump Turbine Discharge Temperature	1	500-2400°R			X	X	±25°R	5 Hz	10 oz		10 oz
Oxidizer High Pressure Turbopump Speed	1	0-70,000 rpm			X	X	±0.5%FS	5000 Hz	6 oz		6 oz
Fuel High Pressure Turbopump Speed	1	0-120,000 rpm			X	X	±0.5%FS	5000 Hz	6 oz		6 oz
Oxidizer High Pressure Turbopump Vibration	1	40-1000 Hz 30G			X	X	±5%FS		4 oz		4 oz
Fuel High Pressure Turbopump Vibration	1	40-2000 Hz 30G			X	X	±5%FS		4 oz		4 oz
Precombustor Oxidizer Valve Position	2		X			X			4 oz		8 oz
Main Chamber Oxidizer Valve Position	2		X			X			4 oz		8 oz
Main Fuel Oxidizer Valve Position	2		X			X			4 oz		8 oz

Space Division
North American Rockwell

An electronic engine control unit will be required to position valves for start sequence automation and safety monitoring with automatic modulation or shutdown. The control unit will sense the various fundamental engine operating parameters of chamber pressure, inlet temperatures and pressures, preburner temperature, pumps speed, pumps NPSH, pumps vibration, and valves position feedback. Sensor redundancy is provided for selected critical parameters to protect the engine against sensor failure. The electronic control unit will convert these analog input signals to digital form for use by a computerized control system. The control unit contains the memory circuit needed for engine control and checkout and performs the required arithmetic calculations.

The engine valves are positioned by the control in accordance with the stored valve schedules and the closed loop trims derived from the sensed engine parameters. Electrically actuated modulating control valves are planned to be utilized for main oxidizer control main fuel jacket bypass, preburner oxidizer, and preburner fuel bypass. Electrically actuated shutoff valves are planned to be utilized for fuel and oxidizer system purge, oxidizer prevalve, fuel prevalve and main fuel shutoff. Stage mounted pressure regulators are utilized for helium purge gas supply, fuel tank pressurization, and oxidizer tank pressurization.

The control unit provides for switching power to the preburner and main chamber ignition system. The ignition is automatically energized at the proper time during the engine start sequence, and no vehicle monitoring or control of the engine ignition is planned.

Commands from the point sensor gaging systems permit closed loop control of propellant consumption to minimize outage. A three position engine mixture ratio control valve is required with the following control settings and tolerances:

Low	5.50 ± 0.12
Nom	6.00 ± 0.12
High	6.22 ± 0.12

Although the capability of the engine is for a maximum mixture ratio of 6.5 only 6.34 is required. This reduces engine operating temperatures.

2.8.3 Operational Characteristics

Engine Capability

The engine system has capability for three operating modes: (1) pressure-fed idle; (2) pump-fed idle with pressurant gas bleed, and (3) main stage at throttled or full thrust. Impulse is derived from each of the modes but at reduced I_{sp} in the idle modes and throttled down conditions as discussed in the paragraphs that follow.

In the pressure-fed idle mode, the propellants pass from the tanks under vapor pressure through the pumps into the thrust chamber. The turbines are bypassed and the pumps are not operating, permitting operation on either liquid or vapor. Impulse thereby obtained results from the propellant vapor pressures and, considering the system hydraulic resistance, provides a thrust of approximately 35 lb. The exact thrust level can be preselected by design through use of a fixed oxidizer valve position for this idle mode. At the 35-lb thrust idle condition the preliminary values of specific impulse, mixture ratio, fuel flow, and oxidizer flow are 407 sec, 1.7, 0.031 and 0.055 lb/sec, respectively. Since this mode does not require prechill, it is used to chill the engine prior to the pumped mode.

The pumped fed idle mode may be started after chilldown. It permits bootstrap pressurization of the MPS tanks starting from zero NPSP. The engine thrust varies from 48 to 83 lb and the I_{sp} varies from 419 to 396 during this mode as tank pressure builds up to the value needed to go into main stage thrust buildup. However, the selected method for prepressurization utilizes the APS system, and pumped idle is not needed for this purpose. This capability could be used as a backup if required.

After tank pressurization, idle mode could be continued to provide vernier (low delta V) propulsion capability. However, as noted in a following section, the APS is superior for this purpose. Since pumped idle is not needed for either prepressurization or vernier propulsion, this mode is not used in the current baseline Tug design.

The normal (main stage) mode of engine operation is initiated after pump chill down and tank pressurization. The baseline engine design provides for proportional throttling between 2,000 and 10,000 lb thrust. However, for this study only full throttle of 10,000 lb thrust will be used. The nominal engine I_{sp} varies from 462 to 473.8 for the low to high thrust ranges at a mixture ratio of 6.0:1.

Start Sequence

The main engine start sequence is preceded by a settling maneuver using the APS. This assures nearly simultaneous presence of liquid at the engine inlets. Pressure-fed idle initiates the start sequence in order to chill the engine and results in useful impulse. During chill idle the tanks are pre-pressurized by the APS system. The pressurization bleed gas control valves (Figure 2.8-2) are closed during this period since no turbine exhaust heat is available for bleed gas generation. The engine is then sequenced to build up thrust to the throttle setting required. During the buildup transient, (one to two seconds after start) pressurization gas bleed from the engine to the main tanks is initiated by opening the bleed gas control valves. A study of losses involved in start and shutdown of the main engine revealed that it is more efficient to use the APS for maneuvers requiring 29,000 lb-sec impulse or less. At this crossover point the corresponding engine throttle setting is 70 percent. Except as part of the normal start chill period and thrust buildup transient, pressure fed and pumped idle modes are not used; either the APS or the main engine throttled mode is superior in performance.

Performance

The established engine ratings and performance capability for this study have been provided by MSFC (Ref. 2.8-1, 2.8-3) and are summarized in Table 2.8-3. In brief, the propulsion system utilizes a 10,000-lb thrust single engine operating at a nominal mixture ratio of 6.0 and a nozzle expansion area ratio of 400. Most of the remaining performance characteristics are dependent variables which reflect attainable and normalized values as indicated by engine study contractor analyses.

The engine operating and service life requirements reflect program level ground rules. The engine assembly weight includes an oxygen bleed vapor heat exchanger in the turbine exhaust duct. The engine assembly weight does not include the thrust vector control actuators or the vehicle thrust structure and purge gas supply provisions.

The engine's performance variation over the mixture ratio range 5.0 to 7.0 is presented in Figure 2.8-5. Engine specific impulse is maximum at 100 percent rated thrust. Performance variations over the available throttle range is presented in Figure 2.8-6. Predicted engine thrust characteristics during the normal start and shutdown transients are presented by Figure 2.8-7. The transient propellant weights for the engine start and shutdown cycles corresponding to these curves are 24 and 18 lbs, respectively (Ref. 2.8-3).

Small ΔV Considerations

The main propulsion system (MPS) furnishes all major velocity increments for Tug vehicle operations. Small velocity increments below the capability of the MPS or in an inefficient range for the MPS will be provided by the auxiliary propulsion systems aft firing thrusters. To establish the efficiency crossover point for selection of the proper system for any given ΔV requirement, an analysis of main engine start and shutdown losses is presented in the following paragraphs. The MPS losses result from the idle mode chill cycle, startup and shutdown performance degradations, and the loss of residual propellant downstream of the valves at shutdown.

Since main engine usable propellant is defined as providing an Isp of 470 sec, any impulse of the main engine at less than 470 sec is entered in the weight statement as a loss which is the difference between the propellant used and the amount required to obtain the same total impulse at an I_{sp} of 470 sec.

All of these losses represent loaded propellant and along with the amount credited at 470 I_{sp} in the main engine usable entry, total the actual amount consumed. The additional usable propellant required to provide momentum to the propellant associated with the losses for each maneuver was determined from the flight performance mission timeline computer program.

Start Loss. Engine chill requirements are estimated to be 3.3 lbs on the oxidizer side and 3.7 lbs on the fuel side (Ref. 2.8-3). Using the idle mode data of Table 2.8-4, the fuel side is controlling and requires a duration

Table 2.8-3. Space Tug Main Engine Ground Rules

Number of Main Engines	1
Propellants	Liquid Oxygen/Liquid Hydrogen
Vacuum Thrust, pounds	10,000
Nominal Engine Mixture Ratio	6.0:1
Engine Mixture Ratio Range	5.5 - 6.5
Engine Chamber Pressure, Psia	2020
Engine Length/Diameter, Inches	68/36
Vacuum Thrust Throttling Capability	5.0:1
Nozzle Configuration	Bell
Nozzle Expansion Ratio	400:1
Turbine Drive Cycle	Staged Combustion
Min. Guaranteed Vacuum Specific Impulse, Seconds	470
Nominal Vacuum Specific Impulse, Seconds	473.8
Engine System Weight, Pounds	213
Number of Vacuum Starts	160
Service Life Between Overhauls (Reusable Mode), thermal cycles	300
Service Life Between Overhauls (Reusable Mode), Hours	20
Gimbal Angle (Square Pattern), Degrees	±7
Gimbal Acceleration, Radians/$(\text{second})^2$	20
Minimum Natural Frequency of Gimbal System, Hertz	10
Fuel Pump NPSH, feet of hydrogen	15
Oxidizer Pump NPSH, feet of oxygen	2
Maximum Single Run Duration, Seconds	1400
Maximum Storage Time in Orbit (Dry), Weeks	1
Maximum Time Between Firings (Coast Time), Days	3
Minimum Time Between Firings (Coast Time), Minutes	10
Service-Free Engine Run Time, Hours	1
Service-Free Engine Firing Cycles	10

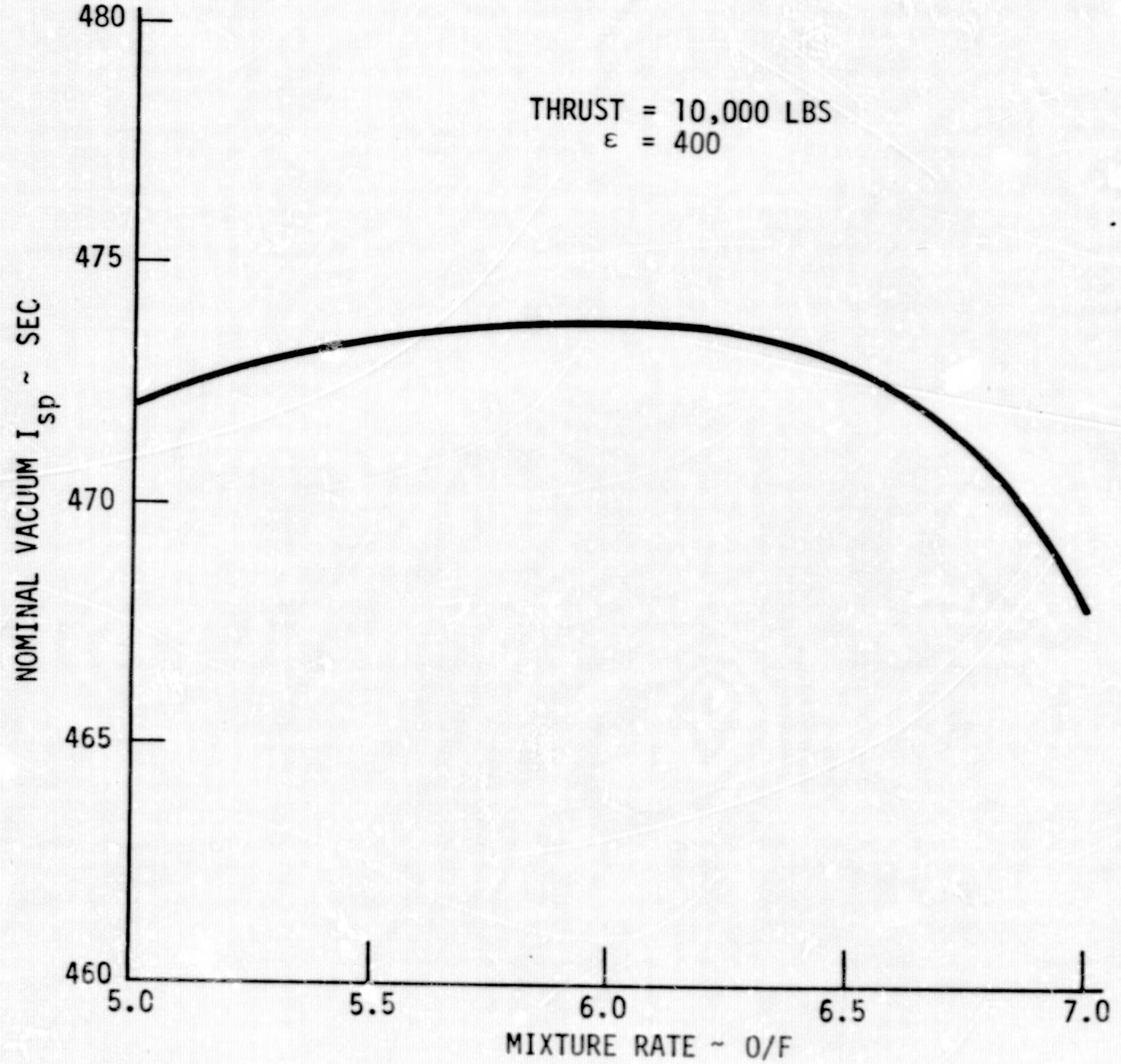


Figure 2.8-5 Mixture Ratio vs Nominal Vacuum I_{sp} for Engine Design at MR = 6.0

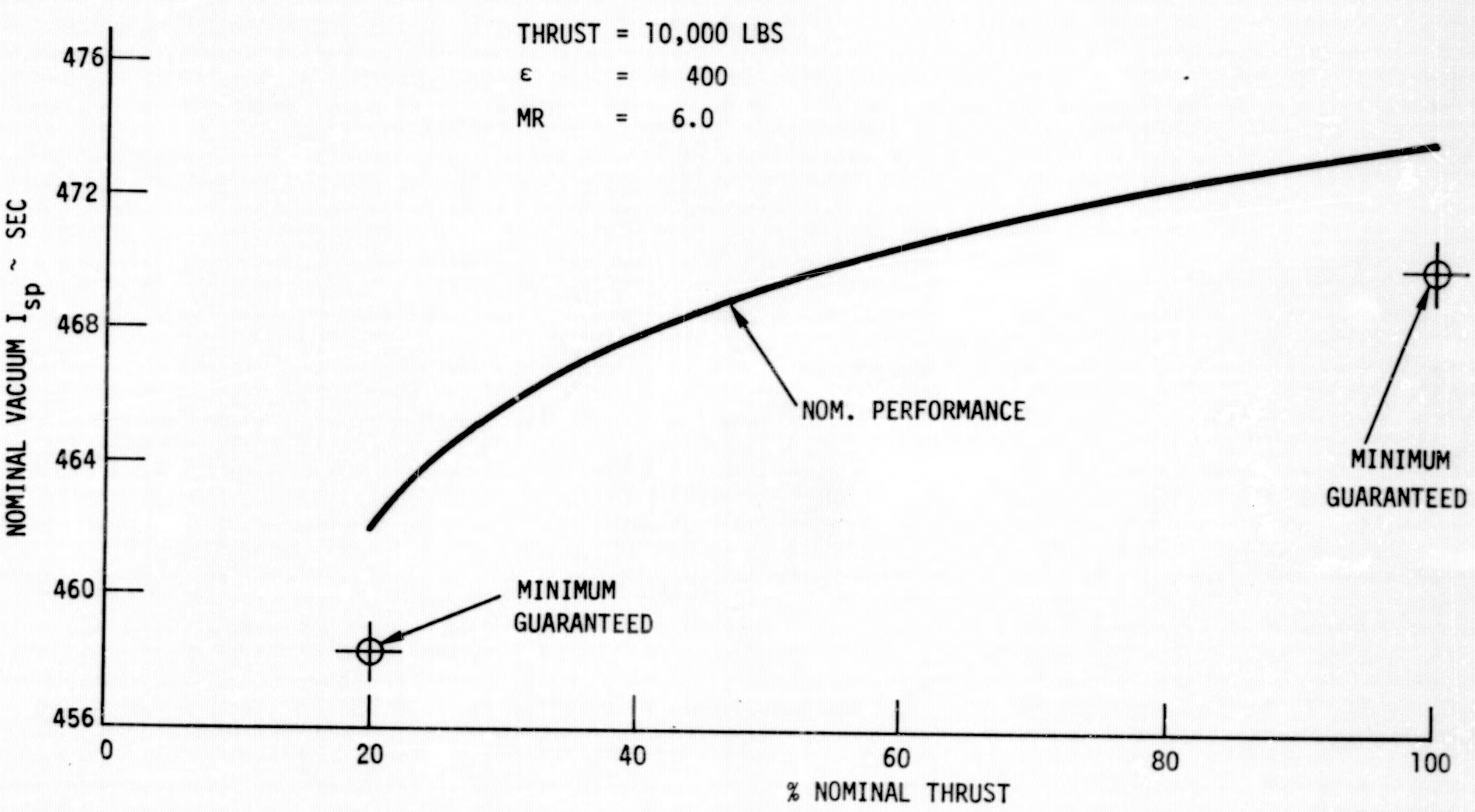


Figure 2.8-6 Throttled Engine Performance

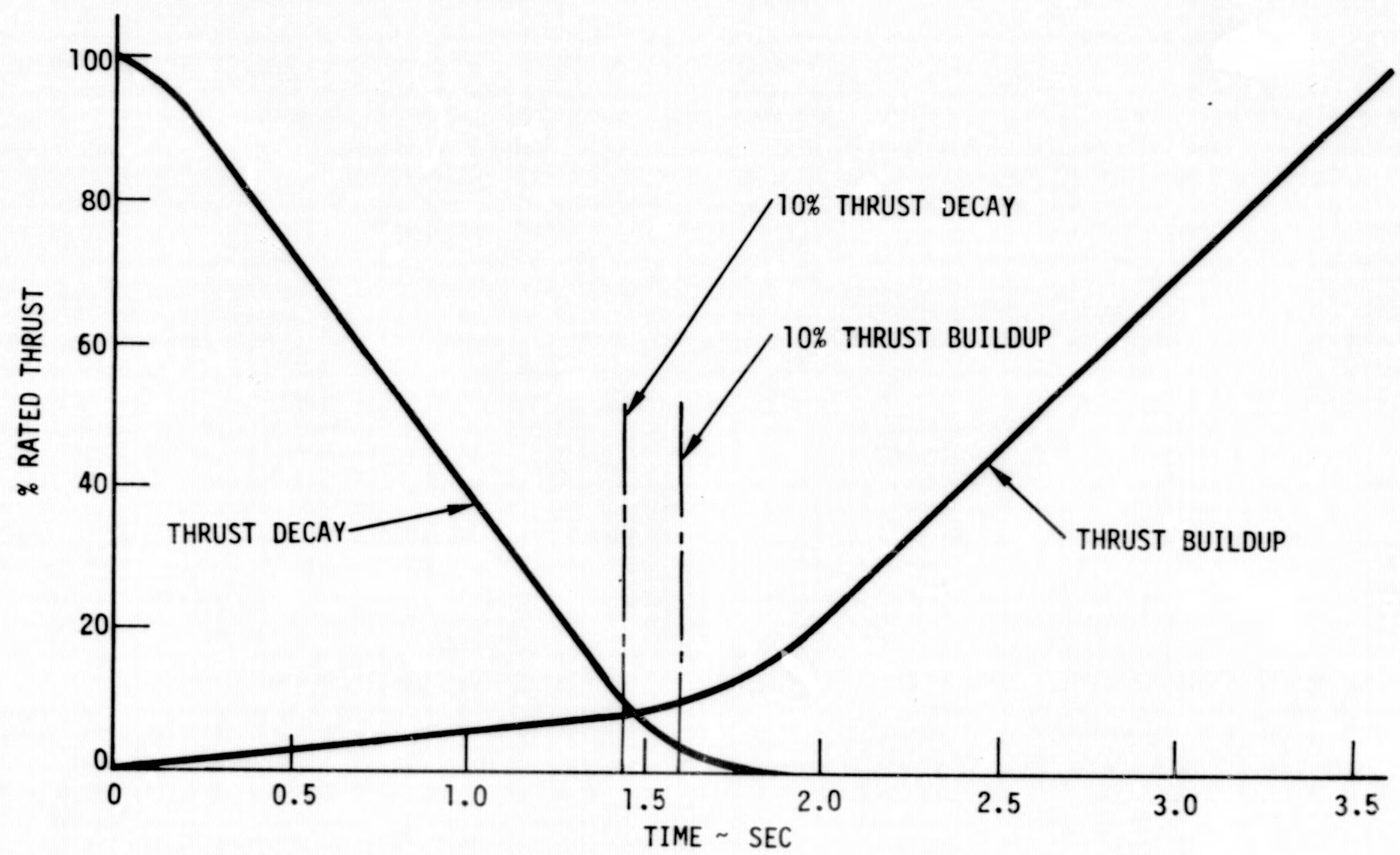


Figure 2.8-7 Thrust Transients

Table 2.8-4. Main Engine Performance

Main Engine Operating Mode	Thrust lbs	I_s sec	MR	NPSH (ft)	
				Ox	Fuel
Chill Idle	35	407	1.7	0	0
*Press Fed Idle After Chill	42	442	3.0	0	0
*Pumped Idle	83	396	1.4	2	15
Full Thrust	10,000	470	6.0	2	15

*Engine capability but not used for this study

of 120 sec. Thus, a total impulse of 4200 lb-sec results from chilldown. The net loss is 1.4 lbs per start which is the difference between a gain (due to the low mixture ratio at idle) of 1.1 lbs of oxidizer and a loss of 2.5 lbs of fuel.

The start transient impulse derived from Figure 2.8-7 is 11,000 lb-sec with a total consumption of 24 lb. The net thrust buildup loss is 1.0 lb and the duration of the start transient is 3.6 sec. The total of all losses is then 2.4 lb per start.

Shutdown Loss. During thrust decay, the engine delivers 8300 lb-sec over a period of 1.8 sec after cutoff and consumes 18 lb of propellant with a loss of 0.5 lb. The shutdown loss is the transient performance loss plus the wet minus dry weight of the engine plus the contents of the feed lines below the prevalves. The engine weight difference is estimated from informal data from Rocketdyne and Aerojet. The line is assumed to be a 24-inch long 2.5 dia. section.

The losses are as shown in Table 2.8-5.

Table 2.8-5. Shutdown Losses

	Ox	Fuel	Total
Thrust Decay	0.4	0.1	0.5
Engine Trapped	1.5	3.0	4.5
Suction Line	4.8	0.3	5.1
Total	6.7	3.4	10.0

APS Performance. The APS performance is at a specific impulse of 380 sec after allowing for conditioning unit consumption. To compare this system with the MPS, its loss for 10,000 lb -sec of impulse compared to performance at 470 sec is 5.04 lbs. No other losses are now estimated although there may be some associated with conditioning unit start.

Throttled Main Engine Performance. Engine performance during steady-state throttled operation is as defined in Figure 2.8-6 and is shown for reference as only full thrust will be used.

Main Engine Minimum Total Impulse. The minimum duration capability of an engine firing at all pumped thrust levels is approximately 5.4 sec. The integrated minimum impulse (including buildup and decay impulse) beneath the transient is assumed to be equivalent to a square wave of 3 sec duration at the peak thrust (throttle setting). Figure 2.8-8 presents the result of this assumption. The effective square wave I_{sp} is 100% of that for a steady-state firing but the start and shutdown losses at full thrust apply.

Settling Requirement. The settling impulse requirement, in terms of free fall durations (τ) is

$$I = \tau \sqrt{\frac{2WFh}{g}}$$

where:

I = total impulse - lb-sec

W = Tug weight - lbs

h = fall height - ft

F = thrust - lbs

and the burn time is

$$tb = I/F$$

The worst case is for the hydrogen tank (h = 17 ft).

$$I = \tau \sqrt{(37.6W - .516W^2) \frac{F}{g}}$$

The maximum is reached when the tank is 54% full. The corresponding impulse and burn time for two free falls and the APS minimum thrust of 140 lbs are 3500 lb sec and 25 sec. The propellant loss compared to the impulse at 470 sec is 1.8 lbs.

Although the chill idle thrust is sufficient to settle propellants the recommended normal mode is to precede the start with APS settling. This

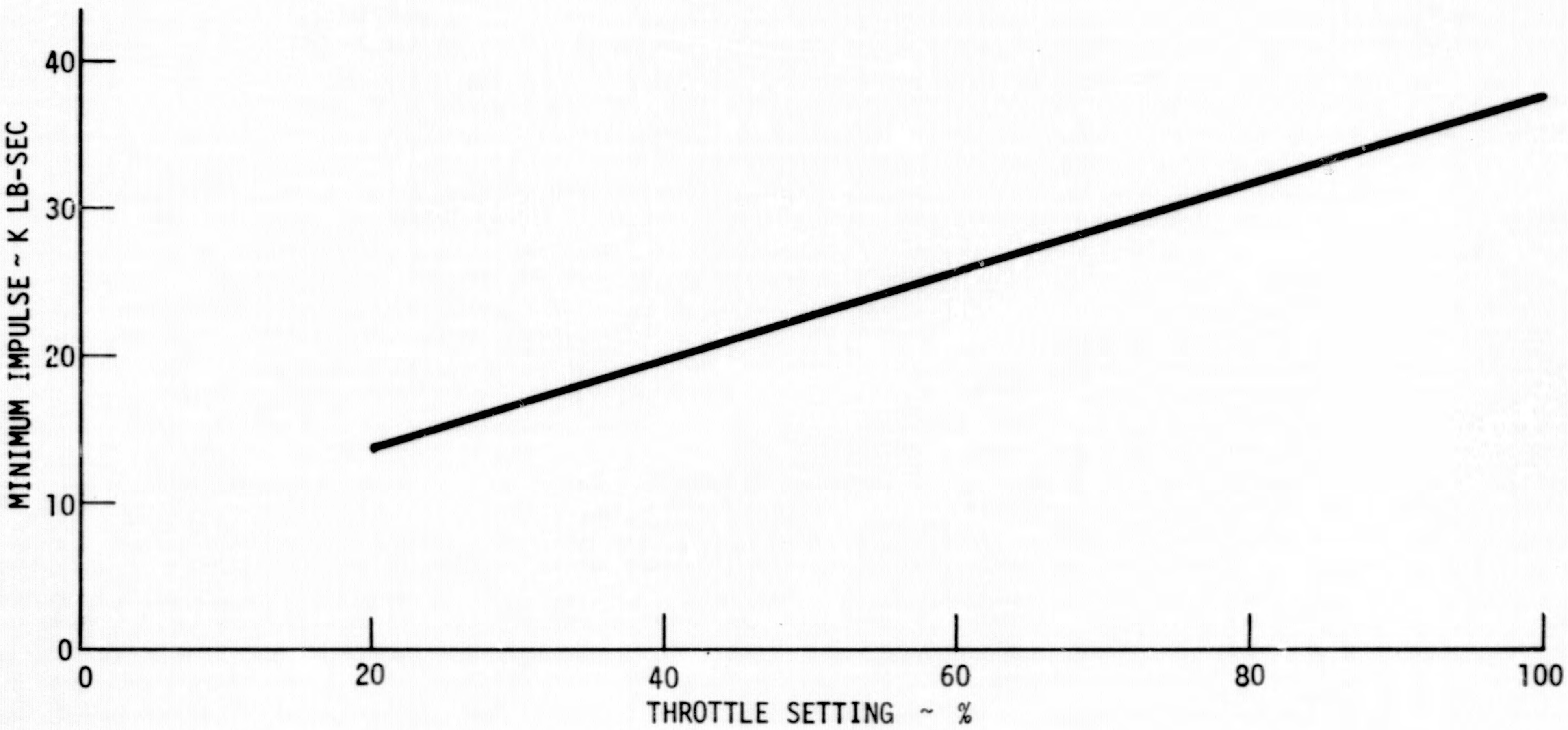


Figure 2.8-8 Main Engine Minimum Impulse Capability

appears to be a prudent choice from a slosh control viewpoint and also assures that each start avoids entrance of excess vapor and an attendant prolonged idle mode. The first start requires idle mode for only 2 sec. to clear the line of vapor; and with prior APS settling, the turbopump can be safely started after chill (120 sec). Figure 2.8-9 is a time-line diagram of the firing sequence showing the overlap of APS and MPS operation.

Summary. Figure 2.8-10 diagrams the performance of the several modes available. As may be seen the APS is most efficient for all impulses less than 29,000 lb-sec. At this point its low I_{sp} crosses over the losses of starting the main engine, and the minimum capability of the main engine corresponds to a throttle setting of 70%. At 38,000 lb-sec the maximum throttle setting is at rated thrust. Due to its low specific impulse, pumped idle is not competitive. In general, the highest throttle setting permitted by the magnitude of the maneuver impulse is most efficient.

Figure 2.8-11 displays the maneuver times required for each mode. It reveals that the most efficient mode for small impulse maneuvers (APS) is also the shortest until the MPS/APS efficiency crossover is reached where there is a sacrifice of 42 sec (including settling) in using the main engine. This time differential vanishes at a maneuver impulse of 41,000 lb-sec. If the most rapid maneuver is desired, as might be the case in terminal rendezvous, the APS should be used with a slight loss in propulsion efficiency.

To apply the efficiency crossover criteria, Figure 2.8-12 indicates the applicable mode as a function of stage weight and maneuver velocity.

The preceding criteria were used in establishing the timeline for the reference round-trip payload mission. For this mission, the required maneuvers were all in the range below 29,000 lb-sec or above 40,000 lb-sec. In consequence, the engine's capability for throttled operation was not needed. In accordance with the criteria, the low impulse maneuvers of the timeline are all performed by the APS and the higher impulse maneuvers are performed by the MPS at full throttle setting.

2.8.4 Design Alternatives

APS Thruster Impingement Effects

The physical arrangement of the APS thrusters on the TUG vehicle results in minor exhaust gas impingement on the main engine nozzle from any one of the aft firing engines. Recognition of this condition has led to an analysis to determine the magnitude of the heat flux at the nozzle, the estimated resulting maximum nozzle temperature, and the possible detrimental effects. This study was undertaken to determine if any engine design changes would be required. It has been concluded, based on present knowledge, that no special design nor operational impact will result.

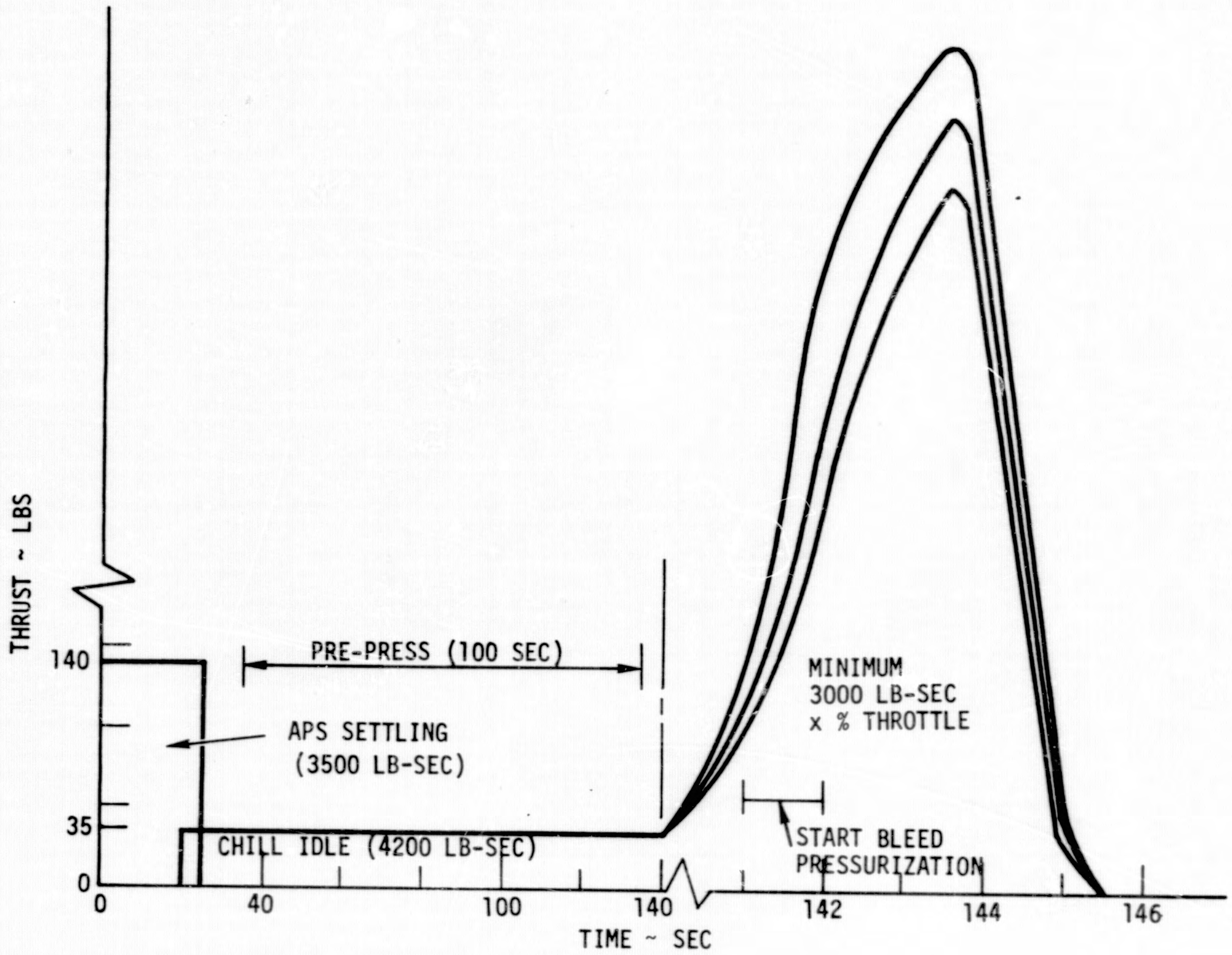


Figure 2.8-9 Main Engine Firing Sequence

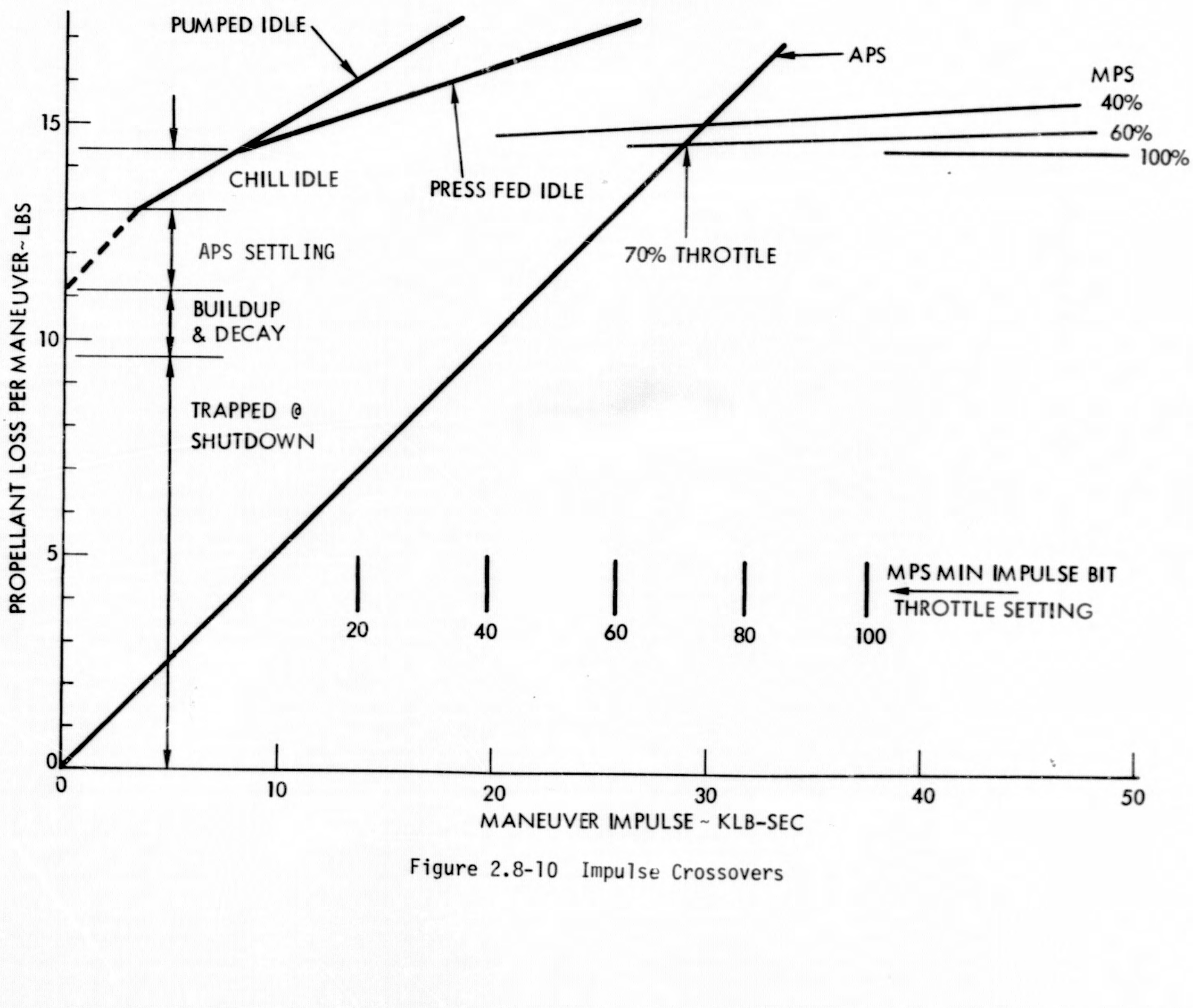


Figure 2.8-10 Impulse Crossovers

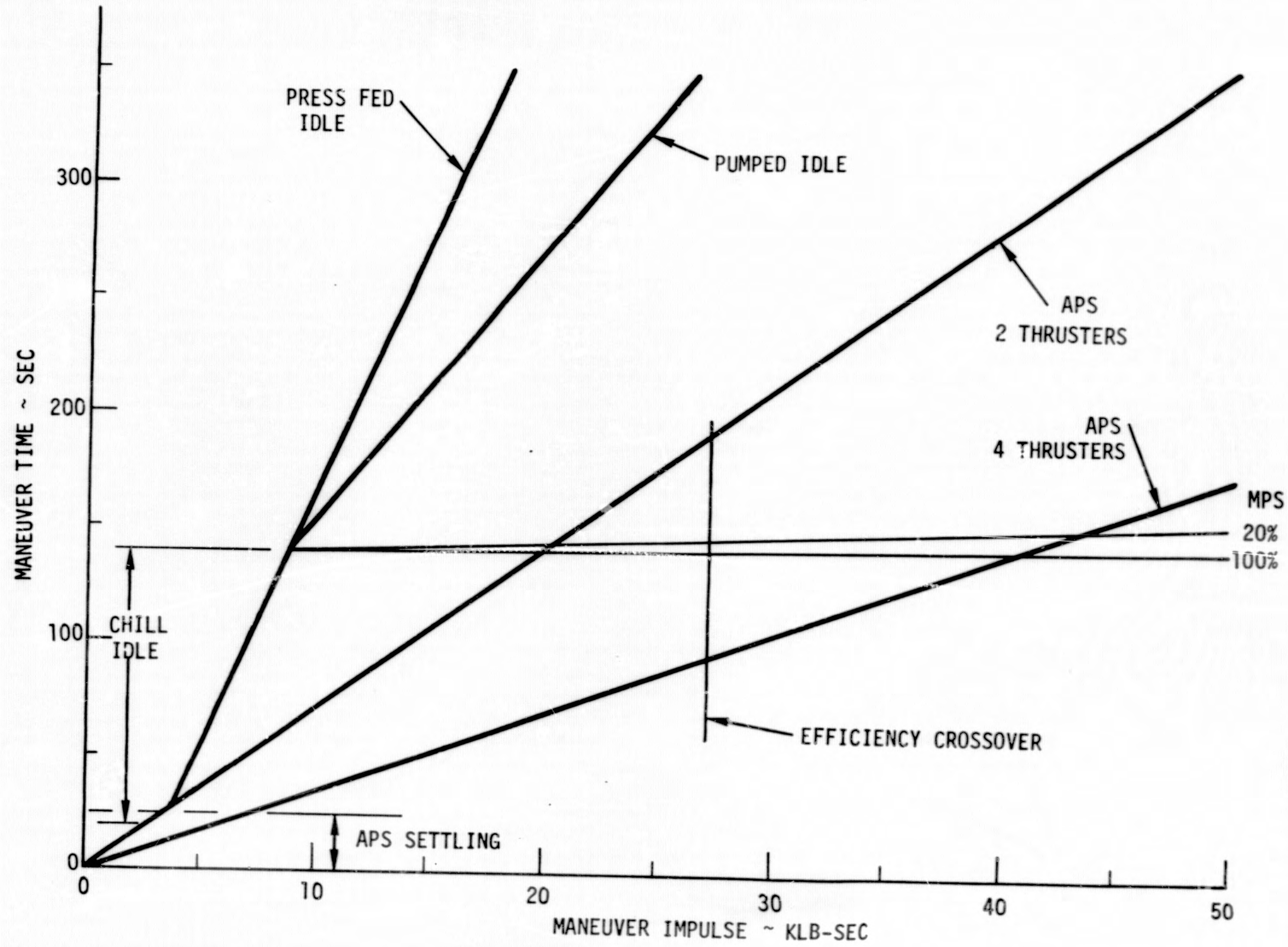


Figure 2.8-11 Maneuver Time

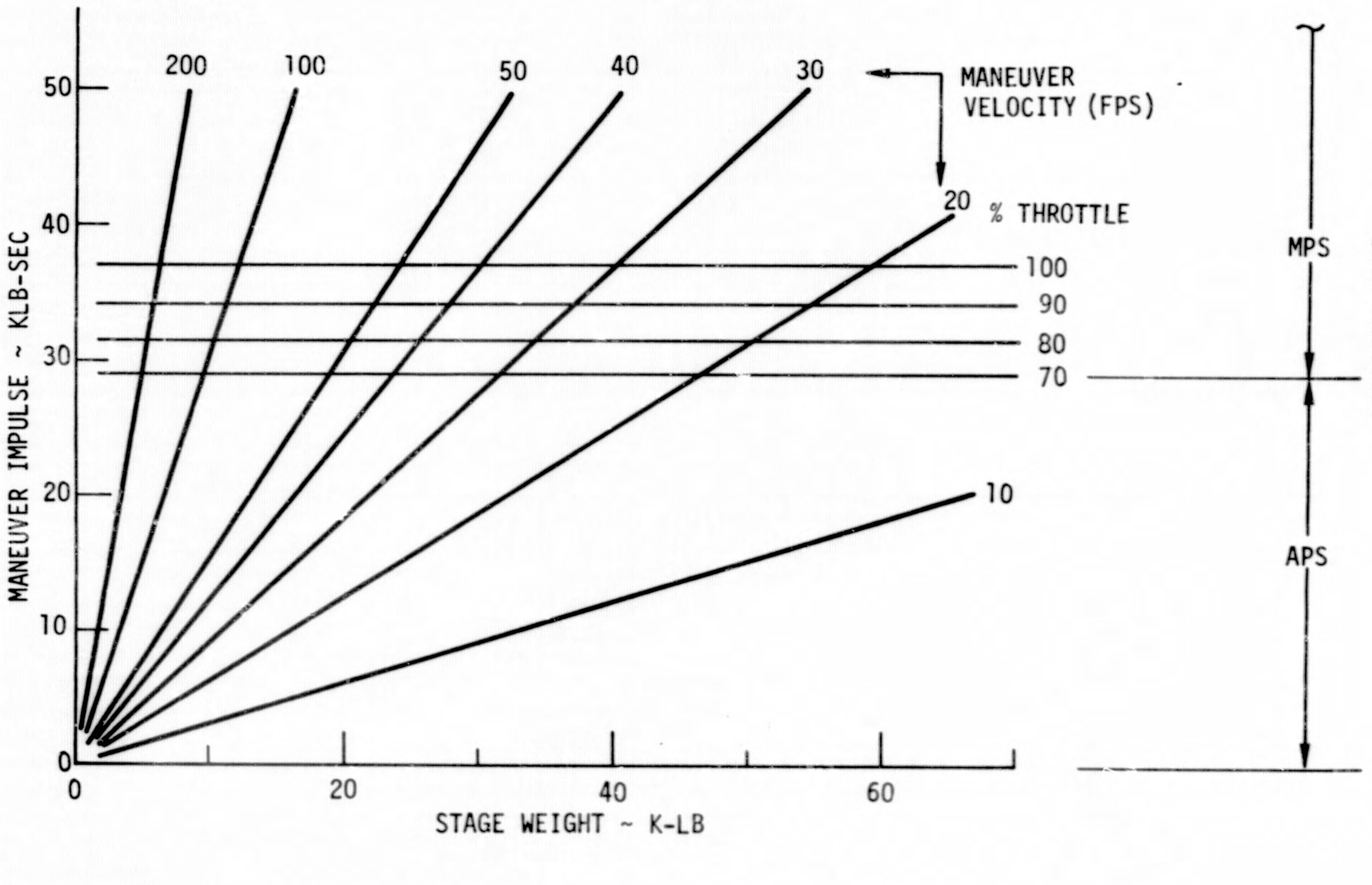


Figure 2.8-12 Crossover Velocity

The calculated main engine nozzle heat flux from a single aft firing APS thruster is shown by Figure 2.8-13. As indicated, the worst case maximum heat flux at the nozzle is 1.06 Btu/sec ft^2. Resulting predicted nozzle temperature with a maximum of 735 F are shown by Figure 2.8-14. The temperature effects analysis was addressed to the following areas of concern with the tentative conclusions indicated.

The possibility exists that the nozzle heating may cause reduced strength or introduce distortion and/or thermal stresses. The engine candidate manufacturers are proposing to use inconel or a similar steel material with dump cooling through axial channels welded to a removable nozzle extension. Such a design and material is predicted to tolerate temperatures to 750 F and any resulting circumferential differential heating. The nozzle is structurally adequate to accomodate relatively severe load reversals and vibration conditions which occur during firing, and therefore, the nozzle is designed to accomodate the stresses associated with this heating condition.

The APS heating of the engine nozzle was reviewed in connection with possible affect on the chill down cycle. Since there is no purpose in chilling the nozzle itself and the heat transfer resistance from the nozzle to the pump assembly is high, it has been determined that any affect on engine chill down is insignificant.

The desirability for a special nozzle coating because of the APS engine gas impingement was considered. Nozzle surface treatment will normally be required to resist corrosion and to attain satisfactory optical properties to assume reasonable space equilibrium temperatures. Use of grey oxidized surface attained by heating the nozzle in air is tentatively planned. Such a surface provides an emissivity of about 0.8 which is the value used to calculate the data of Figure 2.8-14. It is concluded that no special coating is needed.

Retractable Nozzle Extension

Consideration has been given to the use of a two-position, radiation cooled nozzle extension for the engine for the purpose of reducing the overall length of the Tug vehicle. Parametric data provided by Ref. 2.8-6 was utilized in connection with inputs from P&WA and Aerojet.

The established physical changes of 30 pounds increased weight and an engine stowed length of 37 inches were reviewed in connection with the vehicle design parameters. This design feature provides an improvement of approximately 45 inches in the Tug vehicle length when stowed in the orbiter cargo bay but with the introduction of the problem areas noted below.

It is apparent that engine reliability will be degraded by the use of a mechanized nozzle extension, the exact degree is dependent upon the design

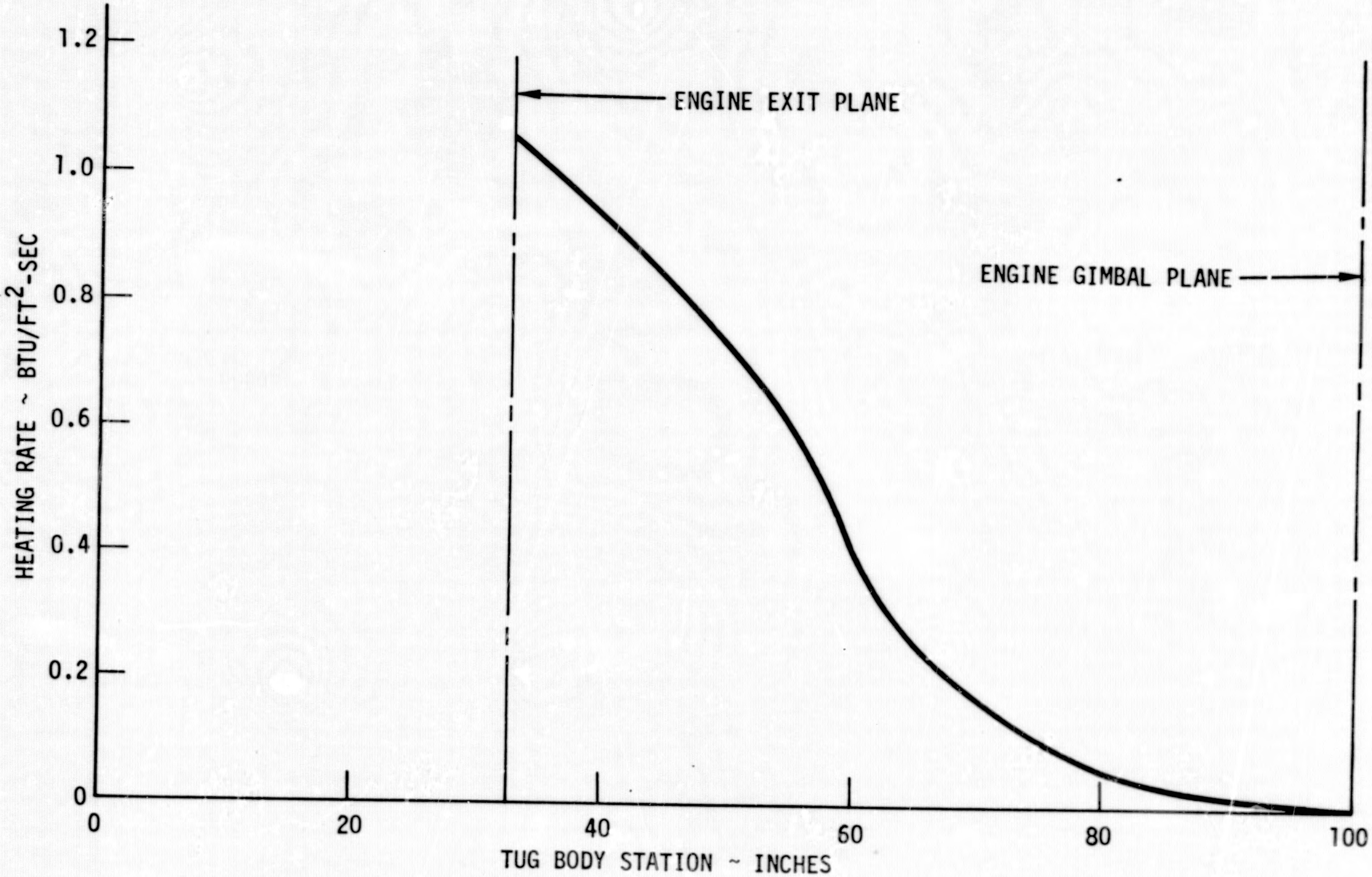


Figure 2.8-13 APS Impingement Heating Rates on Main Engine

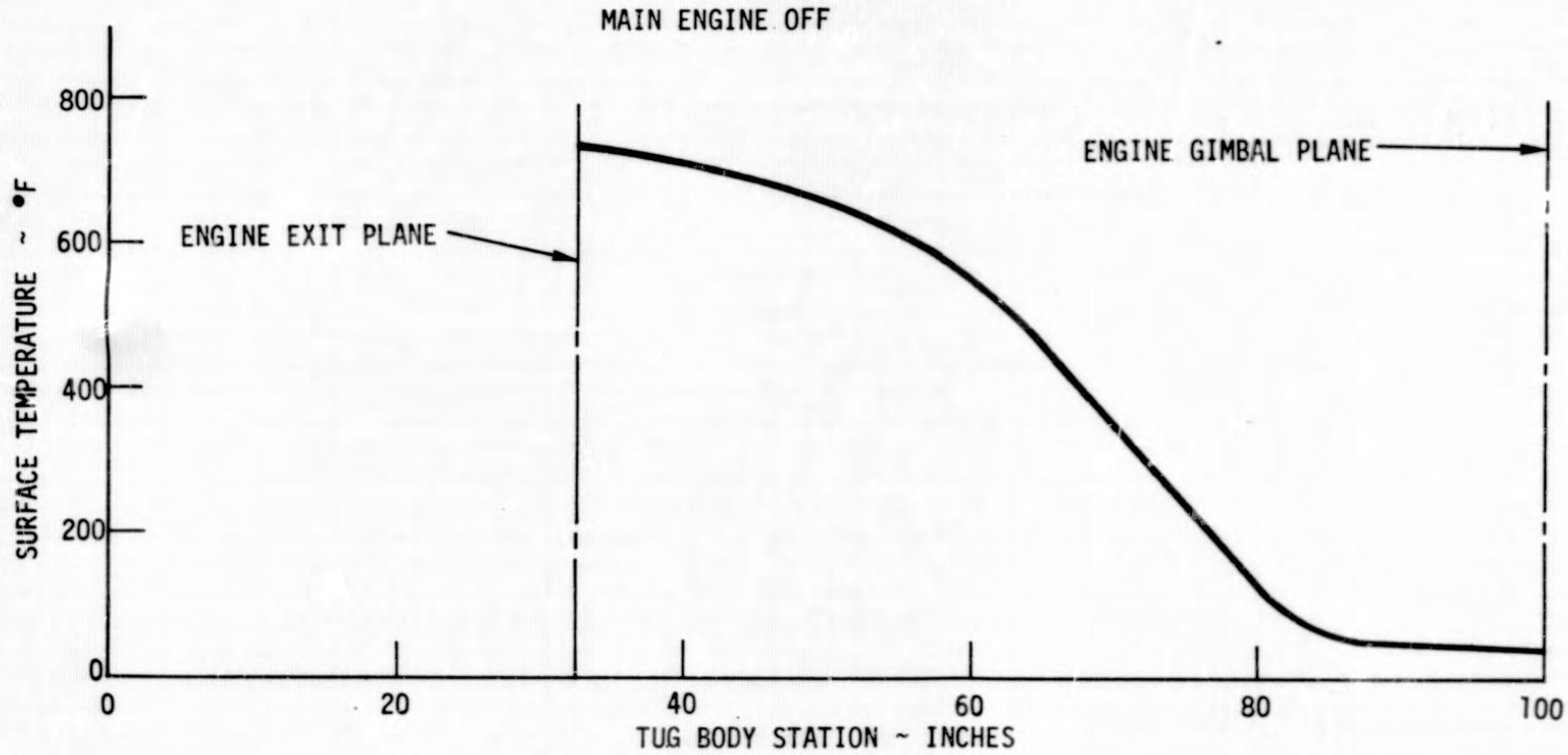


Figure 2.8-14 Main Engine Steady-State Nozzle Wall Temperature Due to APS Impingement

implementation and qualification performance requirements. At least three failure modes would be involved with the following noted consequences:

a. Failure to extend - inability to use engine or perform mission. Failure detection by position sensors.

b. Extension incomplete; lack of seal at nozzle joint - inability to use engine or perform mission; probable burning or erosion of nozzle parts and adjacent structure. Possible detachment of fragments. Failure detection by GN&C on first firing due to excessive and random thrust misalignment. Possible detection by position sensor.

c. Failure to retract - inability to stow tug in cargo bay and loss of tug vehicle. Detection by position sensor.

Use of an extendable nozzle extension dictates use of either radiation heat transfer for nozzle cooling with an associated higher nozzle temperature, or a dump cooled extension with a coolant manifold disconnect seal. The engine candidate manufacturers have investigated available materials for radiation cooled nozzle without immediate success. Columbium meets the temperature strength requirements but is subject to stress corrosion failures in the presence of hydrogen gas. Development of a satisfactory advanced material or coating would be anticipated during the time period of 1976 but an element of program risk is involved.

It has been concluded that at present the advantages of reduced Tug vehicles stowed length does not warrant the introduction of lower engine reliability and possible development problems associated with an extendable nozzle.

Higher Thrust Rating

The alternative of selecting an engine thrust rating in the range of 12,000 to 14,000 lbs instead of 10,000 lbs is worth further consideration. It would simplify engine design and development and allow for stage growth. At 10,000 lbs the engine life requirement for 20 missions is 14 hours. This difficult requirement would also be alleviated by higher thrust. No decrease in stage performance (see thrust optimization in flight performance section of this report) would result but the larger nozzle would add six to eight inches to stage length.

Redundant Engines

The use of two engines instead of one could add reliability with some penalty in stage performance. Evaluation of this advantage depends on a thorough engine installation design study for both concepts and a failure mode effects analysis to determine the extent of redundancy that is achievable.

Sea Level Test Capability

For convenience and economy in engine or stage static firing tests, a removable nozzle to permit sea level firing is preferred. This would alleviate the need for a ground test diffuser which may be difficult to apply at this high expansion ratio. An engine design incorporating a removable dump cooled extension may be best if performance degradation is not severe. A design using a removable, regeneratively cooled extension with a separate cooling circuit could also be used but a ground test heater to replace this element of the cycle or some other test provision would be needed. In this case, full flight operational simulation of the engine would not be obtained.

2.9 THRUST VECTOR CONTROL SUBSYSTEM

2.9.1 Introduction

Studies have been conducted to select the preferred thrust vector control system concept. Electric motor driven hydraulic, engine driven hydraulic, electro-mechanical and pneumatic designs have been reviewed as presented in the paragraphs that follow. The thrust vector control system selected for TUG application is a conventional hydraulic system utilizing an electric motor-driven pump as the source of power. This concept was favored over the others on the basis of weight, reliability, and minimum development requirements.

2.9.2 Selected Subsystem

Based on the studies to date, the preferred concept for the thrust vector control system is a conventional electric-powered hydraulic system. The system would employ two servo-controlled hydraulic actuators with mechanical feedback to provide ± 7 degrees gimbal capability in the pitch and yaw planes. As envisioned, all supporting system components will be contained within the actuators. For example, the electric motor and hydraulic pump would be housed in one actuator and the reservoir, accumulator, relief valve, etc., would be mounted in the other actuator (see Figure 2.9-1). Thus, the only plumbing external to the actuators would be a supply line and a return line interconnecting the two actuator assemblies. Installation of the actuators is shown in Figure 2.8-3.

Requirements for the proposed system are:

Gimbal capability	± 7 deg.
Gimbal velocity at design load	8 deg/sec
Gimbal acceleration	12 deg/sec^2
Electrical Power for Gimballing (400 Hz)	900 watts (max) 540 watts (Ave)
Electrical Power for Heating	200 watts (max) 70 watts (Ave)
Total weight	40 lbs
Operating pressure	3000 psi
Actuator stroke	± 1.5 in.
*Actuator piston area	0.6 sq. in.

*Actuator load requirements for Tug are as yet unknown. The values shown are commensurate with S-IV/RL-10-actuator parameters.

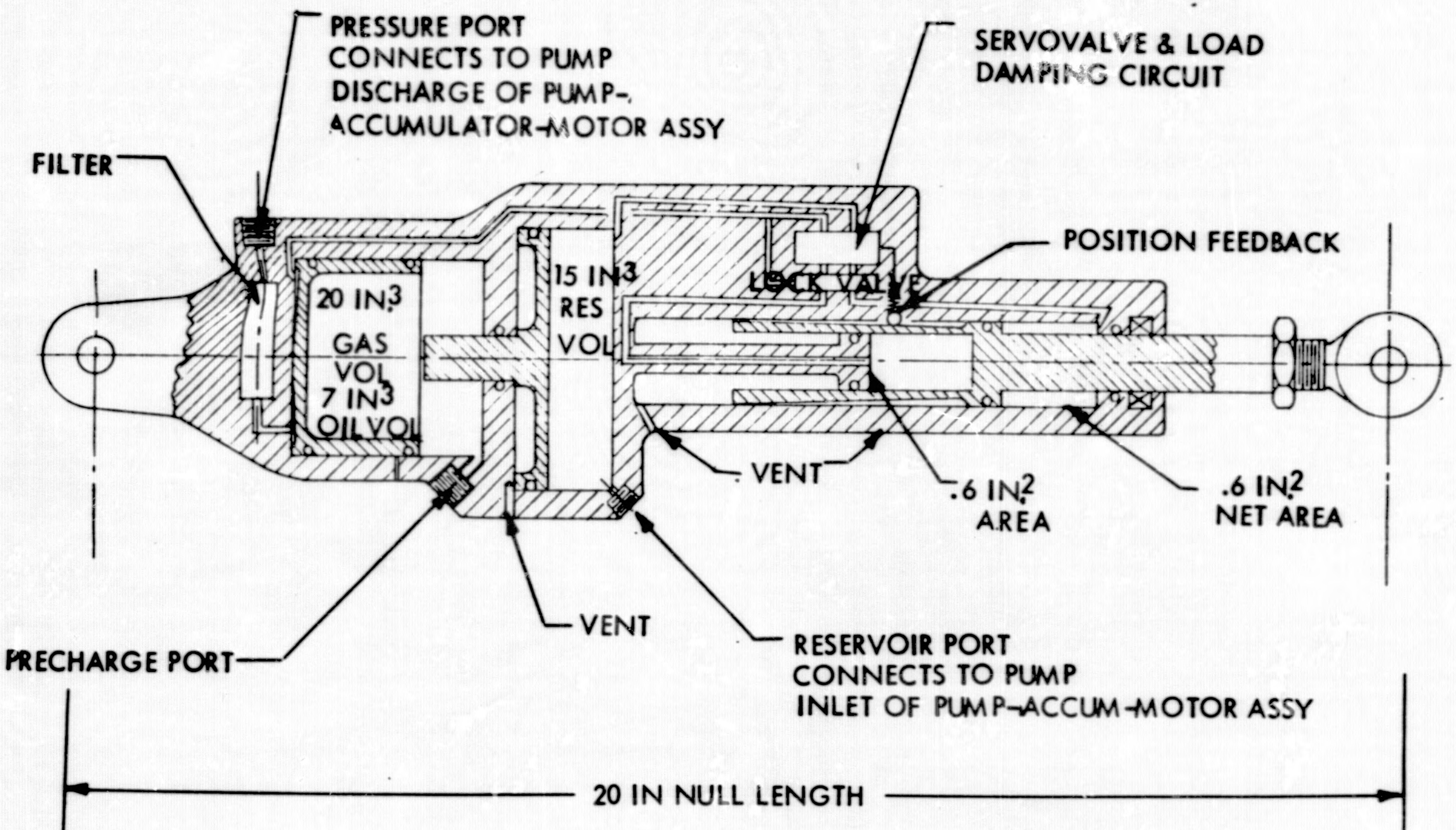


Figure 2.9-1 TVC Accumulator-Reservoir-Actuator-Manifold Assembly

Accumulator gas vol.	20 cu. in.
Accumulator oil vol.	7 cu. in.
Reservoir vol.	15 cu. in.
Actuator moment arm at Null	12 in.
*Actuator stall force output	1800 lbs.
*Actuator force output at rated velocity	1200 lbs.
Actuator pin-to-pin length	20 in.

2.9.3 Alternate Systems Considered

Engine-Driven Hydraulic

This concept is similar to several past booster vehicle design including this contractor's S-II stage and assumes that a shaft output from one of the propellant pumps will be used to power a hydraulic pump. On the surface, this approach locks appealing in that the power is virtually "free;" however, it has been tentatively rejected based on the following rationale.

The engine will be operated in an idle-mode wherein the propellant pumps, and hence the hydraulic pump will be non-operative. Therefore, if thrust vector control is to be employed during idle mode operation, the power must come from either an accumulator or from the electric-driven pump. For the anticipated idle-mode duration (one minute or more), the required accumulator size and weight would be objectionably large; therefore, the electric-power concept appears more favorable. Also, even if steering was accomplished by the reaction control system during idle mode an electric motor-driven pump (of smaller size) would be required anyway to circulate fluid for thermal conditioning between engine burn periods.

Electric Ball Screw Actuator

The Apollo SPS actuators are of this type and have performance characteristics that would probably be acceptable for Tug usage. The major disadvantages in order of importance are: (1) weight (almost twice that of the selected system) (2) predicted reliability and (3) cost. It should also be noted that the electrical power requirement during gimballing is approximately the same for electric actuators as for the electric-driven hydraulic; however, the electric actuators would probably require less electrical input for thermal conditioning than the hydraulic type.

*Actuator load requirements for Tug are as yet unknown. The values shown are commensurate with S-IV/RL-10-actuator parameters.

Pneumatic

Pneumatic power could be used directly to drive the actuators (turbine or position displacement motor driving ball screw actuator) or indirectly by charging hydraulic accumulations which in turn would provide hydraulic power for conventional hydraulic actuators. The source of pneumatic power during burn could be the high pressure hydrogen developed by the engine for tank pressurization. During idle mode, the hydrogen could be obtained from the high pressure hydrogen storage employed in the APS.

The pneumatic powered screw jack actuators are extremely attractive from the weight standpoint and their compatibility with the environment (would require minimal heating) but were rejected because of the development requirements.

The pneumatic/hydraulic concept was rejected because of the unacceptable low temperature of the hydrogen available from the APS storage. The beneficial aspects of this concept (low weight and minimal electric power requirement) were considered to be outweighed by the additional heating requirements imposed on the APS gas generation and storage system.

2.10 AUXILIARY PROPULSION SYSTEM (APS)

2.10.1 Function

The Auxiliary Propulsion System (APS) has several functions. They are as follows:

(1) Attitude Control

(2) Roll Control during Main Engine Burns

(3) Vehicle Orientation

(4) Attitude Hold

(5) Near Rendezvous Maneuvering

(6) ΔV for Flight Trajectory (Impulse 28,000 LB-SEC Max.)

(7) ΔV for Propellant Settling

(8) Reactants for Fuel Cell operation

(9) Supply gases (GOX and GH_2) for pressurization of Main Propellant Tanks Prior to Main Engine Main Stage Operation

(10) Thermodynamic (T/D) Vent Cooling Subsystem

These functions are performed by the APS in orbit after removal of the TUG from the Orbiter Cargo Bay. To perform the functions, the propellant

conditioning systems of the APS convert LH_2 and LOX from the auxiliary tanks in the Tug main propellant tanks into GH_2 and GOX, and supply them to the using systems.

2.10.2 Description and Operation

a. Design Concept

As specified in the Tug point design study plan, a GOX/GH_2 reaction control system was used as the basis for the Auxiliary Propulsion System (APS) design. The functions of providing vehicle control and gas generation for other Tug uses were combined to take advantage of the gas generating system required for vehicle control. In performing the study, invaluable information and assistance were received from the Rocketdyne Division of North American Rockwell. Due to the limited time for the study, point design system characteristics selections were based on the best information and estimates available at the outset of the study.

The APS, schematically shown in Figure 2.10-1, was designed to provide a total impulse of 430,000 LB-SEC per Ref. 2.10-1 which would satisfy the requirements for the baseline payload mission (3000 lbs.) and two larger payload missions. In addition, the system capacity was planned to permit simultaneous propellant (GOX and GH_2) withdrawal for a maximum thrust of 320 lbs., pre-pressurization of the main propellant tanks at 0.12 lbs. per second GH_2 and 0.53 lbs. per second GOX, and maximum demand by the fuel cell at 3 KW (0.333 LB per hr. GH_2 and 2.667 lb per hr. GOX).

Whenever the accumulator pressure in one of the systems drops to a predetermined pressure, the associated propellant conditioning system will operate to produce gas at the designed rate: 0. 5 lb. per sec. GH_2 for the GH_2 system, and 1.5 lb. per sec. GOX for the GOX system.

The repressurization rate of the accumulator to 1250 psia is determined by the difference in the rates for gas generation and gas withdrawal.

b. Description, Location, and Arrangement

Integration of the APS into the TUG Mechanical Systems is schematically shown in Figure 2.0-1, and installed in Figure 2.10-2 (Dwg V7-949112). The APS consists of two propellant conditioning systems (GH_2 and GOX), two accumulators, ten 70 LB_F and four 20 LB_F thrusters with an I_{sp} of 416 secs. and 409 secs. respectively, valving and lines. The propellant conditioning systems are arranged in two modules located on the thrust cone as shown in Figure 2.10-3. Each module consists of a turbopump and heat exchanger including flow controllers, igniters, and gas generators as well as valving and lines. The two accumulators (GOX 1.5 FT^3 and GH_2 2.4 FT^3) can be seen nearby and are supported by the aft skirt. The GOX and GH_2 are routed from the accumulators through regulators and lines to the points of use. These lines supply the thrusters which are located on the aft skirt, between stations 126 and 162, as shown in Figures 2.10-4 and 2.10-5. The thrusters are arranged in two pentad modules and two duad modules. The pentads are located on the

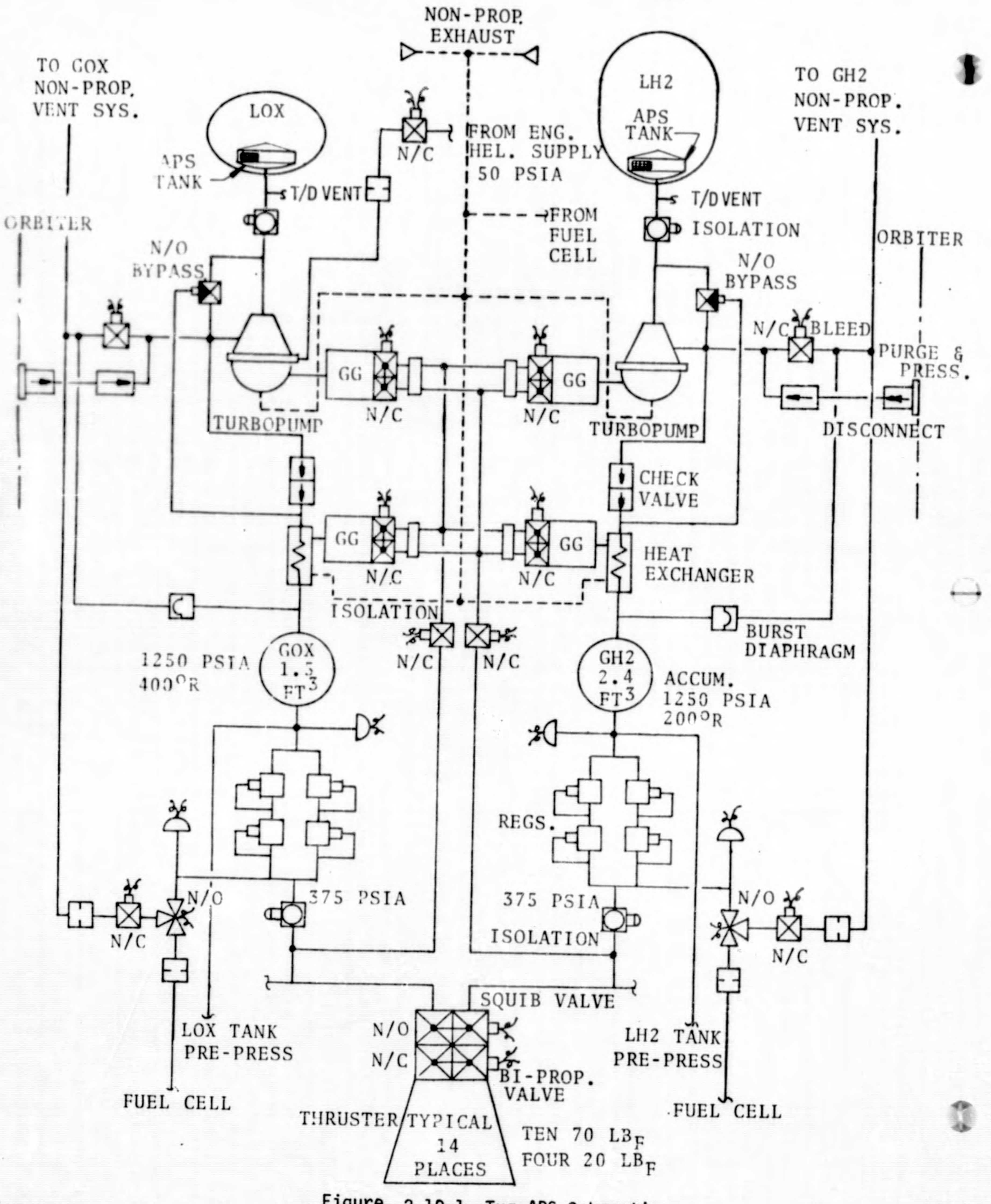
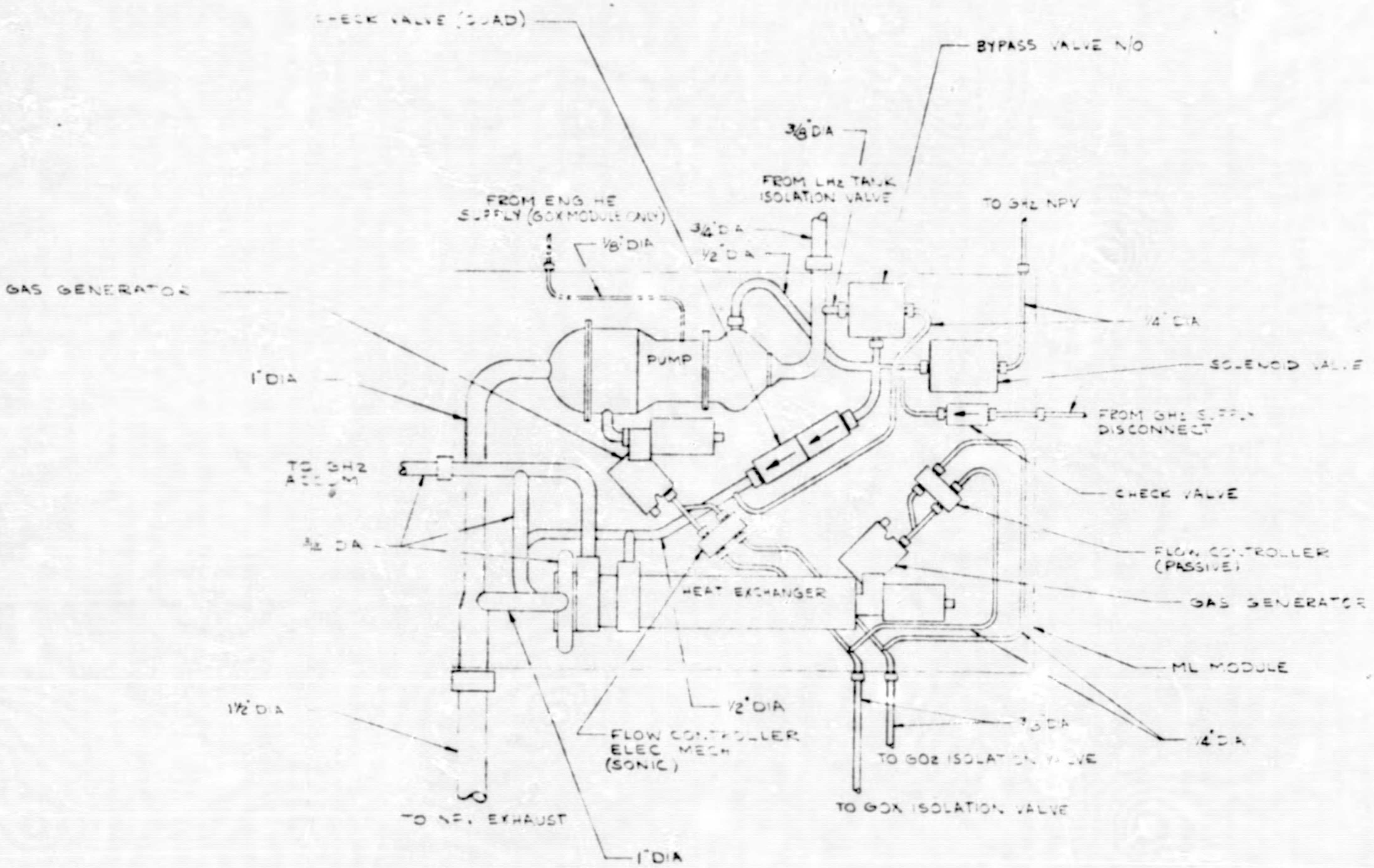
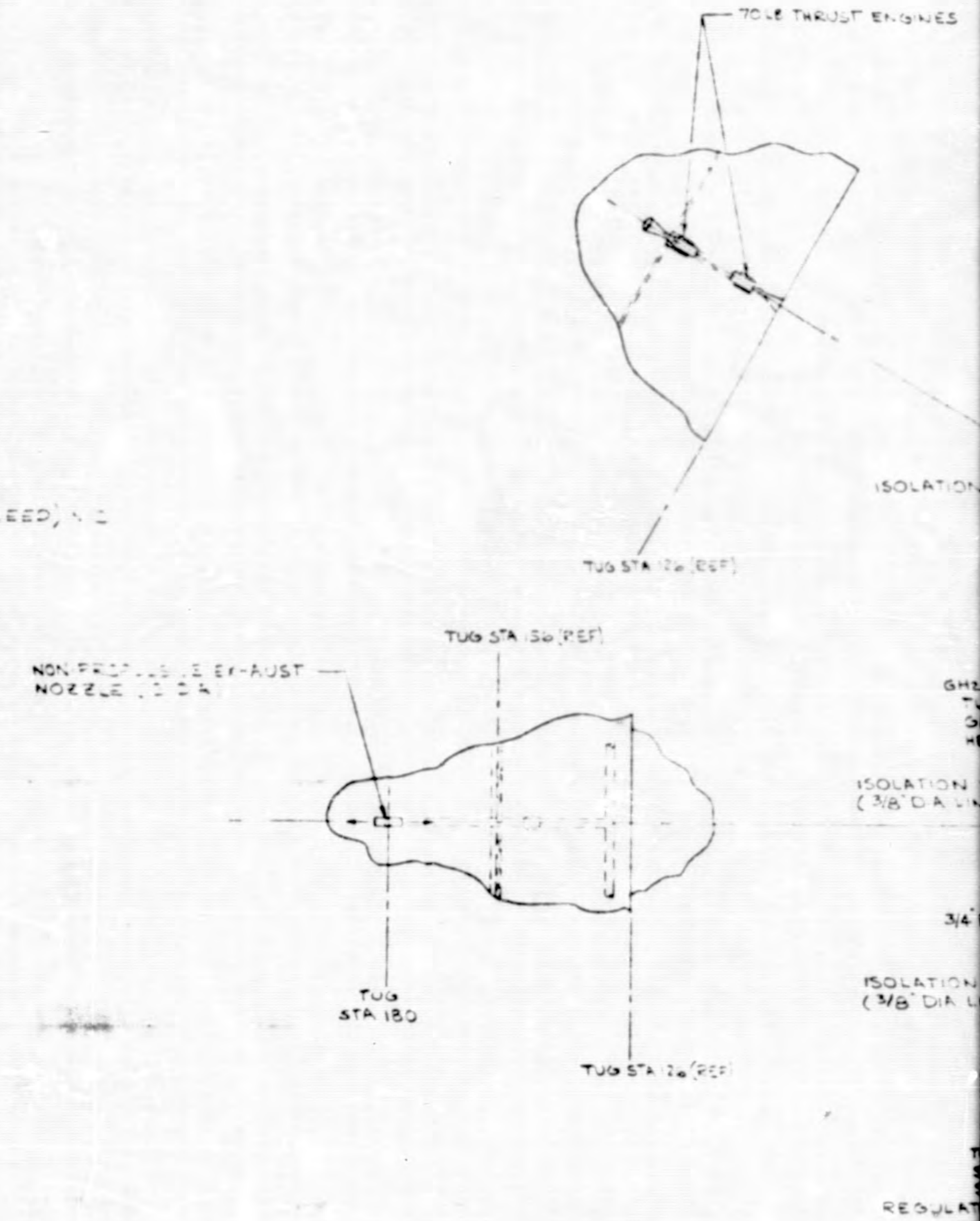


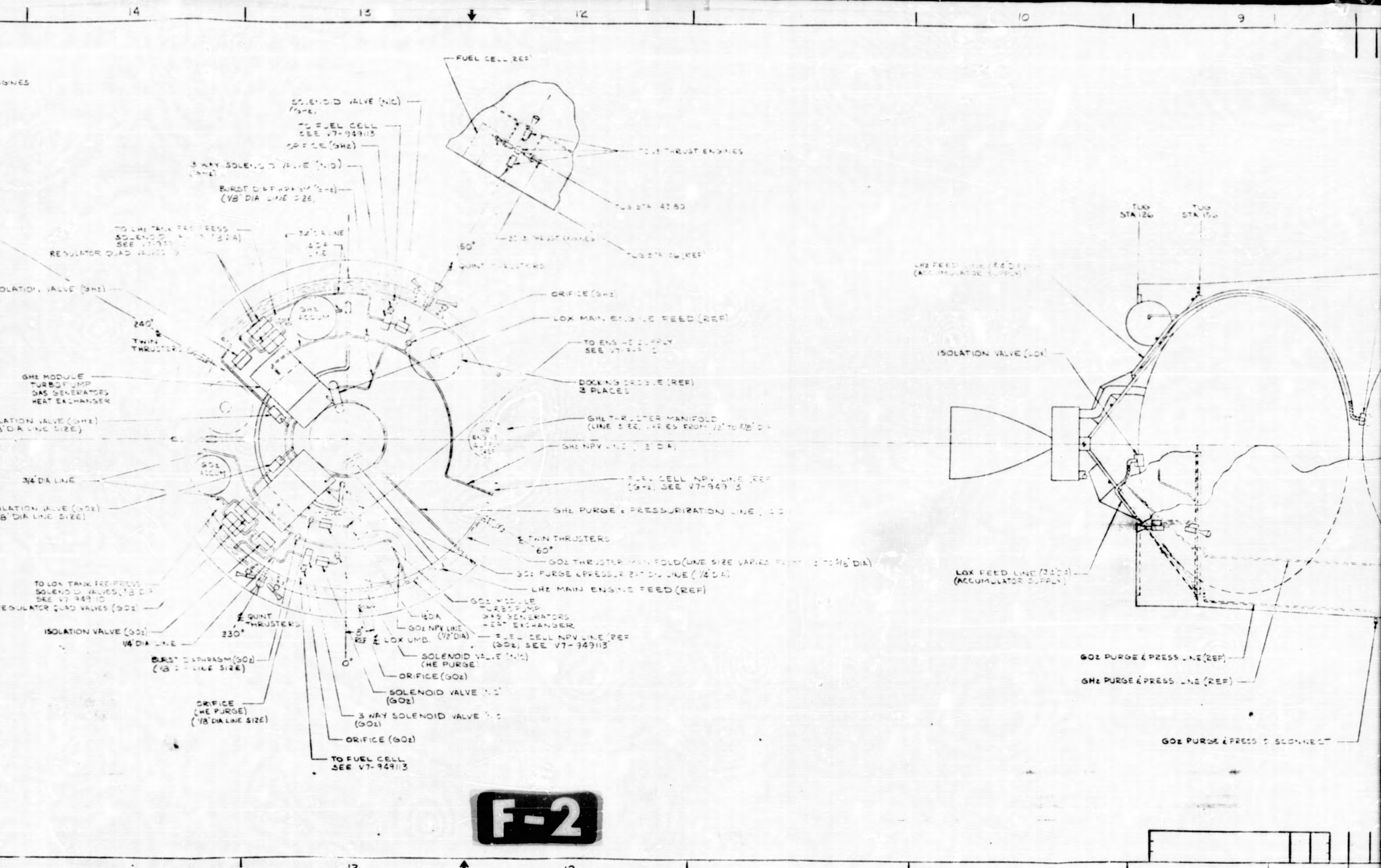
Figure 2.10-1 Tug APS Schematic



PLAN VIEW OF GH2 MODULE (ROT. C.C.W 45°)
GOX MODULE (OPPOSITE HAND) SAME AS GH2 EXCEPT AS NOTED
SCALE 1/4

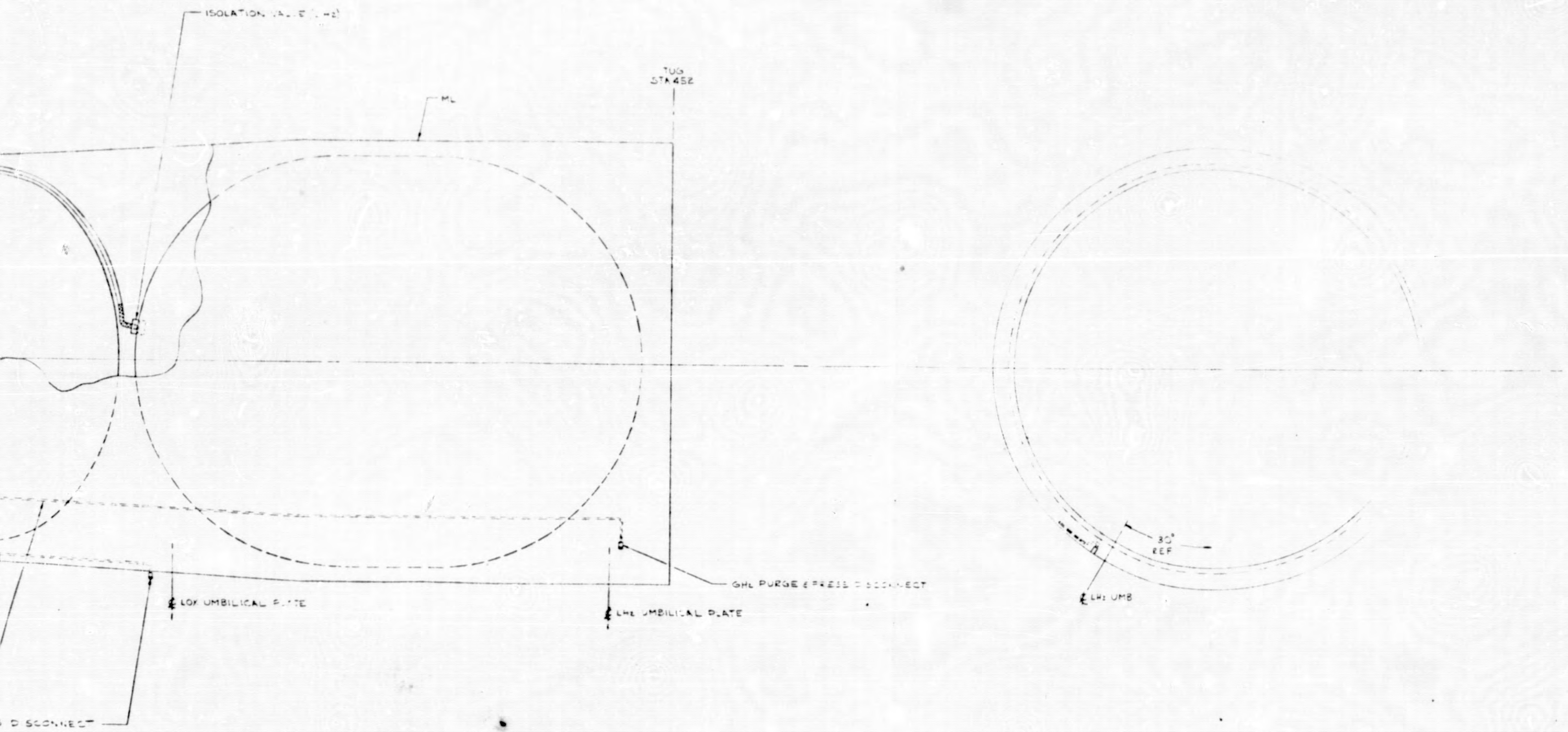


F-1

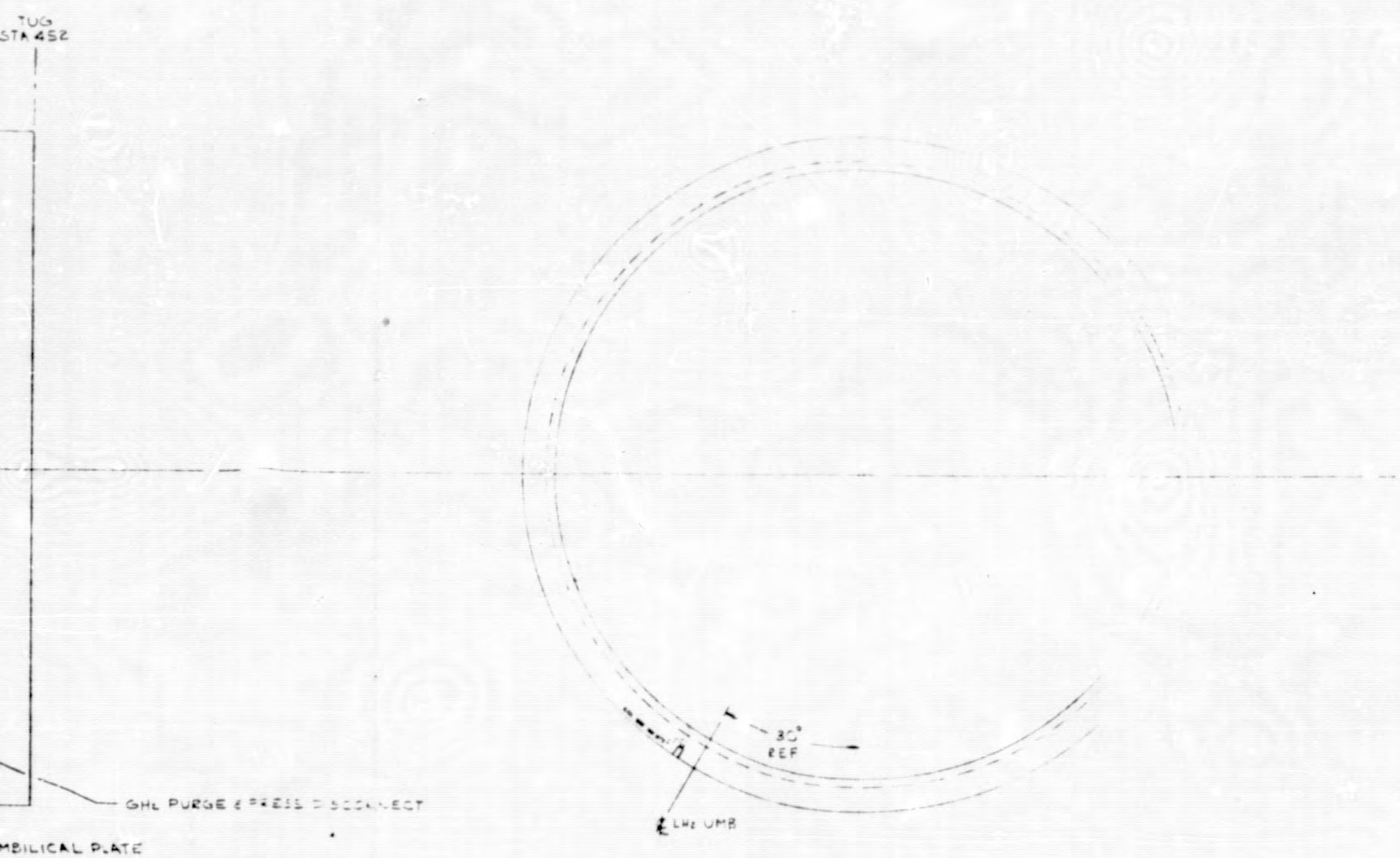


FUEL CELL (REF)
SOLENOID VALVE (N/C) (GH2)
TO FUEL CELL SEE V7-949113
ORIFICE (GH2)
3 WAY SOLENOID VALVE (N/O) (GH2)
BURST DIAPHRAGM (GH2) (1/8" DIA LINE SIZE)
TO LH2 TANK PRE-PRESS SOLENOID VALVES (1/8" DIA) SEE V7-
REGULATOR QUAD VALVES
ISOLATION VALVE (GH2)
240°
TWIN THRUSTERS
GH2 MODULE TURBOPUMP GAS GENERATORS HEAT EXCHANGER
ISOLATION VALVE (GH2) (1/8" DIA LINE SIZE)
3/4" DIA LINE
ISOLATION VALVE (GO2) (1/8" DIA LINE SIZE)
TO LOX TANK PRE-PRESS SOLENOID VALVES (1/8" DIA) SEE V7-949110
REGULATOR QUAD VALVES (GO2)
ISOLATION VALVE (GO2)
1/4" DIA LINE
QUINT THRUSTERS
330°
BURST DIAPHRAGM (GO2) (1/8" DIA LINE SIZE)
ORIFICE (HE PURGE) (1/8" DIA LINE SIZE)
LOX UMB.
SOLENOID VALVE (N/C) (HE PURGE)
ORIFICE (GO2)
SOLENOID VALVE (N/C) (GO2)
3 WAY SOLENOID VALVE (N/O) (GO2)
ORIFICE (GO2)
TO FUEL CELL SEE V7-949113
GO2 NPV LINE (1/2" DIA)
1/4" DIA
GO2 MODULE TURBOPUMP GAS GENERATORS HEAT EXCHANGER
FUEL CELL NPV LINE (REF) (GO2), SEE V7-949113
LH2 MAIN ENGINE FEED (REF)
GO2 PURGE & PRESSURIZATION LINE (1/4" DIA)
GO2 THRUSTER MANIFOLD (LINE SIZE VARIES FROM 1/2" TO 1/8" DIA)
60°
TWIN THRUSTERS
GH2 PURGE & PRESSURIZATION LINE
FUEL CELL NPV LINE (REF) (GH2), SEE V7-949113
GH2 NPV LINE
GH2 THRUSTER MANIFOLD (LINE SIZE VARIES FROM 1/2" TO 1/8" DIA)
DOCKING PROBE (REF) 3 PLACES
TO ENGINE SUPPLY SEE V7-
LOX MAIN ENGINE FEED (REF)
ORIFICE (GH2)
QUINT THRUSTERS
60°
TWIN THRUST ENGINES
TUG STA 26 (REF)
TUG STA 42.80
GH2 ACCUM
GO2 ACCUM
TUG STA 126
TUG STA 156
LH2 FEED LINE (ACCUMULATOR SUPPLY)
ISOLATION VALVE (LOX)
LOX FEED LINE (ACCUMULATOR SUPPLY)
GO2 PURGE & PRESS LINE (REF)
GH2 PURGE & PRESS LINE (REF)
GO2 PURGE & PRESS DISCONNECT
14
13
12
10
9

F-2


ISOLATION VALVE
TUG STA 452
ML
GH2 PURGE & PRESS DISCONNECT
LOX UMBILICAL PLATE
LH2 UMBILICAL PLATE
30° REF
LH2 UMB
DISCONNECT

F-3



REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED

Figure 2.10-2 Layout - System Installation, Reaction Control Sys, Space Tug

2-139,2-140

SIZE	CODE IDENT NO	DRAWING NO
L	03953	V7-949112

UCN 7007

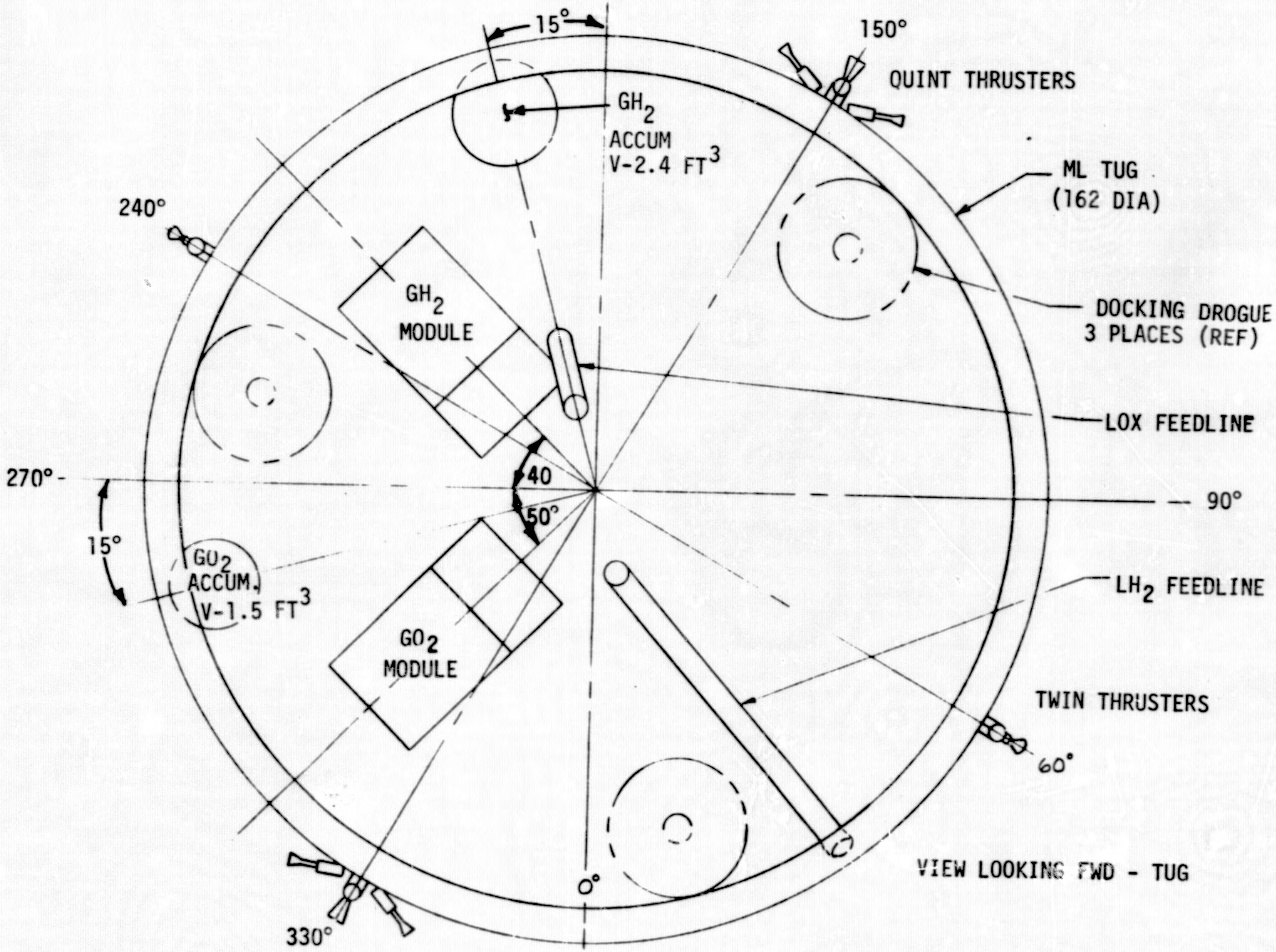


Figure 2.10-3 APS (Aft View)

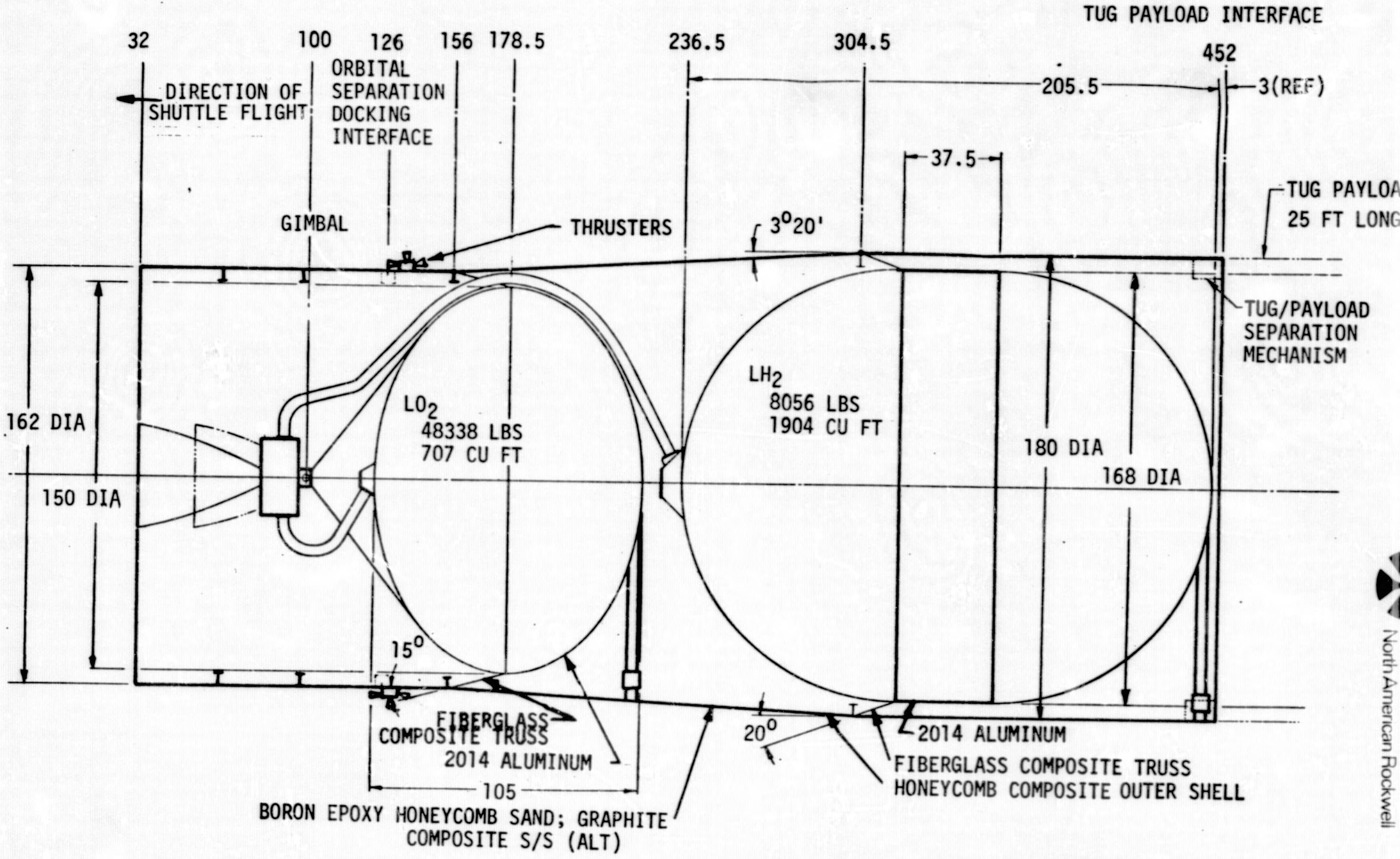


Figure 2.10-4 Thrusters Locations

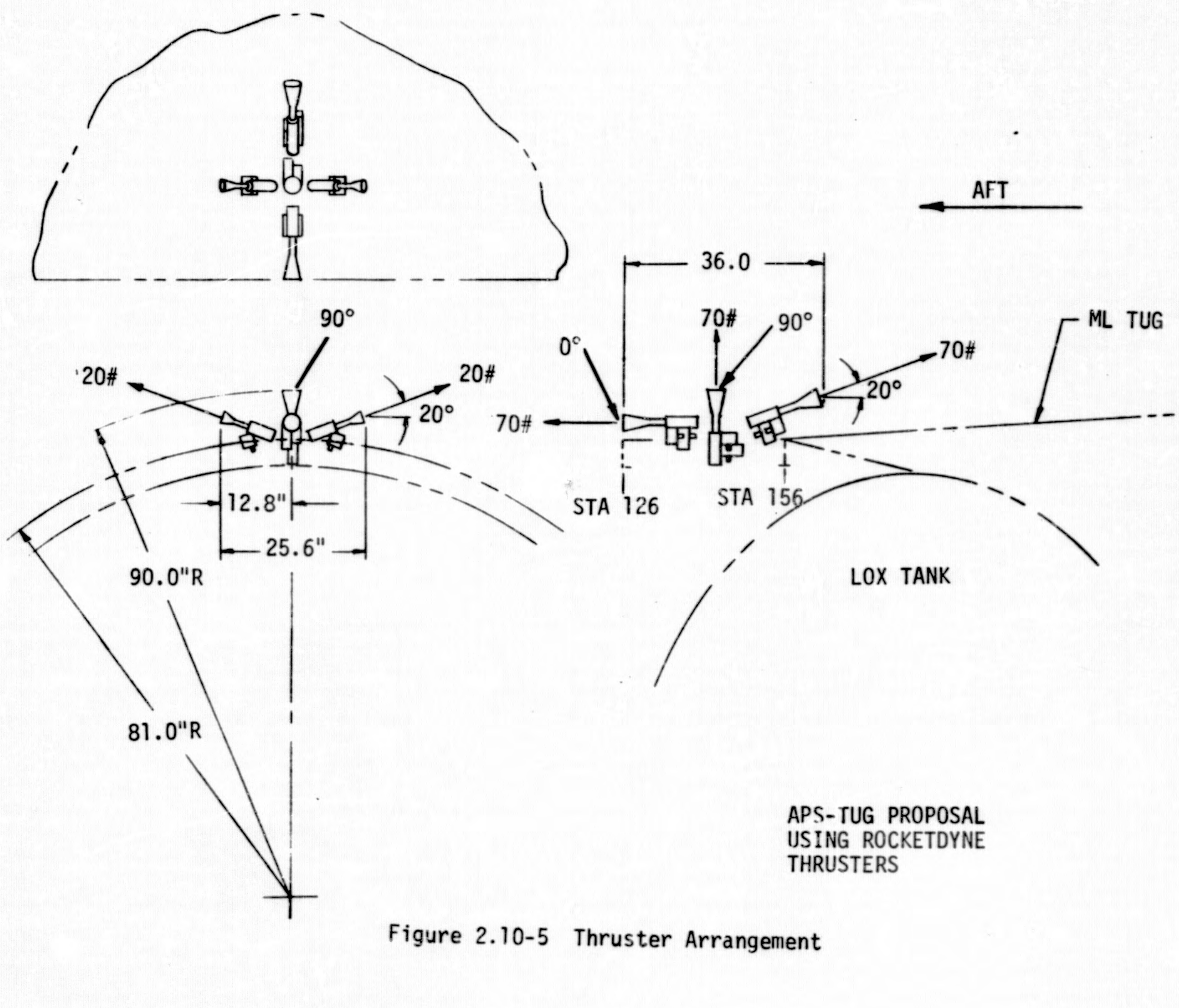


Figure 2.10-5 Thruster Arrangement

Y-axis and the duads on the z-axis, Figure 2.10-6. All thrusters are 70 LB_F except for two roll thrusters of 20 LB_F in each pentad. The number of thrusters (14 total) and arrangement including canting were planned to reduce plume impingement and weight, yet provide adequate control while meeting the fail-safe requirement. Figure 2.10-7 shows how attitude control can function with any thruster inoperative. A failed radial thruster will result in the loss of lateral control which is acceptable for attitude hold. The aft location of the thrusters also minimizes line requirements and is compatible with the Cargo Bay envelope.

Location of the APS on the aft skirt rather than in the intertank area resulted from several considerations. These included weight, plume impingement, and heating of the intertank area.

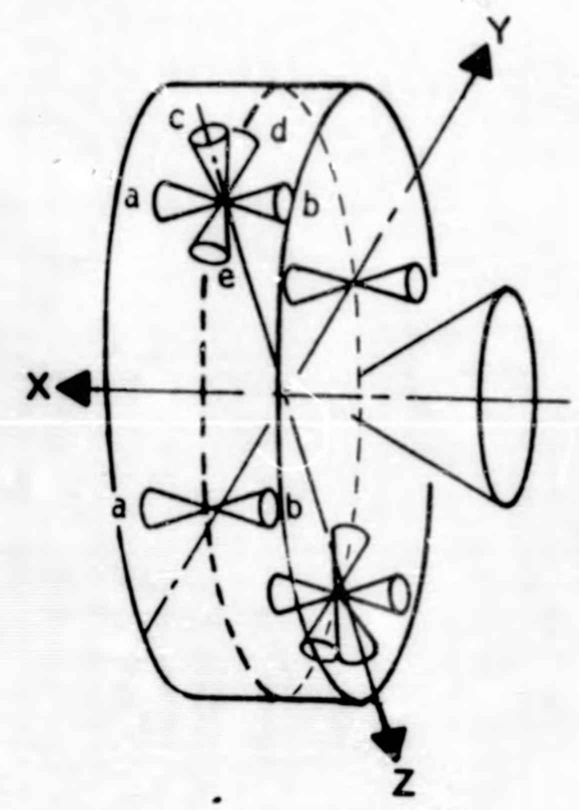
The thrusters were located at the aft skirt to reduce plume impingement on the Tug exterior. The intertank location would result in the plume impingement striking both the main LOX and LH_2 tanks. With the aft firing thrusters providing over 90% of the thruster total impulse, the intertank location could result in serious heating problems for the LOX tank. The aft skirt location for the thrusters minimizes the plume impingement on the guidance and navigation equipment and the two propellant tanks. Impingement on the LOX tank is reduced because the FWD firing thrusters would be used less than 10% as much as the aft firing thrusters.

The accumulators were not placed in the intertank area to minimize the lines between them and thrusters located on the aft skirt. The accumulators and their lines are also considered heat sources to the main propellant tanks even though they would be insulated.

The propellant conditioning subsystems (turbo pumps, gas generators, and heat exchangers) were placed on the thrust cone rather than the intertank area to avoid their being a heat source to the main propellant tanks. With the thrusters on the aft skirt exterior and the accumulators on the aft skirt interior, the thrust cone location for the propellant conditioning would minimize lines to the accumulators.

Venting of GH_2 and GOX from the APS is accomplished through the main propellant tank vent system. A non-propulsive exhaust system is provided for the APS which is shared by the fuel cell. Purging and pressurization of the APS is accomplished prior to liftoff through an interface with the orbiter to the GSE.

Some of the APS operational characteristics are as follows. The thrusters burn GH_2 and GOX at a mixture ratio of 4:1 and chamber pressure of 250 psia with an expansion ratio of 60:1. At present, pulse widths of 20 MSEC and 50 MSEC for the 20 LB_F and 70 LB_F thruster can be realized. By 1976, it is anticipated that a pilot operated bi-propellant valve can be developed with a 25 MSEC pulse width for the 70 LB_F thruster. As a result, the APS design is based on the 25 MSEC pulse width. The gas generators for the turbopumps operate at a mixture ratio of approximately .97:1 and chamber pressure of 270 psia. The integral gas generators for the heat exchangers operate at a mixture ratio of 3:1 and chamber pressure of 250 psia. The system mixture ratio for operation of the propellant conditioning system and thrusters is approximately 3.7:1,



	ENGINES a	b	c	d,e
• THRUST (LB)	70	70	70	20
• CANT ANGLE (DEG)	20	0	90	20
• MAX PULSE (SEC)	120	360	30	60

Figure 2.10-6 Thruster Arrangement

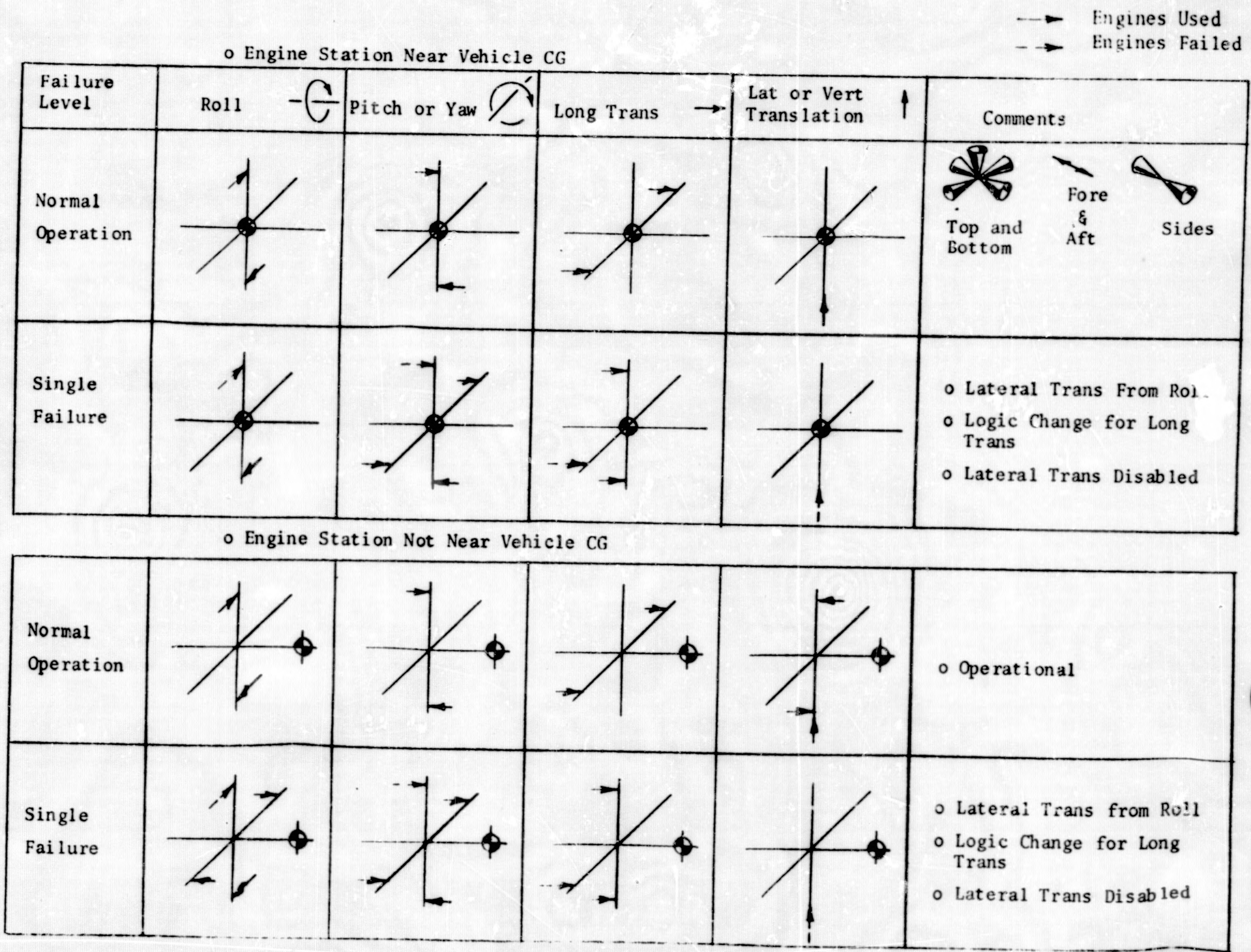


Figure 2.10-7 APS Redundancy Analysis for a 14-Engine System

while the mixture ratio for all operational uses is 3.9:1. The propellant conditioning/thruster system I_{sp} is 381 secs based on an average thruster I_{sp} of 415 secs. Therefore, the ground rule for a system I_{sp} of 380 secs has been met.

Thermal considerations in the operation of the turbopumps were responsible for utilizing a thermodynamic vent system to cool the turbopump feed and outlet lines, and the pumps. The thermal conditions due to exposure of the system to direct sunlight will require the insulation of the accumulator inlet lines and outlet lines as well as the accumulators. Sun shields will be required for the accumulators and could be used for the propellant conditioning system. The incorporation of emissive surfaces and insulation to direct the radiant heat from the heat sources (turbines, heat exchangers, and exhaust lines) in harmless directions might be required.

c. System Operation

In describing the planned system and component operations for the APS, the schematic shown in Figure 2.10-8 will be utilized. The components have been numbered to facilitate identification. Since the LOX/GOX and LH_2/GH_2 systems are similar, the same components will have identical numbers. The two systems will be discussed as a single system for clarity and brevity except where differences exist. The system operation for ground checkout through orbital starting and safing for re-entry will be described.

Ground Checkout. Prior to liftoff preparations, the APS operational readiness can be verified. The 115 VAC isolation valves (3) will be closed and the accumulators (15) charged with helium to 1250 psia through disconnect (1) and check valves (2) (4). The turbopump bypass valve (5) should close whenever the pressures (accumulator and pump outlet) on both sides of check valves (4) are equal. Seating of check valve (2) can be verified by bleeding off the upstream GSE pressure and checking for leakage. Since the charging pressure is trapped between check valve (2) and isolation valve (3), the pump bypass valve will open when the pressure upstream of check valve (4) is vented off by opening bleed valve (6). The system checkout can be performed by using a programmed sequence that will verify operation of each component. Operation of the turbopump gas generators (12) will have to be carefully controlled to avoid excessive pump spinning. Pressure switch operation can be verified by means of a portable pressure source.

Liftoff Preparations. Prior to liftoff, the isolation valves (3) and (18) will be closed when the accumulators (15) are charged through disconnect (1) with GH_2 and GOX to 1250 psia at 200°R and 400°R respectively. To chill-down the insulated accumulators in the nitrogen purged Cargo Bay and maintain them within a ±25°R tolerance, it will be necessary to flow the gases through the quad regulators (17), the 3-way fuel cell/vent selector valve (19), vent valve (20) and orifice (21) to the vent lines. The quad regulators (17) supply the gases at 375 ± 25 psia. The orifices (21) will flow approximately 0.05 LB per sec GH_2 and 0.1 lb, per sec GOX. When chilled prior to liftoff, the charging pressure must be bled from the system through bleed valve (6) to permit opening isolation valve (3). However, before bleeding off this trapped

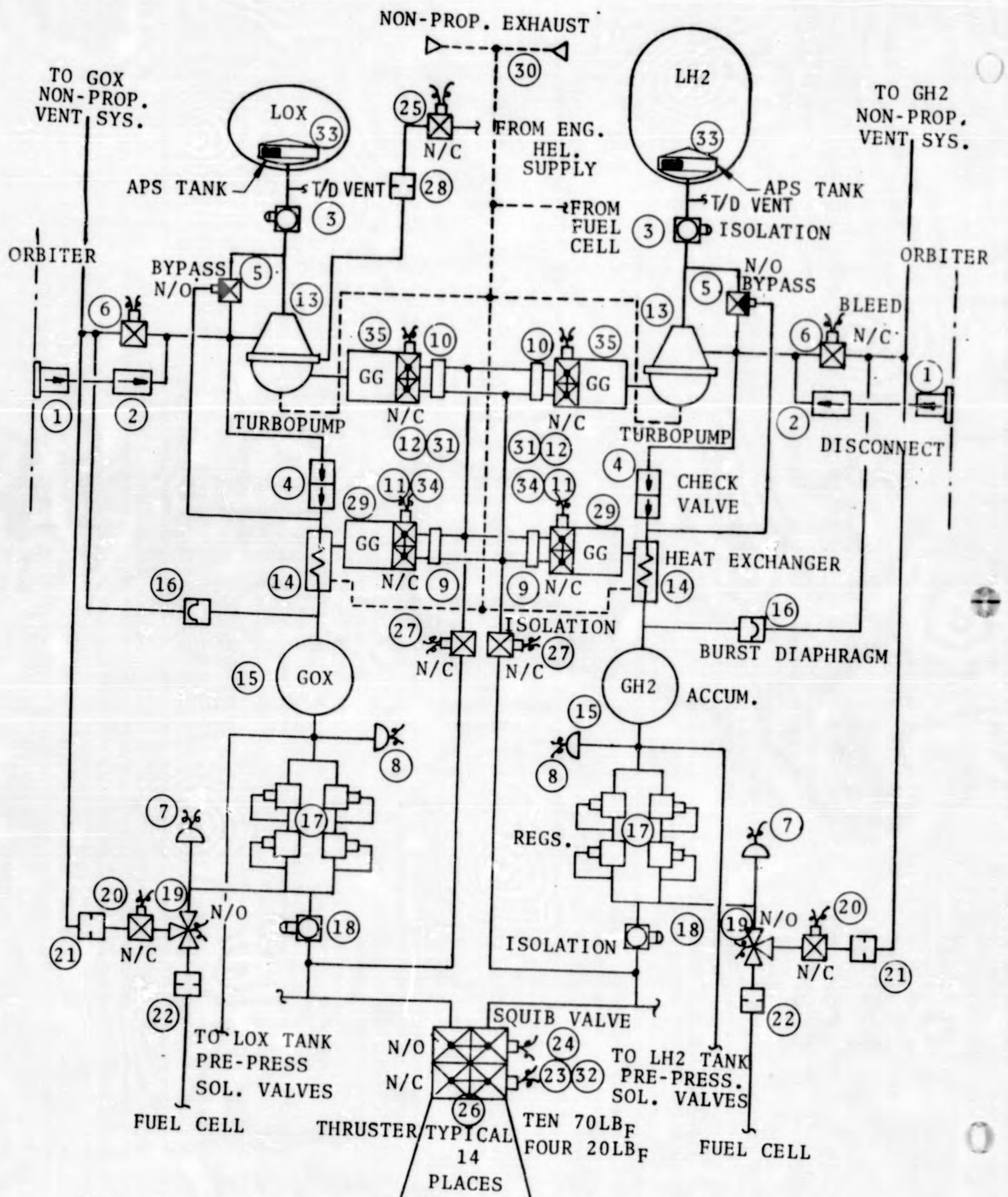


Figure 2.10-8 APS Schematic (Components Numbered to Facilitate Description of System Operation)

gas, proper seating of check valve 2 may be verified as previously described. After separating from the Orbiter, the check valve in disconnect 1 will be redundant to check valve 2. Upon completion of bleeding the trapped high pressure gas through bleed valve 6, the isolation valve 3 may be opened. The isolation valve 18 will be held closed throughout boost to provide redundancy to all downstream valves to prevent the creation of hazardous conditions due to the possible leakage or flow of GOX/GH_2 into the Orbiter Cargo Bay.

Orbital Launch. Verification of all APS valve positions and system pressures can be conducted prior to swinging out the TUG from the Cargo Bay. After the TUG has been swung out but is still electrically connected to the Orbiter, the 115 VAC motor operated isolation valves 18 will be opened. A programmed operational check of the entire system can now be performed. However, operation of the thrusters 26, gas generator bi-propellant valves 11 12, and flow controllers 9 10 must be conducted without concurrent igniter 23 31 34 operation. This checkout will also permit clearing all lines of trapped helium. The trapped helium serves as an additional barrier to GOX/GH_2 leakage. To remove the trapped helium prior to liftoff would have created hazardous conditions in the Cargo Bay since GOX and GH_2 could also flow into the Cargo Bay. Included in the checkout will be operation of the bleed valve 6 to remove trapped gas between the screened auxiliary tank 33 and the check valve 4 so the feed line can fill with liquid propellant. The thermodynamic vent system will be operational at this time to cool the feed lines and pumps. Should it be required, the bleed valve could be operated prior to each operation of the propellant conditioning system. However, it would result in a propellant penalty. Concurrently with these preparations, the fuel cell will be started, and proper operation verified.

After the Orbiter has electrically and physically separated from the Tug and is adequately clear, the APS will be ready for normal operation. Normal operation includes settling main tank propellants for main engine operation in idle mode and then pressurizing the main tanks for main stage start. Idle mode is utilized to purge and chill the main engine for approximately 120 seconds prior to main stage operation.

Prior to starting the main engine in idle mode, commands from the control system will open the appropriate thruster bi-propellant valves 32, and energize the igniters 23. The resulting thrust will be utilized to settle propellants in the main tanks and control TUG attitude. The propellant settling thrust will be terminated approximately 5 seconds after start of idle mode. Next, pre-pressurization of the main tanks will be initiated 105 seconds prior to main stage operation. The main tanks will be pre-pressurized for approximately 100 seconds from the APS accumulators. GOX/GH_2 flow from the accumulators 15 through redundant solenoid valves and regulators to provide the ullage pressure required for initiation of main stage. During the interval between termination of propellant settling and main stage, the APS will provide attitude and roll control. For main stage operation, the APS will only furnish roll control as necessary.

When the accumulator pressures have dropped to 1050 psia for GH_2 and 925 psia for GOX, the pressure switch 8 will initiate commands to

repressurize the accumulators. The turbopump 13 and heat exchanger 14 will be operated by opening the propellant conditioning system isolation valves 27, gas generator valves 11 12, and energizing flow controller 10 and igniters 31 34. The isolation valves 27 are redundant to the gas generator bi-propellant valves 11 12 to prevent loss of gas due to leakage between operations. The electro-mechanical flow controller 10 and the passive flow controller 9 will regulate flow to the gas generators 35 29 to provide the required mixture ratio for proper combustion, and operation of the turbopump 13 and heat exchanger 14. The turbopump 13 and heat exchanger 14 exhaust gases will vent through the non-propulsive exhaust system 30.

Propellants for the turbopumps 13 will flow from the screened auxiliary tanks 33 through the isolation valves 3. The bypass valves 5, which are normally open, will permit the pump outlet to recirculate to the pump inlet until the pump outlet pressure is equal to the accumulator 15 pressure. The bypass valve 5 will then close and the turbopump output will open the redundant check valves 4 when the pump pressure exceeds the accumulator 15 pressure. The propellants will then flow through the heat exchangers 14 where they are vaporized and superheated to 400°R for GOX and 200°R for GH_2. From the accumulators 15, the gases will flow to meet all requirements. The accumulators have been sized to contain sufficient gas to allow simultaneous flow demands by all systems for 3 seconds from an initial pressure of 925 psia for GOX and 1050 psia for GH_2 to 575 psia for both. Within 3 seconds, the propellant conditioning system will supply gases to the accumulators at greater rate than the maximum anticipated usage.

Consequently, the accumulator pressures should not drop to less than 575 psia. When the accumulator 15 pressures have returned to 1250 psia, the pressure switch 8 will initiate commands to shutdown the propellant conditioning system. The LOX turbopump helium supply, which is turned on whenever the LOX turbopump is operated, continues to flow for 5 additional seconds after the pump is shut down. The helium gas (from the main engine helium btl) at 50 psia flows through the control valve 25 to actuate the turbopump lift-off seals and purges the intermediate seals separating the turbine and pump.

Cargo Bay Safing. After the Orbiter has docked and mated electrically with the TUG, the APS will be safed for return to the Cargo Bay. The safing will consist of closing the isolation valves 18 and propellant conditioning system isolation valves 27. The gas generator and thruster bi-propellant valves 11 12 32 will be cycled without energizing the igniters 23 31 34 to vent the gases trapped downstream of the isolation valves. The APS is now ready for entry into the Cargo Bay. After entering the Cargo Bay and mating with the Orbiter, the accumulators will be emptied into the main propellant tanks to assist in dumping propellants. Due to the Orbiter ΔV to settle the Tug propellants for dumping and the Tug attitude (upside down in the Orbiter), the propellants in the APS propellant conditioning system upstream of the check valves 4 will flow into the main tanks. During the purging process for the main tanks, the APS can be purged with the main propellant tank gases by using bleed valve 6 and then the vent valves 19 and 20. After completing the APS purge, the system will be at main tank pressure. The pressure will be safe for the APS under all re-entry and ground conditions.

Malfunction Protection. Whenever accumulator 15 pressure exceeds 1450 psia, the pressure switch 8 will initiate commands to shut down the affected propellant conditioning system and open the vent valves. The fuel cell/vent selector valve 19 and vent valve 20 will be operated to let gas flow to the main propellant tank non-propulsive vent line until the accumulator 15 pressure returns to 1250 psia. Likewise, if the pressure downstream of the regulator 17 exceeds 500 psia, the pressure switch 7 will operate the fuel cell/vent selector valve 19 and vent valve 20 to vent the system pressure until it reaches 425 psia.

The system redundancy and safing concept for fail safe have been planned to permit the failure of any component including the fuel cell and still maintain the capability of providing one-half hour of attitude hold with emergency power. When the attitude hold mode is used due to safing after a component failure, the propellant conditioning systems will be shut down, isolation valves 27 closed and the fuel cell/vent selector valve 19 moved to the vent position. The pre-pressurization control valves will be closed to prevent the flow of gas into the main propellant tanks. The gas in the accumulators 15 will be used only for thruster operation to perform attitude hold.

Squib valves 24 are used to permanently isolate any malfunctioning thruster from the remainder of the system. When a Squib valve is operated, the system will be placed in the attitude hold mode as described above. The burst diaphragm 16 functions at 1750 psid to prevent catastrophic failure of the accumulators should the system fail in a manner that will prevent controlled venting. The burst diaphragm outlet flows into the main propellant tank vent system to avoid creating disturbing forces.

d. Control Logic

Pressure, temperature, and component position switches are utilized to detect specific conditions and provide discrete indications for control of the system. Pressure and temperature transducers also facilitate the control and evaluation of system conditions during startup, shutdown, and steady-state operation. The control system will consist of an on-board computer and solid state power control switching. The guidance and navigation computer and the flight computer will issue commands to the control system whenever operation of the thrusters is required. The flight computer will also control the pre-pressurization of the main propellant tanks by the APS.

To comply with the ground rule fail safe criteria, the control system must be able to detect a component failure or malfunction, take appropriate action to shutdown and isolate the propellant conditioning systems, provide notification of the failure, and limit the use of the stored accumulator gas to thruster operation for attitude hold. This action would shift the vehicle to emergency power, because it would stop the flow of reactants to the fuel cell. Likewise, a shift to emergency power for any other reason would cause the APS control system to respond similarly to an APS failure. In both cases, the APS would limit the use of its stored gas to attitude hold. The minimum attitude hold capability is one-half hour.

e. Interface Requirements

The APS has several interface requirements. They consist of those required for proper operation and functioning of the system, and for utilization of the APS services by other systems. The interfaces will now be described.

Electrical and Instrumentation. An electrical and instrumentation interface is required for control and operation of the APS. Sensors and position switches supply the APS control system with information on the condition of the system, pressure and temperature switches activate system operations, and electrical power is required for component operation. The system utilizes 28 VDC except for motor operated isolation valves which require 115 VAC. The 115 VAC is supplied by the GSE and Orbiter and is not required for Tug operation.

Guidance and Control. The guidance and control system issues commands to operate the appropriate thrusters for the desired vehicle changes in attitude and velocity. When sufficient change occurs in the APS accumulator pressures, the APS control system activates the propellant conditioning system to repressurize.

Propellants, Charging, Purging, and Venting. The APS interfaces with the main tank propellants through the screened auxiliary tank. This tank is the source of propellants for the APS turbopumps. The APS receives helium, hydrogen and GOX from the GSE through two 1/4 inch lines which interface with the Orbiter for purging and pressurization. Helium is supplied through the GOX and GH_2 interfaces. The gases are provided at 1250 psia, and the GOX and GH_2 are at 400°R and 200°R respectively. The accumulators are charged with GOX and GH_2 prior to liftoff. The helium is used for purging and checkout. However, an interface with the main engine helium system is used to provide purge gas for the LOX turbopump seals during operation. Venting of system gases is accomplished through an interface with the TUG main propellant tank non-propulsive vent system. Purging of the APS is performed by flowing gas from the purge and pressurization interface through the APS and into the main tank non-propulsive vent system.

Main Propellant Tank Pressurization and Fuel Cell. The APS supplies GOX and GH_2 to pressurize the main propellant tanks prior to starting the main engine in main stage, and to provide reactants for the fuel cell. The APS continually supplies the fuel cell demand, and pressurizes the main tanks when the pre-pressurization solenoid valves are commanded open by the flight computer.

f. System Performance

Propellant Flow Rates. The APS was designed to provide thrust for attitude control and simultaneously pressurize the main propellant tanks and provide reactants for the fuel cell. The propellants for these applications are supplied as GOX and GH_2. To do this, LOX and LH_2 are withdrawn from the screened auxiliary tank in the main tank and passed through GOX and GH_2 propellant conditioning systems. Here, the liquid propellants are converted to

400°R GOX and 200°R GH_2 at 1250 psia. The GOX and GH_2 then flow to the accumulators for use by the thrusters, main tanks, fuel cell and propellant conditioning systems.

The APS was sized to produce GOX and GH_2 at a greater rate than the maximum anticipated demand. This permits the system to meet the maximum demand and simultaneously pressurize the accumulators. Based on estimates made at the outset of the study, the propellant conditioning system of the APS was selected to have the flow capacity of 0.5 #/sec GH_2 and 1.5 #/sec GOX. The difference between the flow rates for supply and demand is available to repressurize the accumulators.

The maximum flow rate was based on the following consideration: The maximum anticipated thrust for propellant settling or attitude control is 320 lbs. It is provided by four 70 LB_F thrusters and two 20 LB_F roll thrusters. The flow rates to the main propellant tanks were arbitrarily determined by specifying 100 secs for pre-pressurizing the main propellant tanks with the largest ullage anticipated during the mission. The maximum GOX/GH_2 consumption by the fuel cell (3KW) was selected since such a demand could occur when preparations are being made for operation of the main engine. In addition, the APS propellant conditioning systems were designed for operation at fixed rates. Therefore, the propellant conditioning systems will always use GOX and GH_2 at constant rates for converting the LH_2 and LOX to GH_2 and GOX at the designed rates. The flow rates for the preceding requirements and propellant conditioning are as follows:

	GH_2 #/sec	GOX #/sec
Thrusters (320 LB_F)	0.1544	0.6176
Main Tank Press.	0.12	0.53
Fuel Cell (3 KW)	0.0000927	0.000741
Prop. Cond.	0.0584	0.1094
TOTAL	0.3329	1.2577

Following is a summary of the turbopump and heat exchanger propellant requirements for the designed flow of 0.5 #/sec GH_2 and 1.5 #/sec GOX.

$\dot{w}_{cold\text{-}side}$ (lb/sec)	GG	$\dot{w}_{hot\text{-}side}$ (lb/sec)	$\dot{w}_{H_2}$ (lb/sec)	$\dot{w}_{O_2}$ (lb/sec)
0.5	H_2 Pump	0.036	0.0184	0.0176
1.5	O_2 Pump	0.027	0.0138	0.0132
0.5	H_2 Conditioner	0.064	0.0160	0.0480
1.5	O_2 Conditioner	0.0408	0.0102	0.0306
2.0	Total	0.1678	0.0584	0.1094

The resulting net flow rates of 0.4416 #/sec GH_2 and 1.3906 #/sec GOX are available for thruster operation, main tank pressurization, fuel cell operation, and recharging the accumulators. Based on the design considerations for the maximum demand, a net flow rate of 0.167 #/sec GH_2 and 0.25 #/sec GOX will remain for repressurization of the accumulators and as an operating margin. In optimizing the system design, this margin could be reduced with some weight saving anticipated.

Operating Characteristics. The APS design was planned to provide instantaneous response for propellant requirements by storing 400°R GOX and 200°R GH_2 in two accumulators at 1250 psia. To minimize accumulator size and weight, GOX and GH_2 propellant conditioning systems were selected to supply the accumulators at a greater rate than the maximum expected demand. Consequently, the accumulator sizes were determined by the desired frequency of propellant conditioning system operation to repressurize the accumulators between main engine operations and to have sufficient capacity to meet all propellant flow requirements until the propellant conditioning system can respond.

The propellant conditioning system is expected to operate every four hours to repressurize the accumulators due to GOX/GH_2 usage by the fuel cell for a 1 KW output. To provide this operating frequency, the GH_2 propellant conditioning system will be activated at a GH_2 accumulator pressure of 1050 psia and the GOX propellant conditioning system at a GOX accumulator pressure of 925 psia. The system operations are independent. Consequently, the accumulator pressures of 1250 psia to 1050 psia for GH_2 and 1250 psia to 925 psia for GOX will be the dead bands for propellant conditioning system operations. When these pressures (lower) are reached, the propellant conditioning system will be activated.

The accumulators were sized, 2.4 ft^3 for GH_2 and 1.5 ft^3 for GOX, to contain sufficient gas to satisfy the maximum flow demands of 0.333 #/sec GH_2 and 1.26 #/sec GOX for three seconds. During this period, the propellant conditioning systems will be able to start and deliver propellants at the designed rates before the accumulator pressures drop to 575 psia. This pressure is the minimum desired to provide the gas required for one-half hour of attitude hold should the propellant conditioning systems be stopped at this time due to an APS malfunction. The 575 psia allows the accumulators to contain sufficient gas for one-half hour of attitude hold before reaching 475 psia. This is considered the minimum inlet pressure for the proper operation of the 375 psia regulators. The regulator output is the thruster supply pressure. Lower regulator inlet pressures are expected to degrade thruster performance by delivering less than 350 psia. Following is a table showing the distribution of the propellants used by the APS propellant conditioning system to supply all requirements. The data has been adjusted to include the operation of the propellant conditioning system:

	GH_2 lbs	GOX lbs	MR
Thruster Impulse	240.0	888.0	3.7
Main Tank Press.	23.0	86.0	3.73
Fuel Cell	17.0	116.5	6.89
Combined	280.0	1090.5	3.9

The overall mixture ratio for all propellant usage is 3.9. Based on an average I_{sp} of 415 secs., the combined operation of the propellant conditioning systems and thrusters results in an I_{sp} of 381 secs. and a mixture ratio of 3.7. (This I_{sp} meets the ground rule requirements of 380 secs.) The thrusters are required to furnish a total impulse of 427,000 lb-secs. For system design, a total impulse of 430,000 lb-secs. was selected as a maximum. Following is a distribution of the impulses for the baseline 3000 LB payload mission and two other payloads:

Auxiliary Propulsion System Mode	Impulse Required for Mission Payload (lb-sec)		
	3,000# P.L.	8,060# P.L.	4,160# P.L.
Hold Attitude	35,100	20,100	5,000
Attitude Maneuver	6,176	6,300	4,550
Propellant Settling	18,950	17,270	19,560
Translation Maneuver	367,500	141,030	218,150
Totals	427,726	184,700	247,260

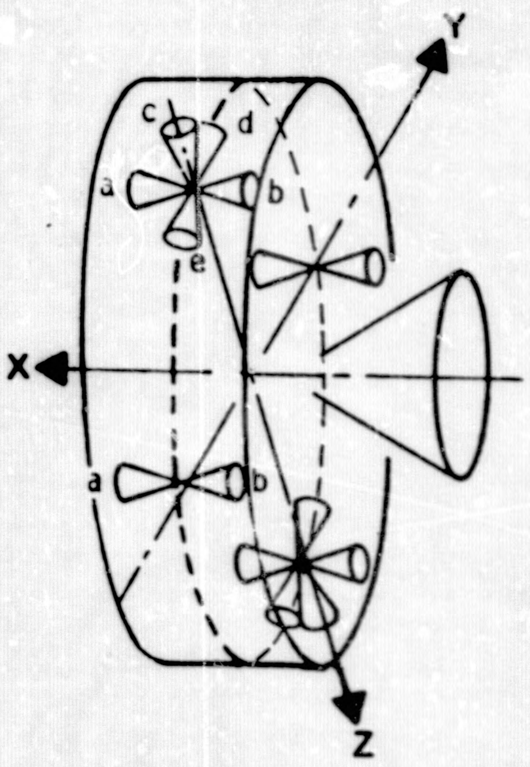
As can be seen, approximately 93.5% of the total impulse (3000 LB payload) is used for propellant settling and translation (ΔV). This means a very large proportion of the impulse will be expended by the aft firing thrusters. The location of these thrusters on the aft skirt minimizes the plume impingement on the TUG.

g. Components

Introduction. The operating, performance, and physical characteristics of the major and some minor components in the auxiliary propulsion system will be described within the limits of present information. The physical, electrical and operating characteristics of these components are compiled in three tables at the end of this section.

The information presented on the thrusters, turbopumps, heat exchangers and flow controllers was received from Rocketdyne.

Thrusters. The thrusters selected for the auxiliary propulsion system are non-throttleable 20 LB_F and 70 LB_F. They are arranged in two pentad modules of three 70 LB_F and two 20 LB_F thrusters, and two duad modules of two 70 LB_F thrusters. Figures 2.10-9, 2.10-10, and 2.10-11 show the thruster arrangement in the modules and location of the modules on the aft skirt exterior. The forward firing and roll thrusters are canted 20° from the vehicle tangent to reduce plume impingement on the TUG exterior. Over 90% of the thruster impulse for the mission will be performed by the aft firing thrusters. These thrusters are not canted, and the nozzle exit is flush with the end of the aft skirt (Sta. 126). The thrusters are not shielded for meteoroid protection to save weight, and because such shielding does not appear necessary due to the material thicknesses of the exposed surfaces.



- GROUND RULES
  - FAIL SAFE ATTITUDE CONTROL
  - MINIMUM WEIGHT
  - MIDCOURSE CORRECTION CAPABILITY
  - GO_2/GH_2 PROPELLANT (I_{SP} = 380 SEC)
- ACCURACY REQUIREMENTS FOR DOCKING
  - CENTERLINE MISS DISTANCE: 0 TO 1.0 FT
  - MISS ANGLE: 0 TO 5.0 DEG
  - LONGITUDINAL VELOCITY: 0.1 TO 1.0 FT/SEC
  - LATERAL VELOCITY: 0 TO 0.3 FT/SEC
  - ANGULAR VELOCITY: 0 TO 0.5 DEG/SEC
- DESIGN REQUIREMENTS
  - 14 ENGINES
  - 430,000 LB-SEC TOTAL IMPULSE
  - 25 MS MIN SQUARE PULSE DURATION
  - NOMINAL ENGINE STATION ≅ 146.5 IN.

	ENGINE			
	a	b	c	d,e
• THRUST (LB)	70	70	70	20
• CANT ANGLE (DEG)	20	0	90	20
• MAX PULSE (SEC)	120	360	30	60

Figure 2.10-9 Auxiliary Propulsion System Flight Control Description

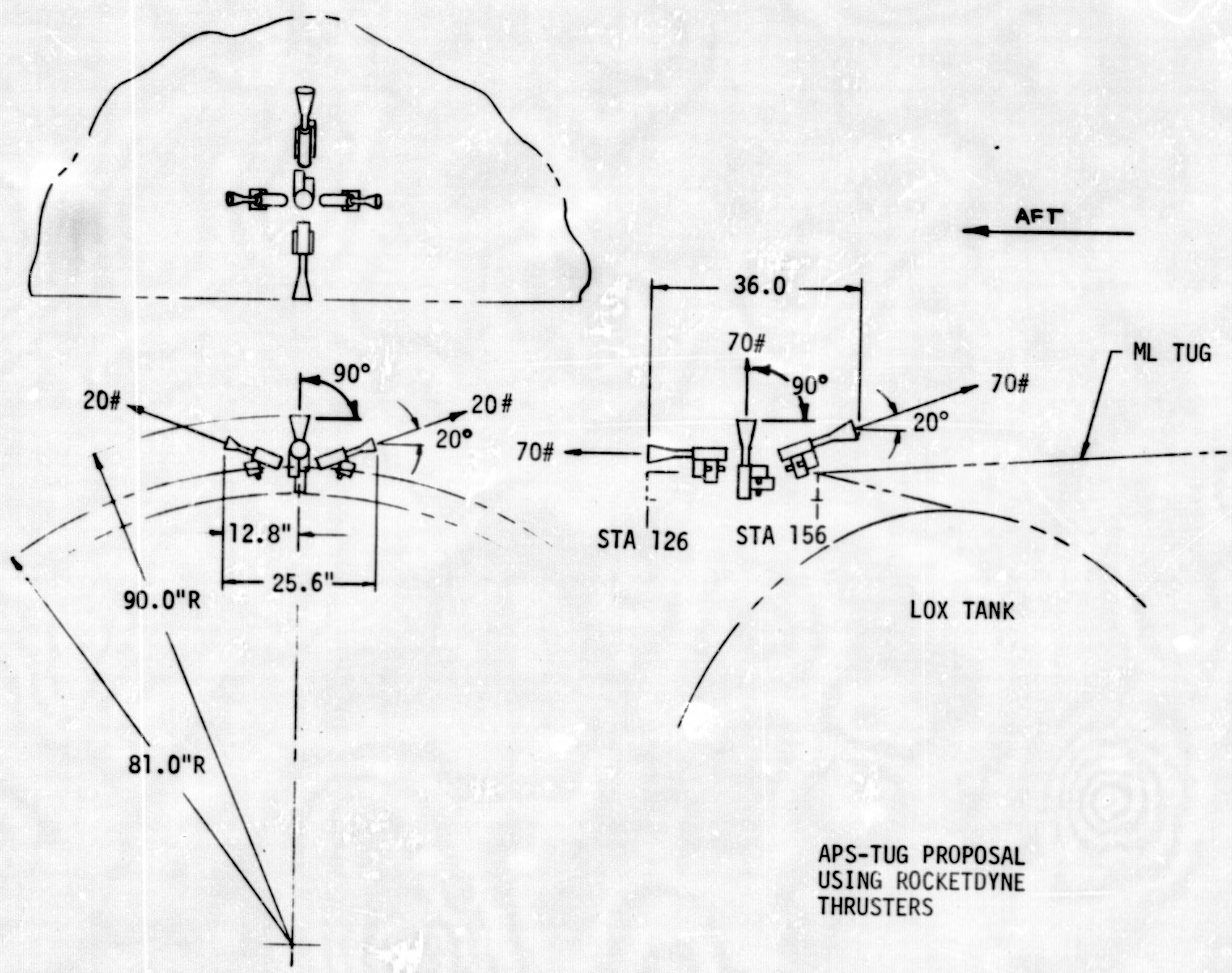


Figure 2.10-10 Thruster Arrangement

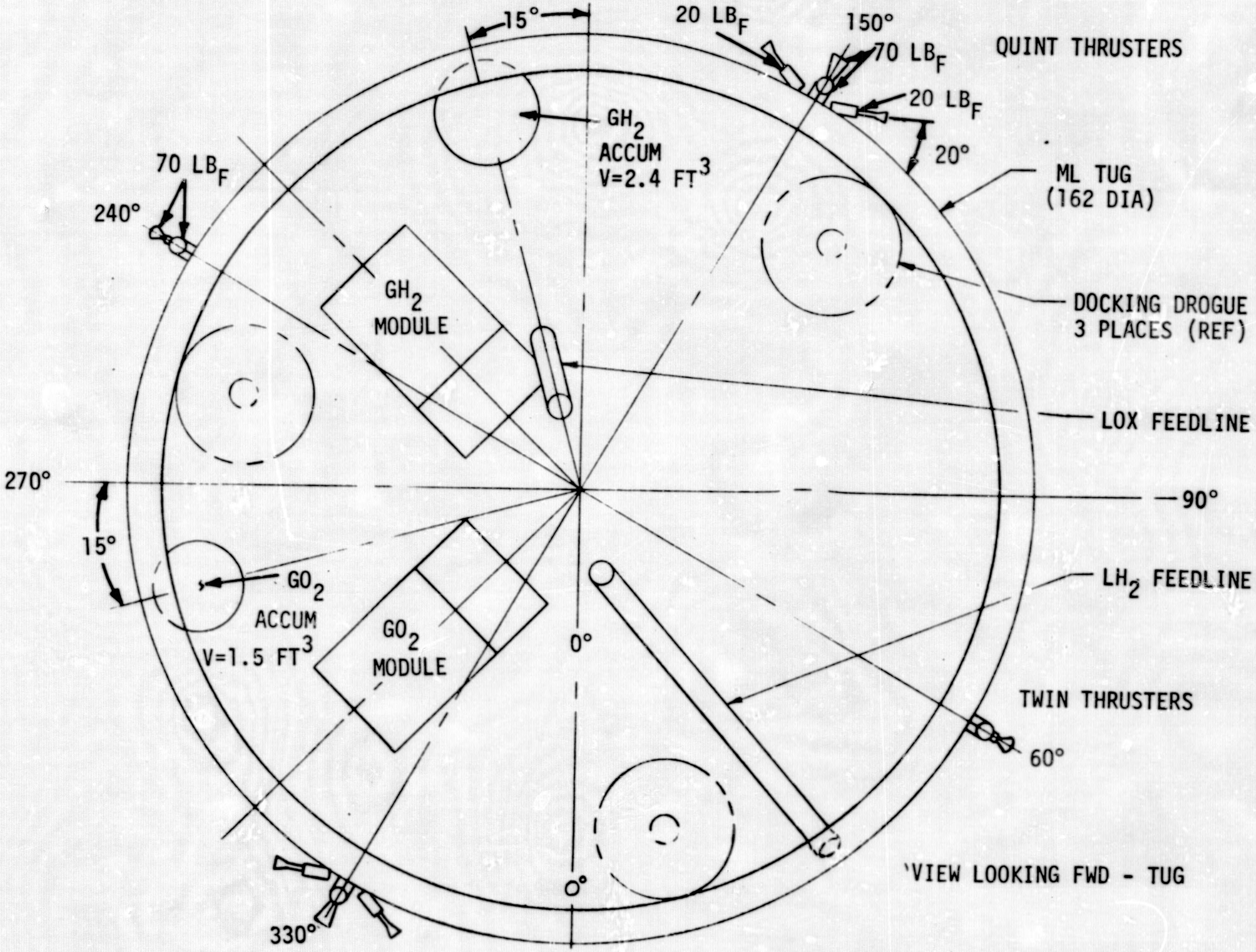


Figure 2.10-11 APS Arrangement (Aft View)

Sketches of the 20 LB_F and 70 LB_F thrusters can be seen in Figures 2.10-12 and 2.10-13. Each thruster consists of an igniter assembly, bi-propellant valve assembly, injector, combustion chamber and nozzle. The nominal design point for the thrusters is a chamber pressure of 250 psia and a mixture ratio of 4:1 with 400°R GOX and 200°R GH_2. The thruster assembly has a life of 10^6 pulses, 10 hours total operation with a maximum continuous operating duration of 1000 seconds.

The 20-pound thruster design employs a copper alloy chamber that is internally cooled (film cooling). The nozzle expansion ratio is 60:1 and the combustor contraction ratio is 7:1 to accommodate the injector pattern. The injector is fabricated of stainless steel with a single concentric tube element in the center and with eight film coolant orifices around the periphery. The concentric tube element also serves as the ignition port. The oxidizer tube provides more than 80 percent of the flow at a mixture ratio of 40:1; GH_2 is injected through an annulus around the tube to reduce the combustion chamber core mixture ratio to 6:0. The electrical spark ignition system is employed to ignite the 40:1 mixture ratio gases in the oxidizer port. GOX is injected around an air gap plug and heated sufficiently to ensure ignition when a small amount of GH_2 is introduced slightly downstream of the plug. The igniter exciter requires 28 VDC, 1 amp to provide sufficient energy for ignition. The valve is a direct actuated, linked bipropellant poppet type with a 1/8-inch seat and requires 28 VDC, 0.7 amp for actuation. The entire assembly including exciter weighs 4.1 pounds and delivers 409 seconds minimum steady-state performance with a 4:1 mixture ratio.

The 70-pound engine is of similar design and material. The injector has the same type concentric tube/ignition post in the center of the injector with an additional four conventional elements located about the control element. A direct actuated, linked bi-propellant valve with a 1/4 inch seat is employed. Energy requirements for igniter and valve are the same as the 20 pounder. The 70-pound assembly weighs 8.0 pounds and delivers 416 seconds minimum performance with a 4:1 mixture ratio.

These engines require no purging. At the engines' exit plane pressure is 0.418 psia, temperature 1590°R, mach number 4.51 and molecular weight 10.08.

Table 2.10-1 is a list of the thruster physical, performance, and operating characteristics. At the 4:1 mixture ratio, the combustion temperature envelope of 5100-5500°R is shown in Figure 2.10-14.

The 70 LB_F thruster design selected in this study requires a maximum pulse width of 25 MSECS. A 50 MSEC pulse width is now available, but Rocketdyne expects a 25 MSEC thruster to be available by 1976. The pulse width is controlled primarily by the bi-propellant valve speed of operation. The present valves are direct acting, electrically operated. To obtain the faster bi-propellant valves, Rocketdyne is certain such a valve would have to be pilot operated for low weight. If a 25 MSEC thruster is not available, more propellant would be required for attitude hold and small attitude control changes due to the larger control band.

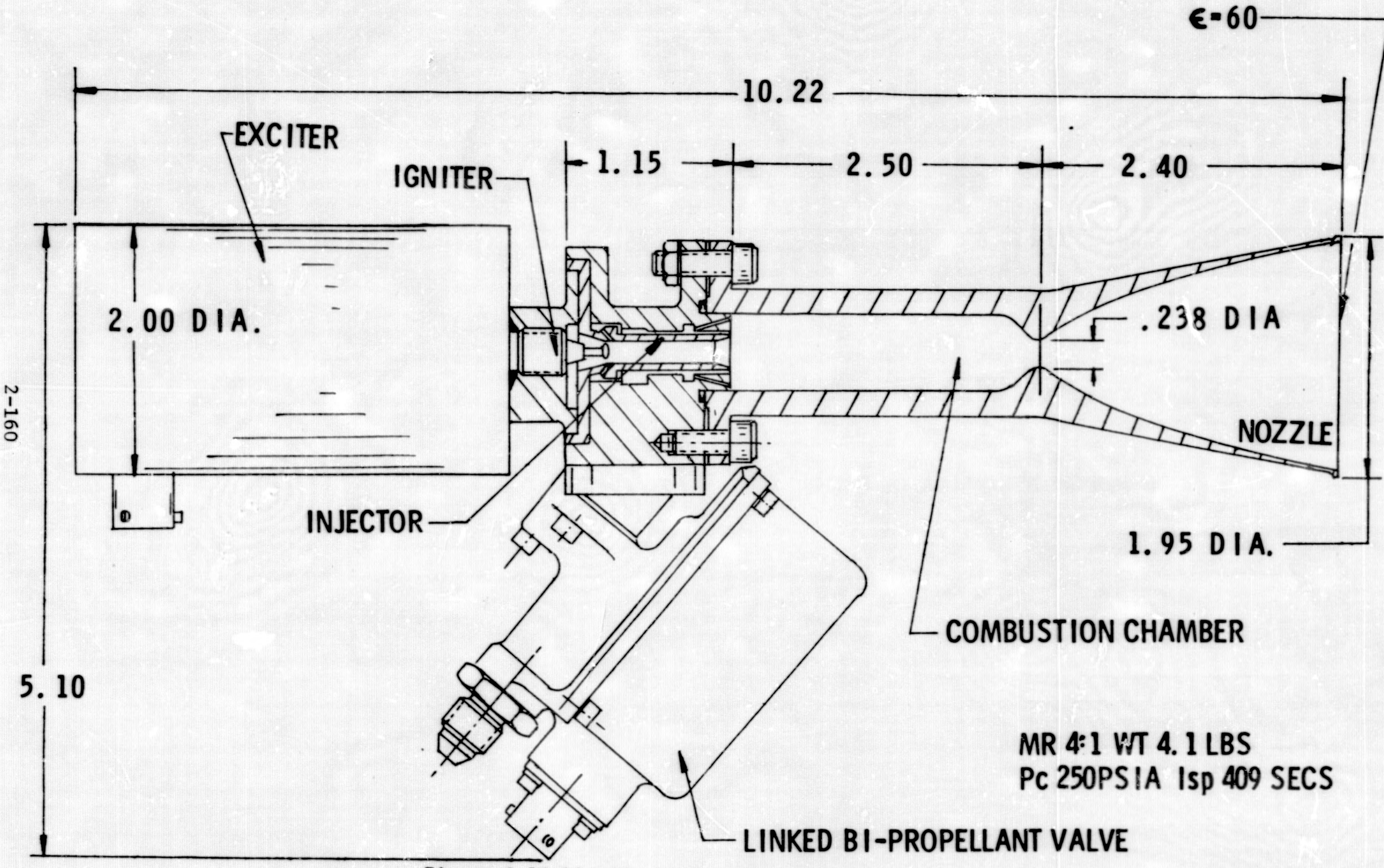


Figure 2.10-12 20 LB_f Thruster Assembly

Space Division
North American Rockwell

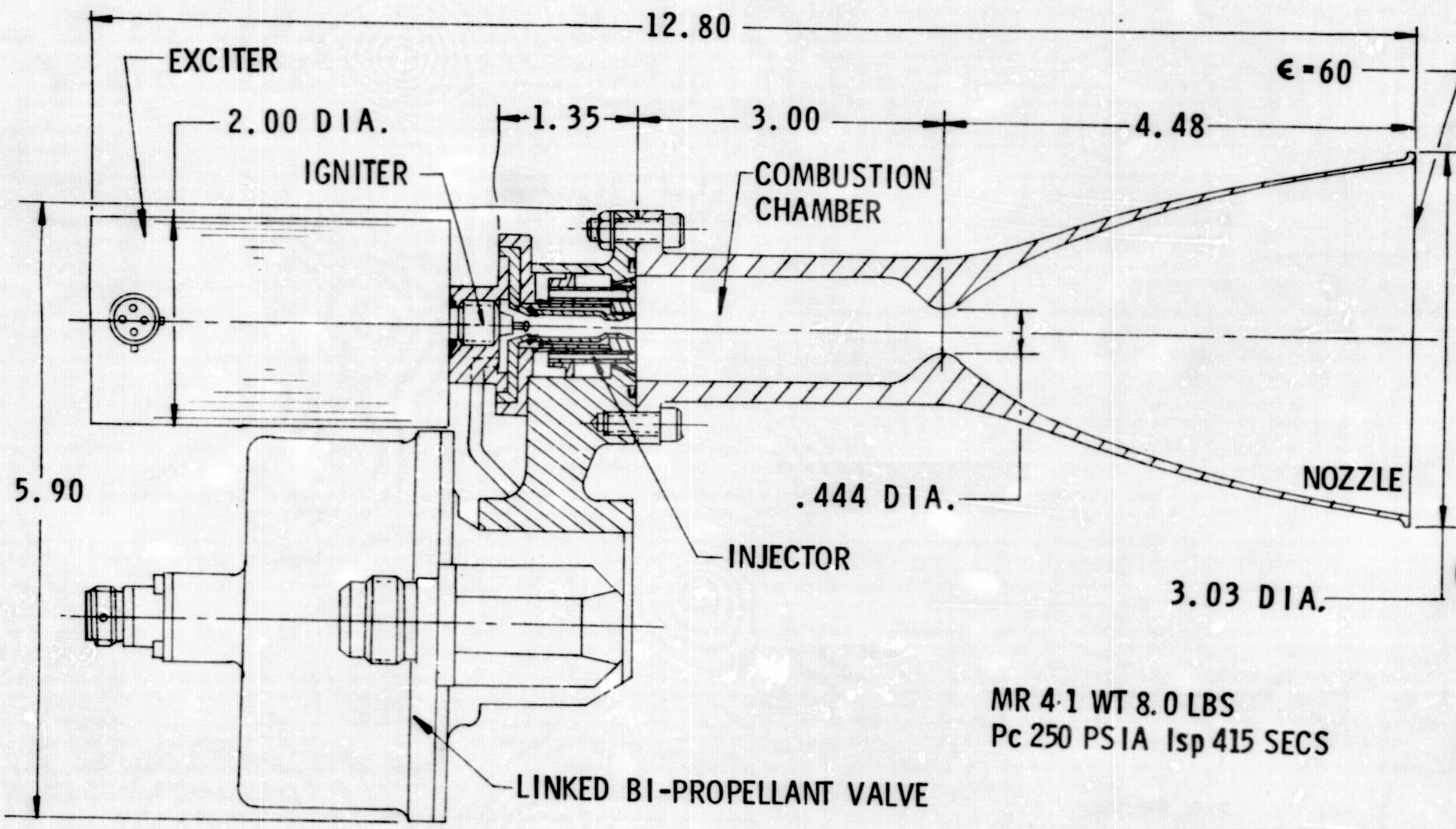


Figure 2.10-13 70 LB_f Thruster Assembly

Space Division
North American Rockwell

Table 2.10-1. Thruster Characteristics

Thrust (lb_f)	20	70
Chamber Pressure (psia)	250	250
Expansion Ratio	60	60
Mixture Ratio, Total	4.0	4.0
Combustor Core	6.0	6.0
Igniter	40.0	40.0
Weight (lb), Total	4.1	8.0
Valve	1.8	4.5
Igniter/Injector	0.8	1.4
Chamber	0.9	1.5
Exciter	0.6	0.6
Core Is_{Theor} (sec)	466	466
ΔIs_{Div}	-3	-3
ΔIs_{Kin}	-14	-13
ΔIs_{Drag}	-13	-12
ΔIs_{Er}	-24	-9
Core Is_{Dlvd}	412	419
$Is_{Film\ Coolant}$	374	374
Thruster Minimum Is (Steady-state; two-zone, zero mixing)	409	416
$\dot{w}_{steady-state}$ (lb/sec) O_2/H_2	0.0382/0.0098	0.1344/0.0336
Valve Response (msec)		
Open	10	35
Close	5	15
Inlet Pressure (psia)	315	320

NOTE: Is = I_{sp}

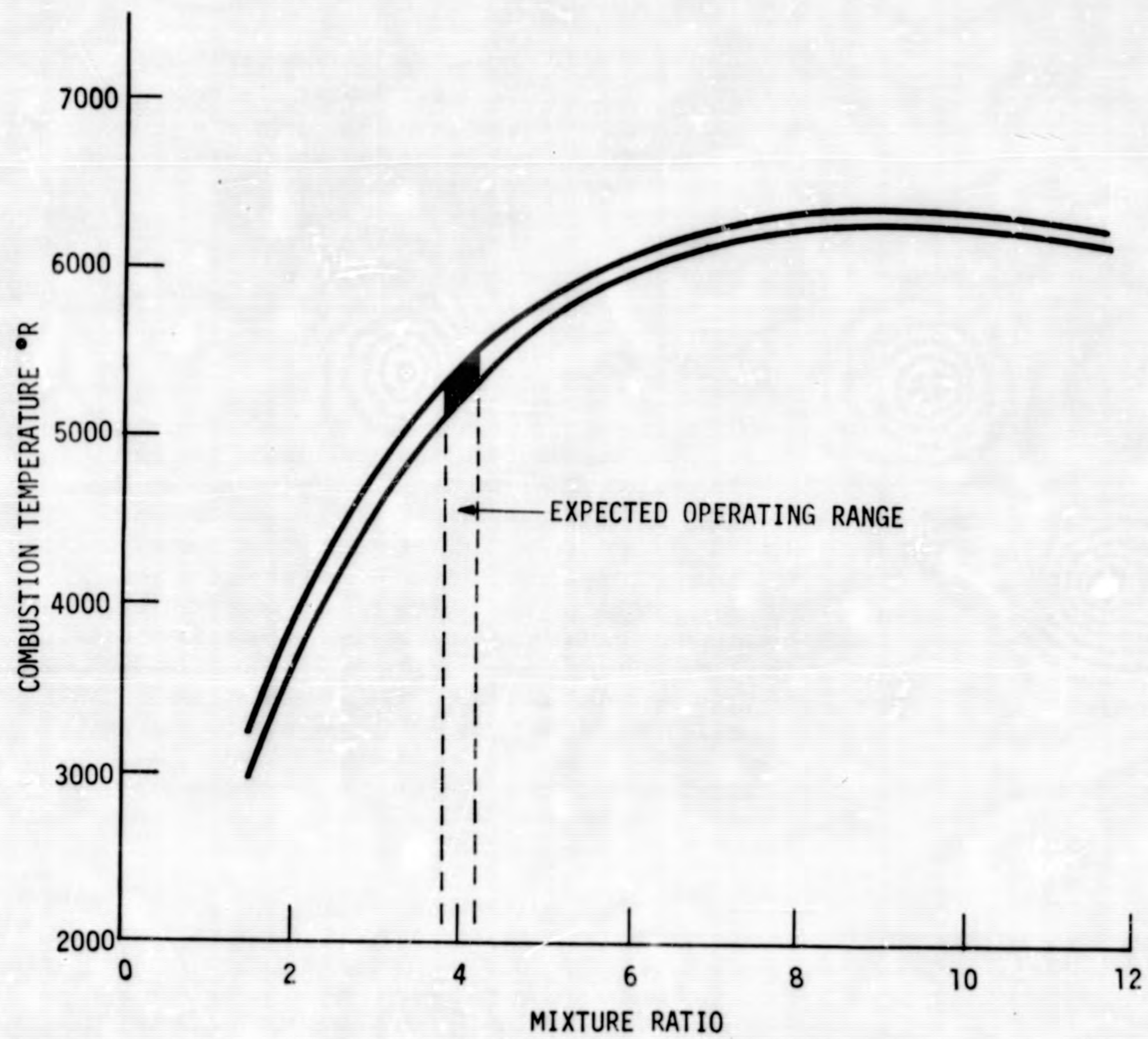


Figure 2.10-14 Thruster Temperature Envelope

The main penalty incurred would be increasing the size of the accumulators to provide one-half hour of attitude hold. For the 25 MSEC pulse width, approximately 90% of the steady state I_{sp} should be realized as shown in Figure 2.10-15.

Turbopumps. The turbopumps are part of the propellant conditioning system. The propellant conditioning systems are located on the thrust cone exterior. Each system is required to withdraw propellants from a propellant tank and to deliver the propellant as gas to the accumulator. The pumps supply the propellants to the heat exchangers at 1250 psia: LOX 0.5 LB per second, LH_2 1.5 LBS per second. The pumps are driven by gas turbines. Gas generators burn GOX and GH_2 which drives the turbines and exhausts at 900°R. This study considered three pump configurations. The third configuration was selected, although heavier than the others, because it did not require any NPSP and to avoid the complications of developing and installing an electric boost pump in each propellant tank.

Data for the three configurations are presented. The first system configuration employs a very small high-speed turbopump/gas generator assembly and operates with 4 psi oxygen system NPSP and 2 psi hydrogen system NPSP. The second system utilizes the same basic turbopump/gas generator assembly as Configuration One plus an electrically driven boost pump which operates with "0" NPSP in both oxygen and hydrogen systems. The third system employs a relatively slow-speed turbopump/gas generator assembly and operated with "0" NPSP in both oxygen and hydrogen systems without the use of separate boost pumps. These configurations are shown schematically in Figure 2.10-16. Data are presented for each of the components for both the oxygen and hydrogen systems. The hydrogen and oxygen pumps require that they are in the wet condition prior to start. Although the pumps in Configurations #2 and #3 are capable of ingesting vapor in excess of 25 percent by volume, the bearings and seals must be chilled prior to start to assure that the life requirements are met.

Helium at 50 psia is required to actuation of the LOX turbopump liftoff seals and for purge of the intermediate seals separating the pump from the turbine. A maximum purge leakage rate of 0.0002 LB per sec through the latter seals will be required during oxidizer pump operation plus 5 seconds additional. Helium for the seals will be obtained from the main engine helium supply. The flow will be controlled by a two-way solenoid valve and orifice.

Turbopump Configuration #1. The preliminary design of the LOX turbopump (inducer plus one centrifugal impeller stage) was based on using the maximum available NPSP (2 psi). The pump design parameters are listed in Table 2.10-2. The maximum speed permitted for the NPSP was 88,200 rpm, which resulted in an impeller diameter of 1.05 inches. The LH_2 pump has an inducer plus two centrifugal impeller stages. Pump speed was obtained by using the same impeller as the LOX pump. LH_2 and LOX turbine design parameters are listed in Table 2.10-3. The LH_2 turbine is a velocity-compounded, two-row design while the LOX turbine is a partial admission impulse, one-row design. Envelope sketches for the LH_2 and LOX turbopumps are shown in Figures 2.10-17 and 2.10-18, respectively, and their corresponding weights are 3 and 4 pounds.

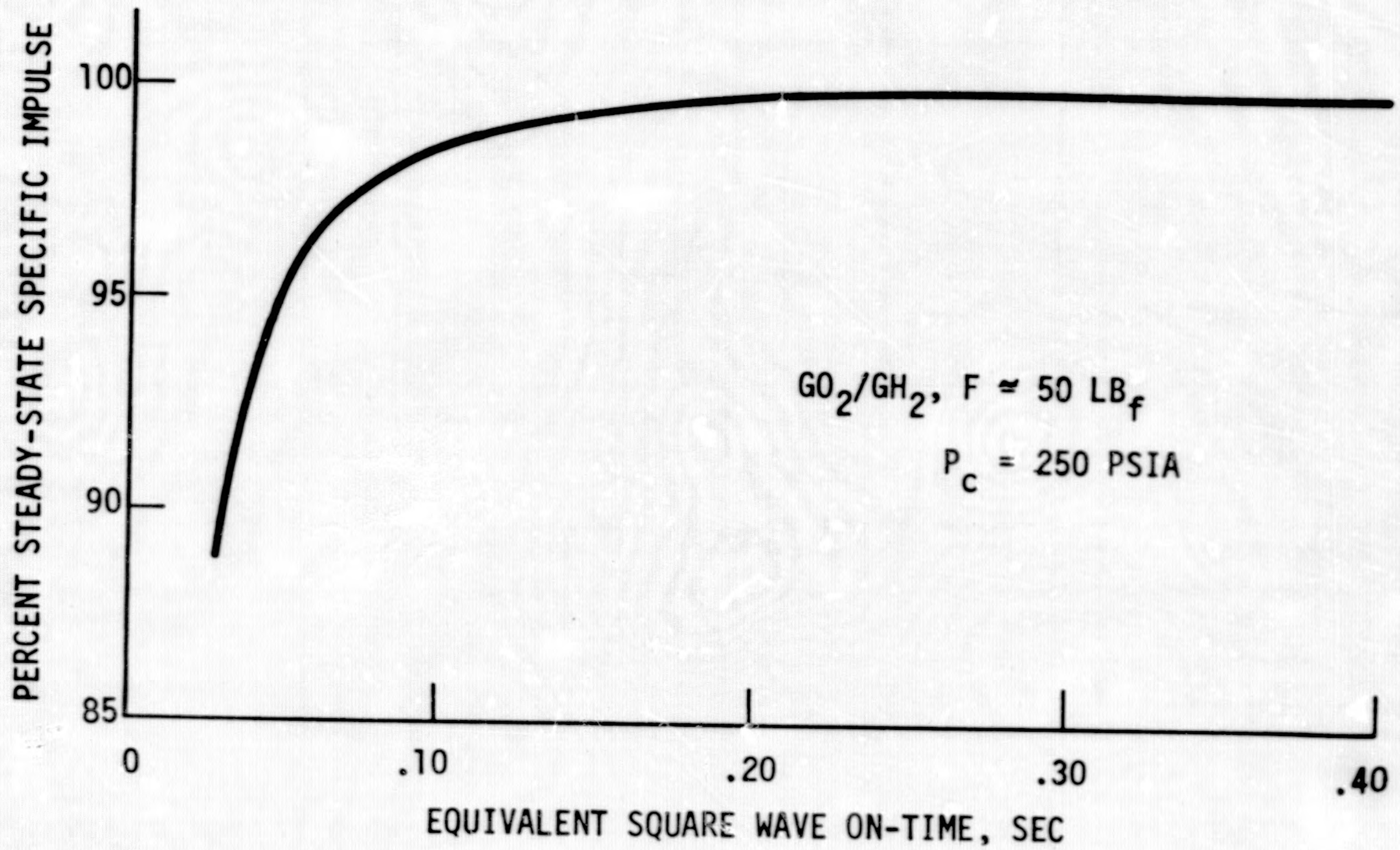


Figure 2.10-15 Thruster Pulse Performance

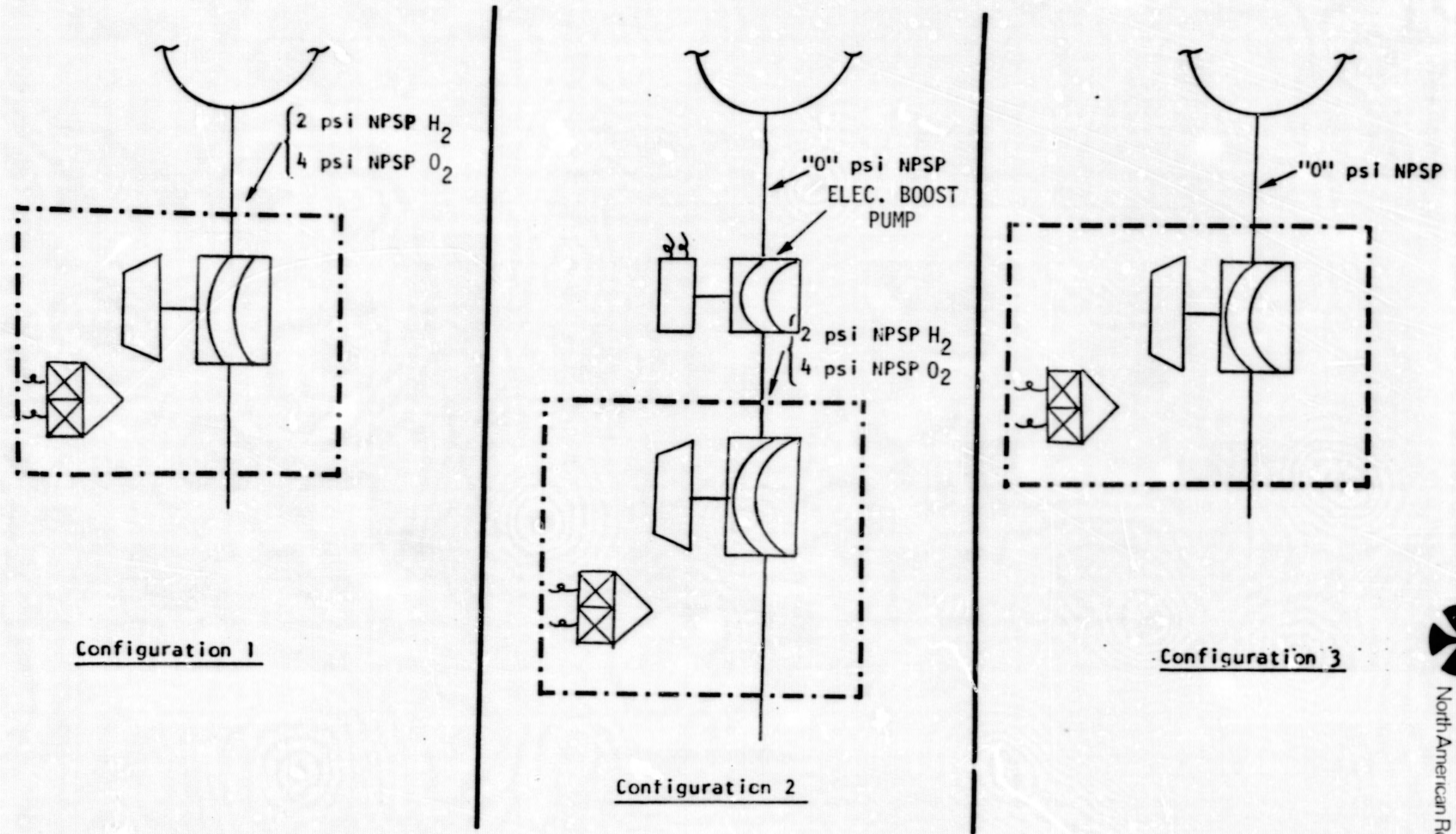


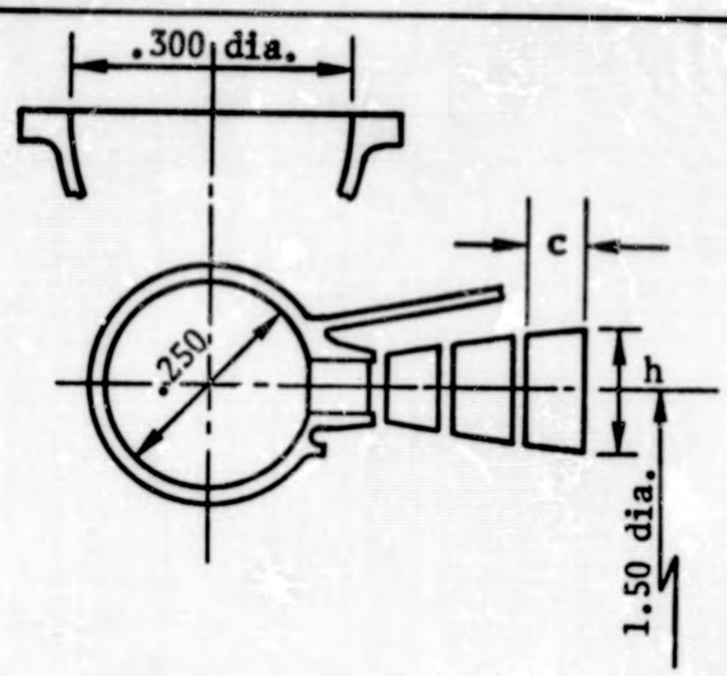
Figure 2.10-16 Alternate Turbopump Configurations

Table 2.10-2. Pump Design Parameters
(Configurations #1 and #2)

Pump Parameters	Main Pumps		Boost Pumps	
	Fuel	Oxidizer	Fuel	Oxidizer
Number of Stages	2+1	1+1	1	1
Type	Cent. + Inducer	Cent. + Inducer	Inducer	Inducer
Weight Flow, ($\dot{W}$), lb/sec	0.5	1.5	0.5	1.5
Volume Flow, (Q), gpm	50.8	9.46	50.8	9.46
Delta Pressure, (ΔP), psid	1,250.0	1,250.0	4.0	8.0
Delta Head, (ΔH), ft	40,800.0	2,530.0	130.0	16.2
Speed, (N), rpm	250,000.0	88,200.0	15,200.0	6,800.0
Horsepower, HP	58.4	13.55	0.20	0.074
Efficiency, (η), percent	63.5	51.0	59.0	60.0
Critical NPSH, feet	65.3	8.1	0	0
Max. Suction Specific Speed, (S_s)	95,000.0	56,500.0	-	-
Stage Specific Speed, (N_s)	1,043.0	760.0	2,810.0	2,600.0
Stage Head Coefficient, (Ψ_{st})	0.57	0.57	0.25	0.25
Inlet Flow Coefficient, (ϕ)	0.08	0.08	0.07	0.07
Bearing Size, Inches	3/16	3/16	3/16	3/16
Bearing DN x 10^{-6}, mm rpm	1.19	0.42	0.07	0.032
Inlet Tip Diameter, (DT_1), inches	0.638	0.518	1.0	1.0
Inlet Tip Speed, (U_{T_1}), fps	696.0	200.0	66.4	29.6
Exit Tip Diameter, (DT_2), inches	1.05	1.05	0.9	0.9
Exit Tip Speed, (U_{T_2}), fps	1,620.0	405.0	59.6	26.6
Turbopump Weight, lb	3.0	4.0	2.4	2.2

Table 2.10-3. Turbine Design Parameters
(Configurations #1 and #2)

LH_2 TURBINE:

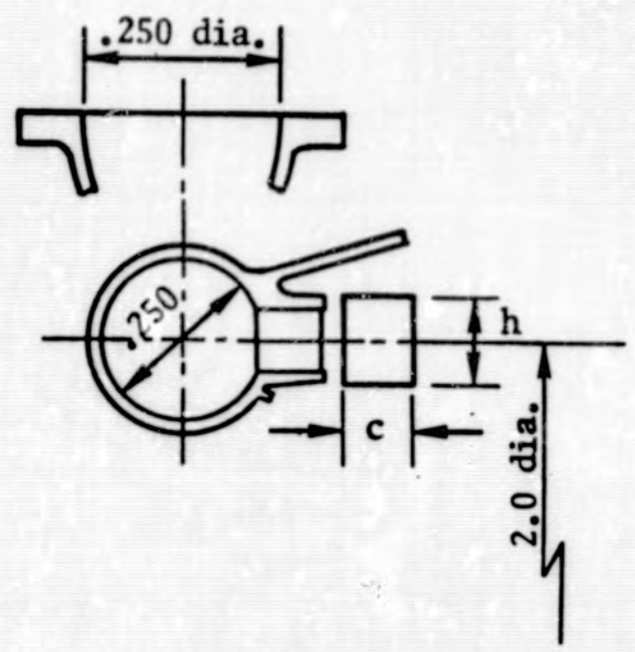


Type: 2-Row Velocity Compounded

Inlet Temp., °R	T_{1t}:	1960
Inlet Press., PSIA	P_{1t}:	270
Exhaust Press., PSIA	P_{2s}:	13.5
Press. Ratio	PR:	20
Speed, RPM	N:	250,000
Blade Speed, f/s	um:	1637
Velocity Ratio	u/c_o:	.162
Efficiency, %	η_{TS}:	57.5
Flowrate, lb/s	w:	.036
Horsepower	HP:	60
Admission, %	ε:	100
Stress, x 10^9	N^2AA:	59.0

	c	h
Nozzle:	.300	.150
1st Rot.:	.200	.160
Stator:	.220	.180
2nd Rot.:	.240	.200

LOX TURBINE:



Type: Partial-Admission Impulse

Inlet Temp., °R	T_{1t}:	1960
Inlet Press., PSIA	P_{1t}:	270
Exhaust Press., PSIA	P_{2s}:	13.5
Press. Ratio	PR:	20
Speed, RPM	N:	88,000
Blade Speed, f/s	um:	770
Velocity Ratio	u/c_o:	.076
Efficiency, %	η_{TS}:	19
Flowrate, lb/s	w:	.027
Horsepower	HP:	15
Admission, %	ε:	25
Stress, x 10^9	N^2AA:	14.6

	c	h
Nozzle:	.300	.177
Rot.:	.220	.294

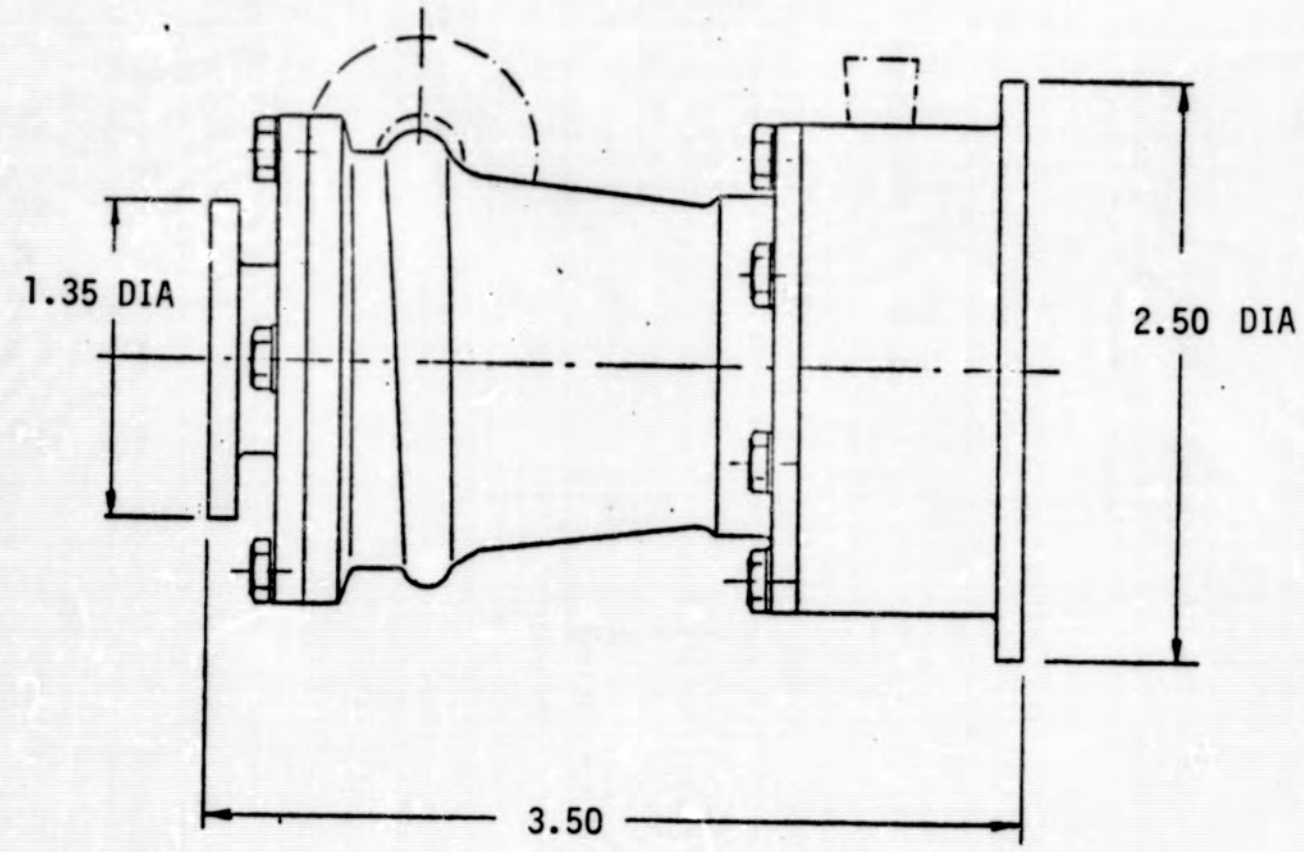


Figure 2.10-17 Configuration #1 Hydrogen Pump

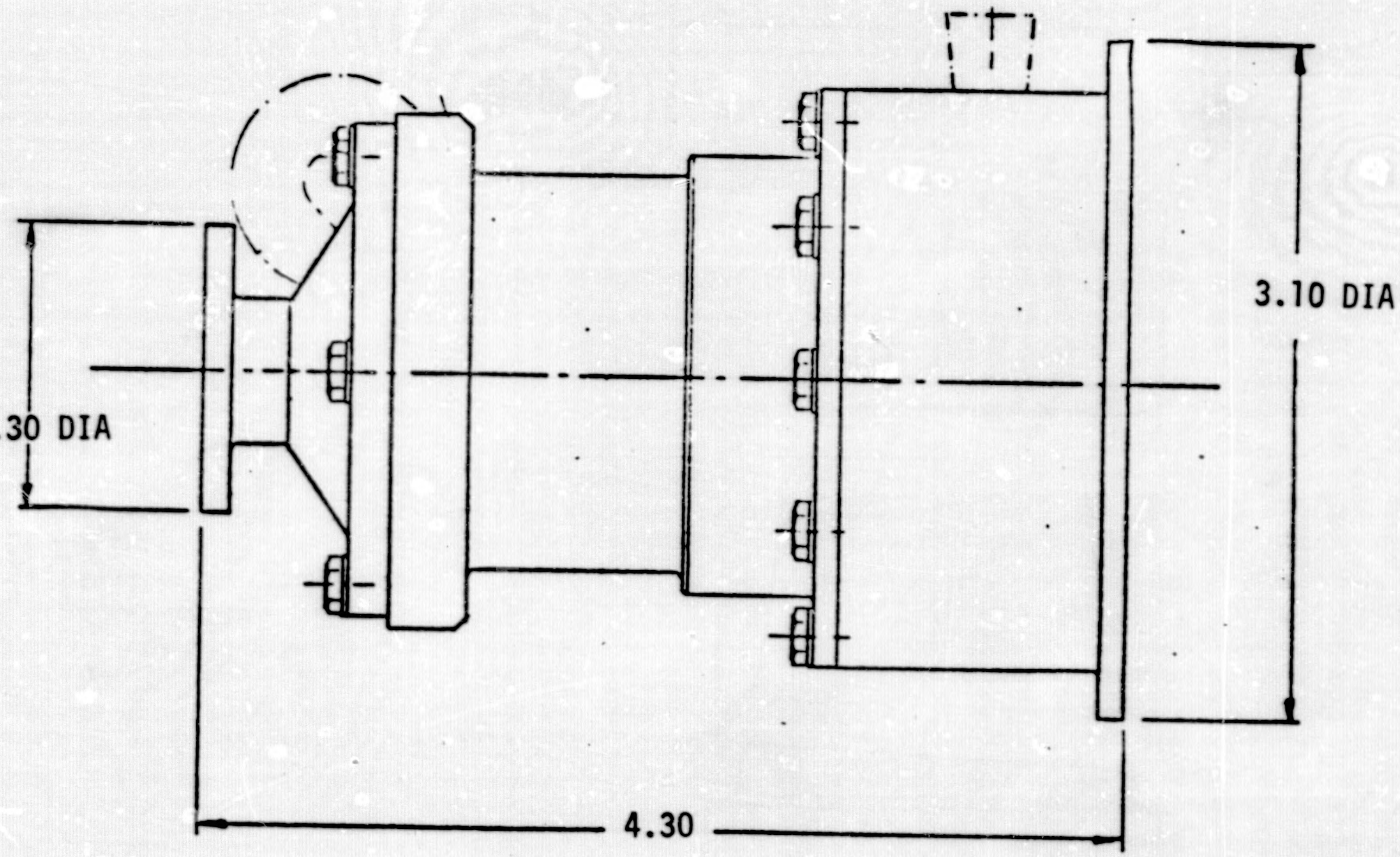


Figure 2.10-18 Configuration #1 Oxygen Pump

Turbopump Configuration #2. The LH_2 and LOX boost pumps (inducers) were designed for zero NPSP operation with vapor volume fraction pumping capability of at least 25 percent. The pumps use identical inducers that are 1 inch in diameter. Boost pump parameters are listed in Table 2.10-2. Power for both of these pumps is supplied by electrical motors. Figures 2.10-19 and 2.10-20 show sketches of the LH_2 and LOX boost pumps and drive envelopes. The estimated weights are 2.4 and 2.2 pounds for the LH_2 and LOX boost pumps, respectively.

Turbopump Configuration #3. The preliminary designs for both turbopumps were based on zero NPSP operational capability without boost pumps. The operating speeds obtained were lower than the Configuration #1 designs, and therefore required additional pumping stages. The LH_2 pump design has three centrifugal impellers plus an inducer, while the LOX pump design has two centrifugal impellers plus an inducer. Design parameters for these pumps are listed in Table 2.10-4. A two-row velocity-compounded turbine drives the LH_2 pump, while a partial admission impulse turbine drives the LOX pump. Design parameters for the turbines are listed in Table 2.10-5. Envelope sketches for the LH_2 and LOX turbopumps are shown in Figures 2.10-21 and 2.10-22, respectively. The estimated weights are 8.2 and 15 pounds for the LH_2 and LOX turbopumps, respectively.

Gas Generators. The gas generator total flow rate was calculated based upon pump horsepower requirements and turbine efficiencies. The gas generator mixture ratio was based upon the required turbine inlet temperature and the temperature of the combustants: GOX 400°R, GH_2 200°R. The design parameters for the four different gas generator configurations are given in Tables 2.10-6 through 2.10-9.

The preliminary design(s) of the four gas generators (two hydrogen sizes and two oxygen sizes) are shown in Figures 2.10-23 through 2.10-26. The design concept is based upon the results of experiments conducted by Rocketdyne. An electrical spark igniter system is used. The oxidizer is injected between the spark plug electrodes and then flows into the combustion chamber. A small portion of the fuel is injected at the head end of the combustor (into the oxidizer) where ignition is initiated and combustion sustained at a mixture ratio of about 40:1. The remainder of the hydrogen is used to cook the gas generator/combustion body and is injected into the primary combustion gases through secondary hydrogen injection orifices. Hot-firing tests with this configuration have demonstrated reliable ignition, good performance, and satisfactory exhaust temperature profile. The gas generator is welded to the turbine manifold in any manner convenient for the installation. No purges are required.

Heat Exchangers. The turbopumps supply 1250 psia LH_2 and LOX to the heat exchangers for conversion to gas: GH_2 at 200°R and GOX at 400°R. The heat source is the products of combustion, GOX and GH_2 are burned, from the integral gas generator assembly. The gases are burned at a 3:1 mixture ratio and enter the heat exchanger at approximately 4500°R. The gases make a single pass through the heat exchanger, Figure 2.10-27, and exit through an annulus at 900°R. The liquid propellants enter in two paths so the hot gases of combustion

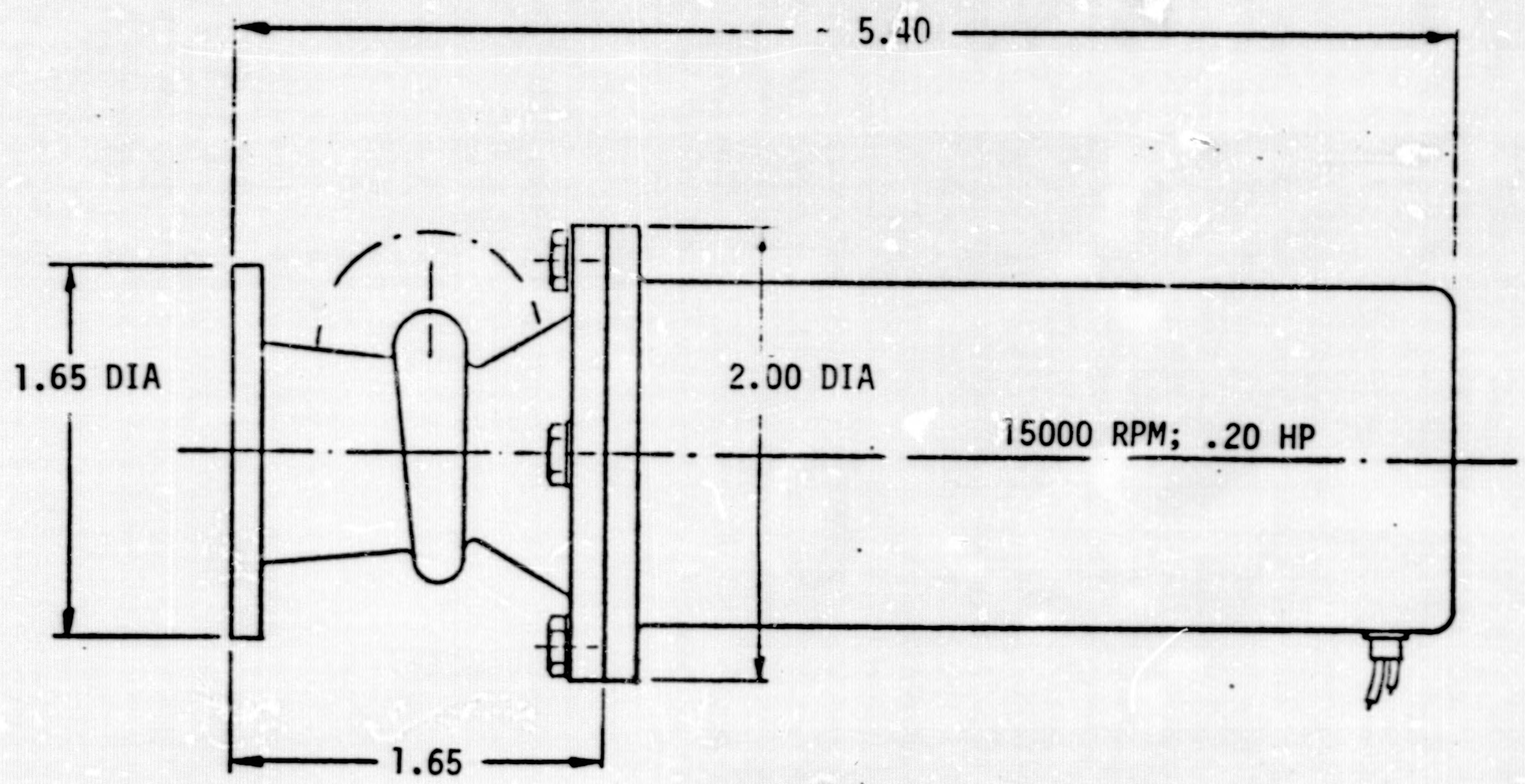


Figure 2.10-19 LH_2 Boost Pump

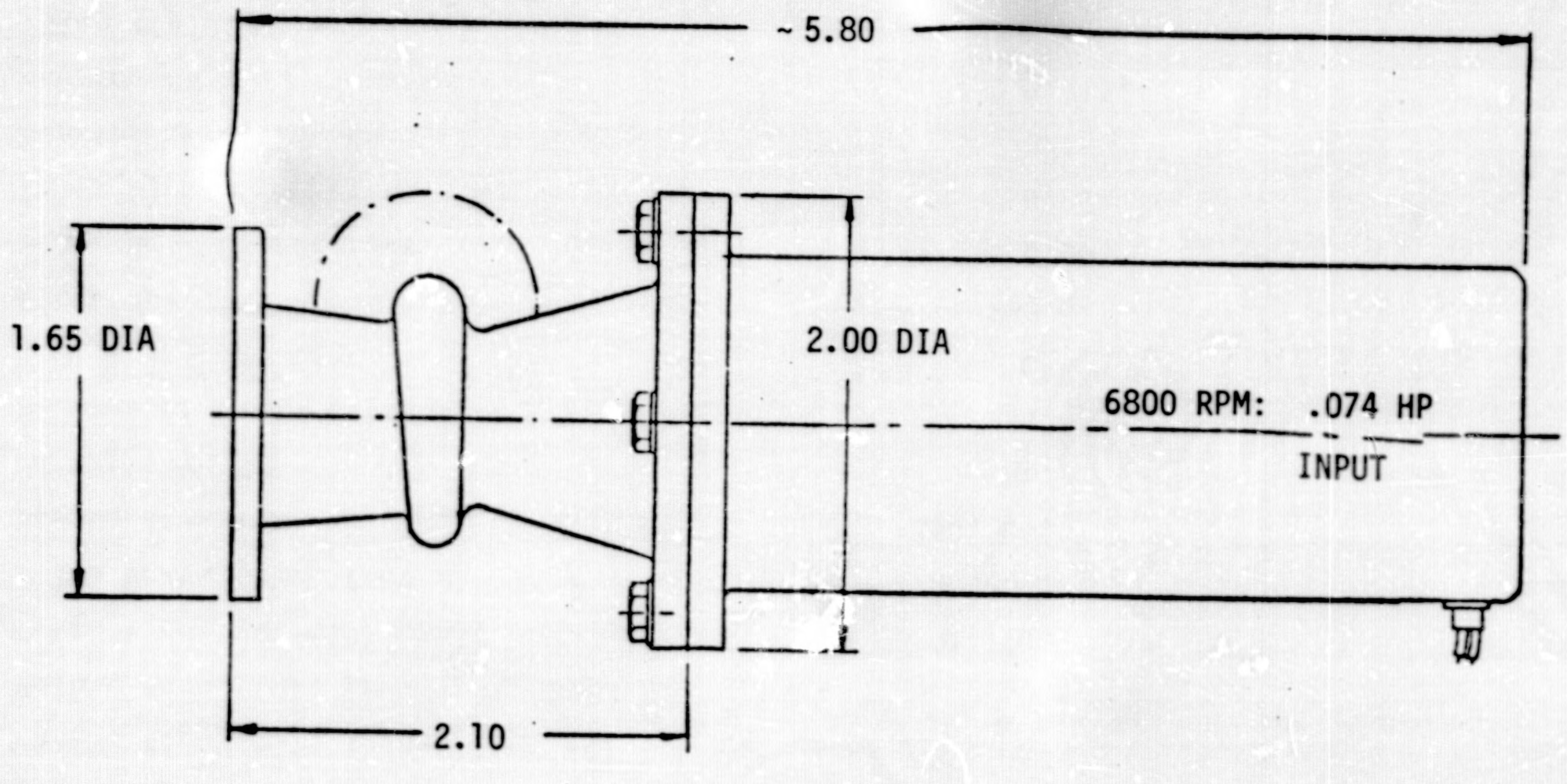


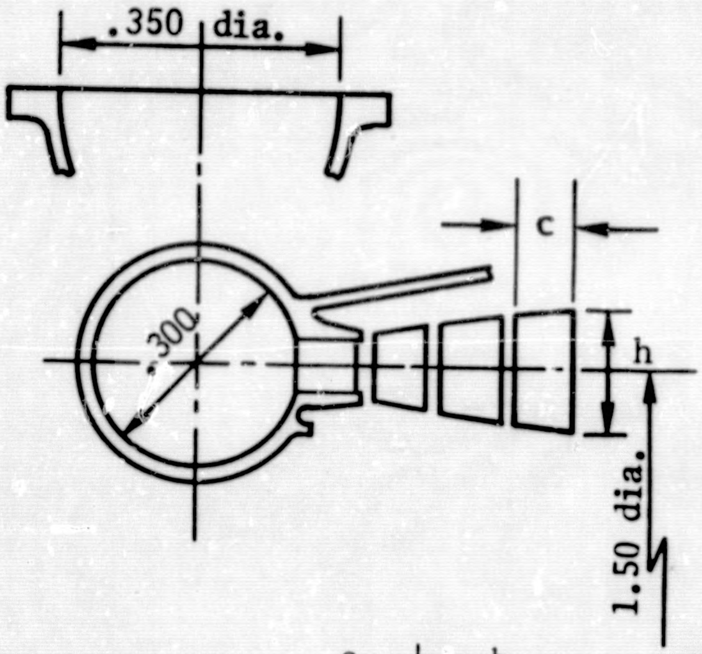
Figure 2.10-20 LOX Boost Pump

Table 2.10-4. Configuration #3 Pump Design Parameters

Pump Parameter	Main Pumps	
	Fuel (Hydrogen)	Oxidizer (Oxygen)
Number of Stages	3+1	2+1
Type	Centrifugal + Inducer	Centrifugal + Inducer
Weight Flow, ($\dot{W}$), lb/sec	0.5	1.5
Volume Flow, (Q), gpm	50.8	9.46
Delta Pressure, (ΔP), psid	1,250.0	1,250.0
Delta Head, (ΔH), feet	40,800.0	2,530.0
Speed, (N), rpm	150,500.0	28,000.0
Horsepower, HP	66.2	13.26
Efficiency, (η), percent	56.0	52.0
Critical NPSH, feet	0	0
Stage Specific Speed, (N_S)	788.0	681.0
Stage Head Coefficient, (Ψ)	0.56	0.56
Inlet Flow Coefficient, (ϕ)	0.07	0.07
Bearing Size, inches	3/16	3/16
Bearing DN x 10^{-6}, mm DN	0.715	0.133
Inlet Tip Diameter, (D_{T_1}), inches	1.0	1.0
Inlet Tip Speed, (U_{T_1}), fps	660.0	122.0
Exit Tip Diameter, (D_{T_2}), inches	1.35	2.21
Exit Tip Speed, (U_{T_2}), fps	886.0	270.0
Turbopump Weight, lb	8.2	15.0

Table 2.10-5. Configuration #3 Turbine Design Parameters

LH_2 TURBINE:

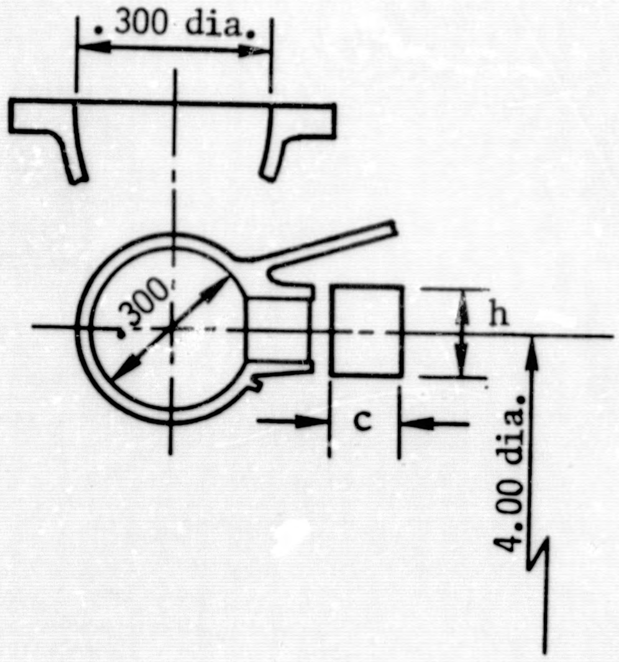


Type: 2-Row Velocity Compounded

Inlet Temp., °R	T_{1t}:	1960
Inlet Press., PSIA	P_{1t}:	270
Exhaust Press., PSIA	P_{2s}:	13.5
Press. Ratio	PR:	20
Speed, RPM	N:	150,000
Blade Speed, f/s	um:	982
Velocity Ratio	u/c_o:	.097
Efficiency, %	η_{TS}:	40.1
Flowrate, lb/s	$\dot{w}$:	.037
Horsepower:	HP:	66
Admission, %	ϵ:	100
Stress, x 10^9	N^2AA:	26.5

	c	h
Nozzle:	.300	.150
1st Rot.:	.200	.180
Stator:	.220	.220
2nd Rot.:	.250	.250

LOX TURBINE:



Type: Partial-Admission Impulse

Inlet Temp., °R	T_{1t}:	1960
Inlet Press., PSIA	P_{1t}:	270
Exhaust Press., PSIA	P_{2s}:	13.5
Press. Ratio	PR:	20
Speed, RPM	N:	28,000
Blade Speed, f/s	um:	489
Velocity Ratio	u/c_o:	.050
Efficiency, %	η_{TS}:	14.2
Flowrate, lb/s	$\dot{w}$:	.037
Horsepower	HP:	13.2
Admission, %	ϵ:	20
Stress, x 10^9	N^2AA:	2.5

	c	h
Nozzle:	.300	.150
Rot.:	.200	.250

2-176

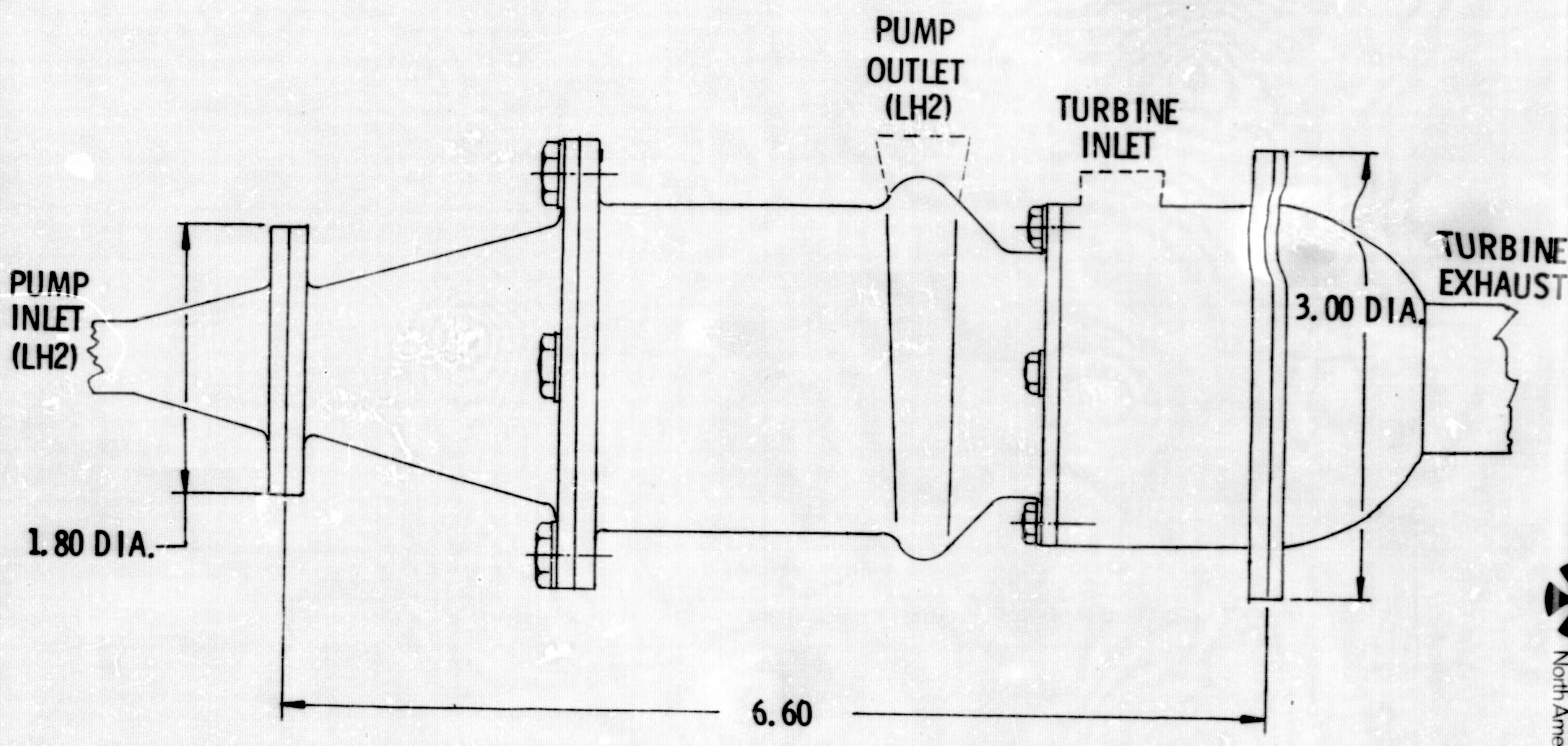


Figure 2.10-21 Config. #3 LH_2 Turbopump (Wt 8.2 Lbs)

Space Division
North American Rockwell

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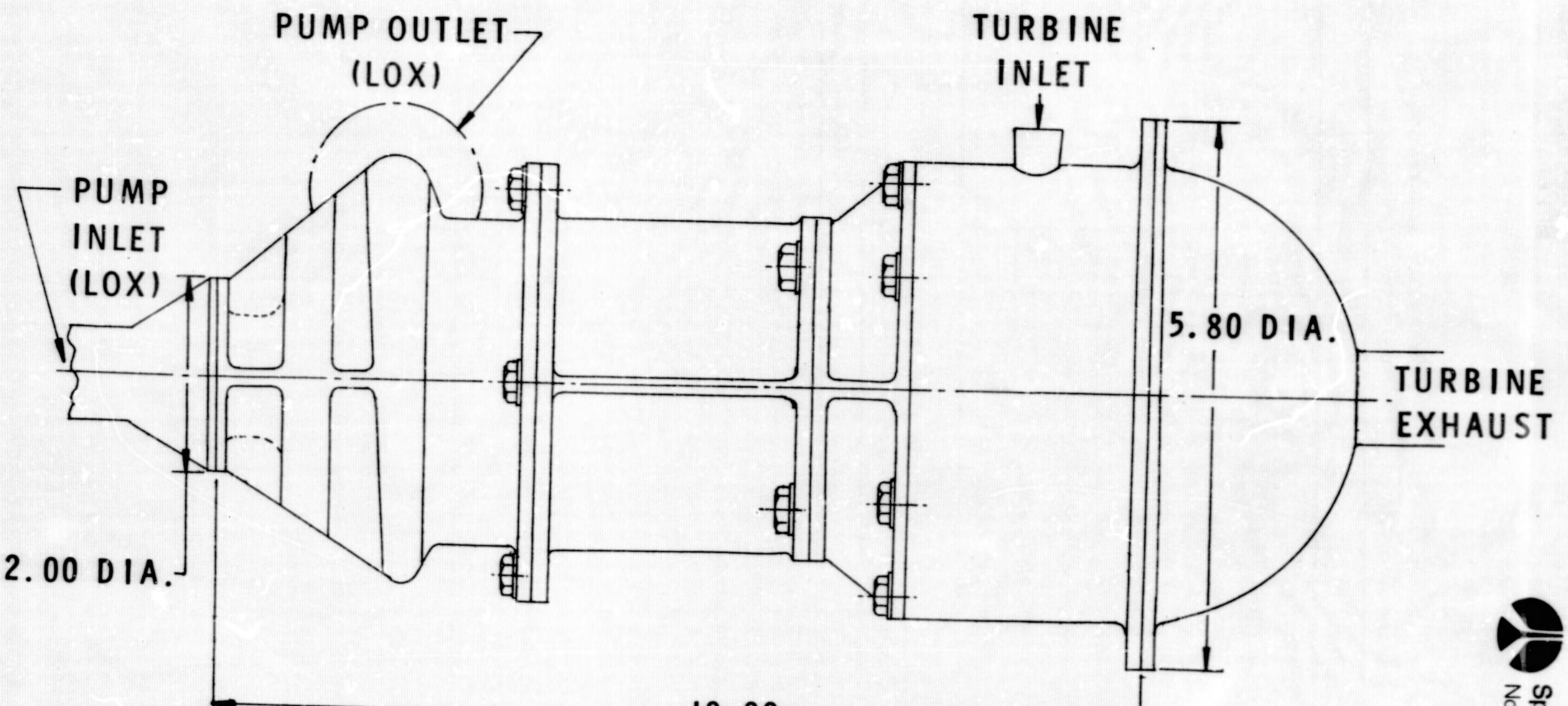


Figure 2.10-22 Config. #3 LOX Turbopump (Wt 15 Lbs)

Space Division
North American Rockwell

Table 2.10-6. Gas Generator Design Parameters
(Configuration No. 1 and 2 Oxidizer System)

Propellants	GH_2/GO_2
Mixture Ratio, o/f	0.97
Flowrate (total), lb/sec	0.027
Inlet Temperature, R	
GH_2	200
GO_2	400
Inlet Pressure, psia (to the valve)	375
Chamber Pressure, psia	270
Exhaust Temperature, R	1960
Weight (GG & Valves), lb	3

Table 2.10-7. Gas Generator Design Parameters
(Configuration No. 1 and 2 Hydrogen System)

Propellants	GH_2/GO_2
Mixture Ratio, o/f	0.97
Flowrate (total), lb/sec	0.036
Inlet Temperature, R	
GH_2	200
GO_2	400
Inlet Pressure (to the valve), psia	375
Chamber Pressure, psia	270
Exhaust Temperature, R	1960
Weight (GG & Valves), lb	3

Table 2.10-8. Gas Generator Design Parameters
(Configuration No. 3 Oxygen System)

Propellants	GH_2/GO_2
Mixture Ratio, o/f	0.97
Flowrate (total), lb/sec	0.037
Inlet Temperature, R	
GH_2	200
GO_2	400
Inlet Pressure (to the valve), psia	375
Chamber Pressure, psia	270
Exhaust Temperature, R	1960
Weight (GG & Valves), lb	3

Table 2.10-9. Gas Generator Design Parameters
(Configuration No. 3 Hydrogen System)

Propellants	GH_2/GO_2
Mixture Ratio, o/f	0.97
Flowrate (total), lb/sec	0.057
Inlet Temperature, R	
GH_2	200
GO_2	400
Inlet Pressure (to the valve), psia	375
Chamber Pressure, psia	270
Exhaust Temperature, R	1960
Weight (GG & Valves), lb	3
Valve Power, watts	
Pull	13
Hold	1.5
Igniter Power, watts	2.5
volts	28

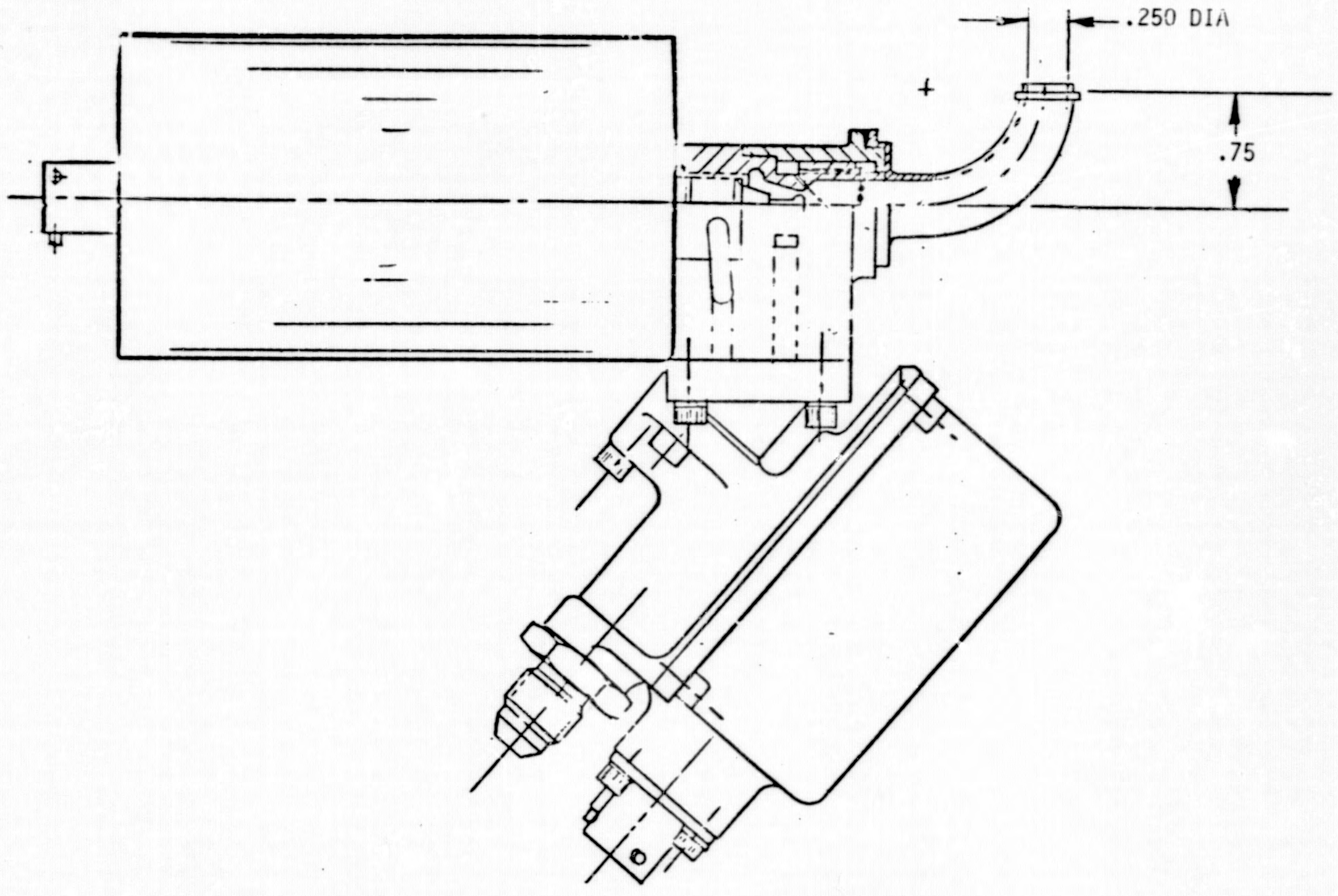


Figure 2.10-23 Configurations #1 and #2. Oxidizer System Gas Generator

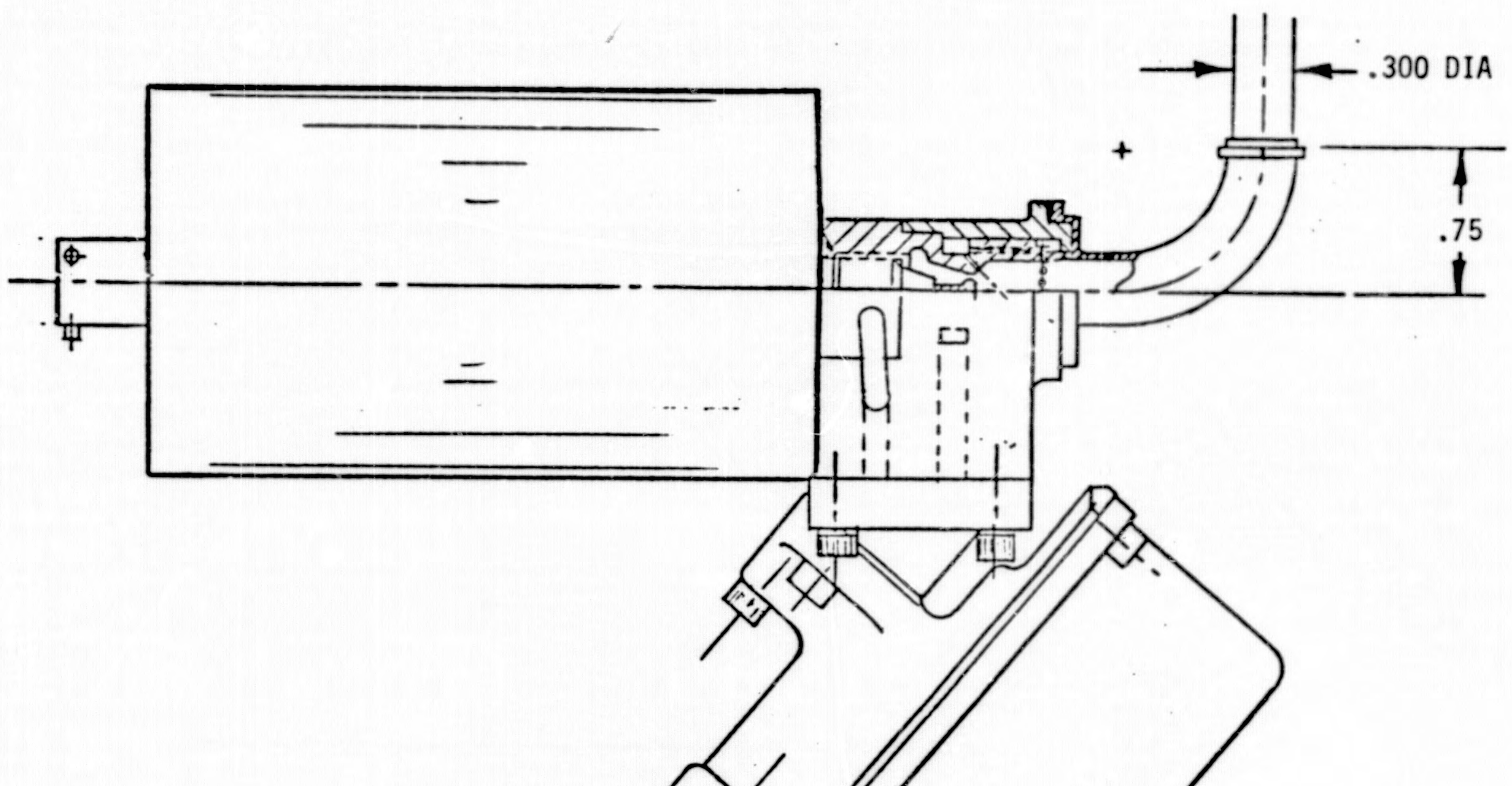


Figure 2.10-24 Configurations #1 and #2. Hydrogen System Gas Generator

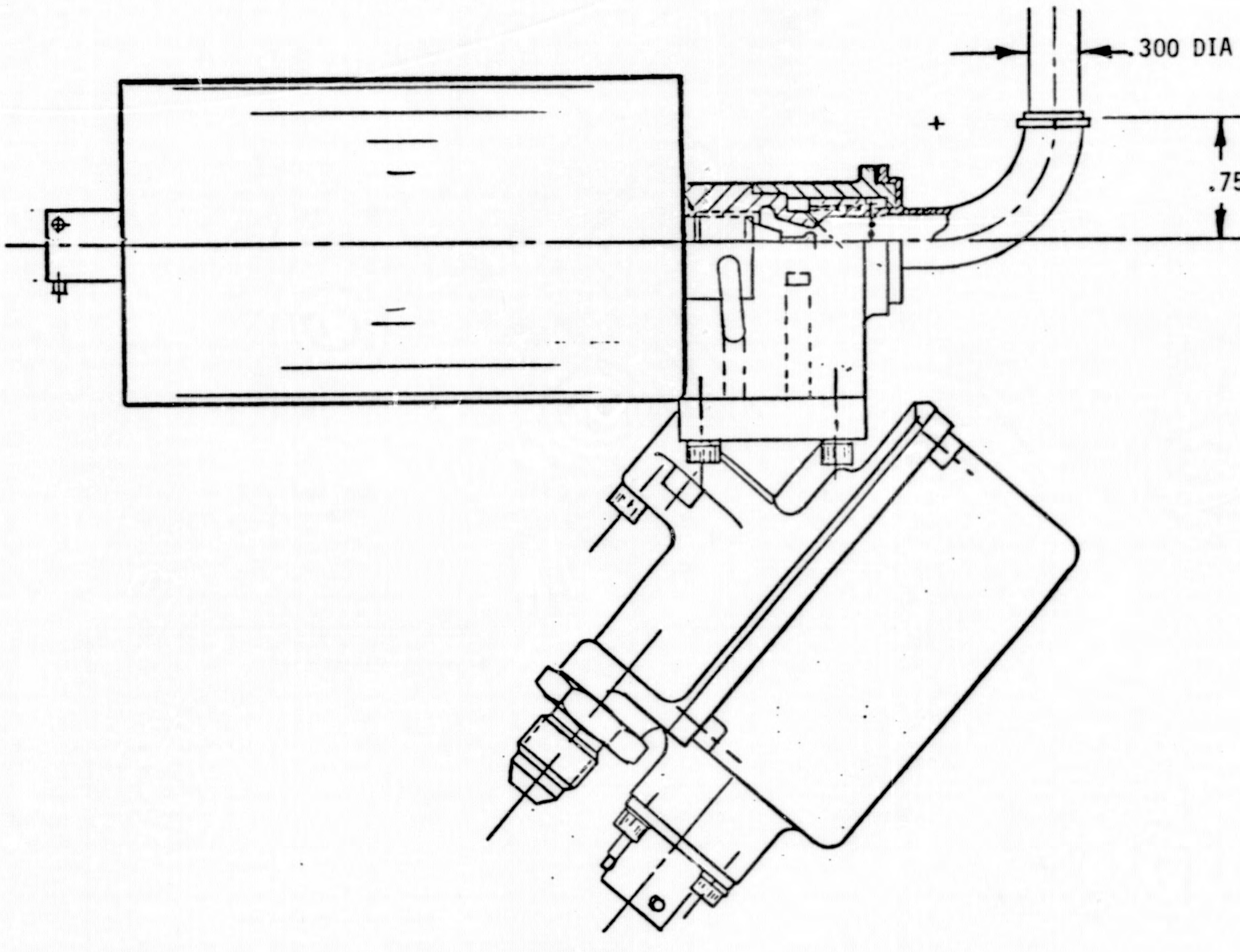


Figure 2.10-25 Configuration #3. Oxidizer System Gas Generator

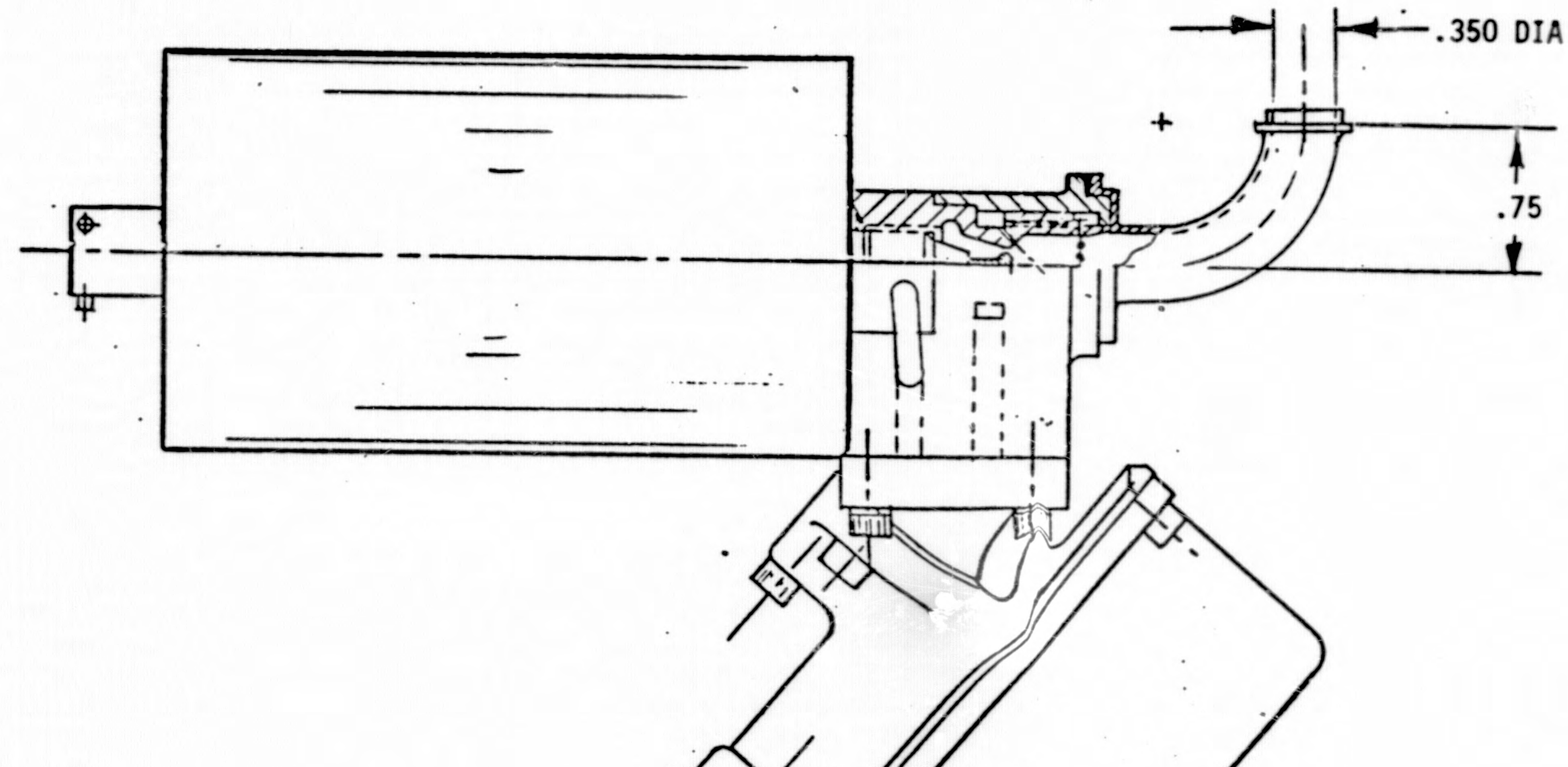
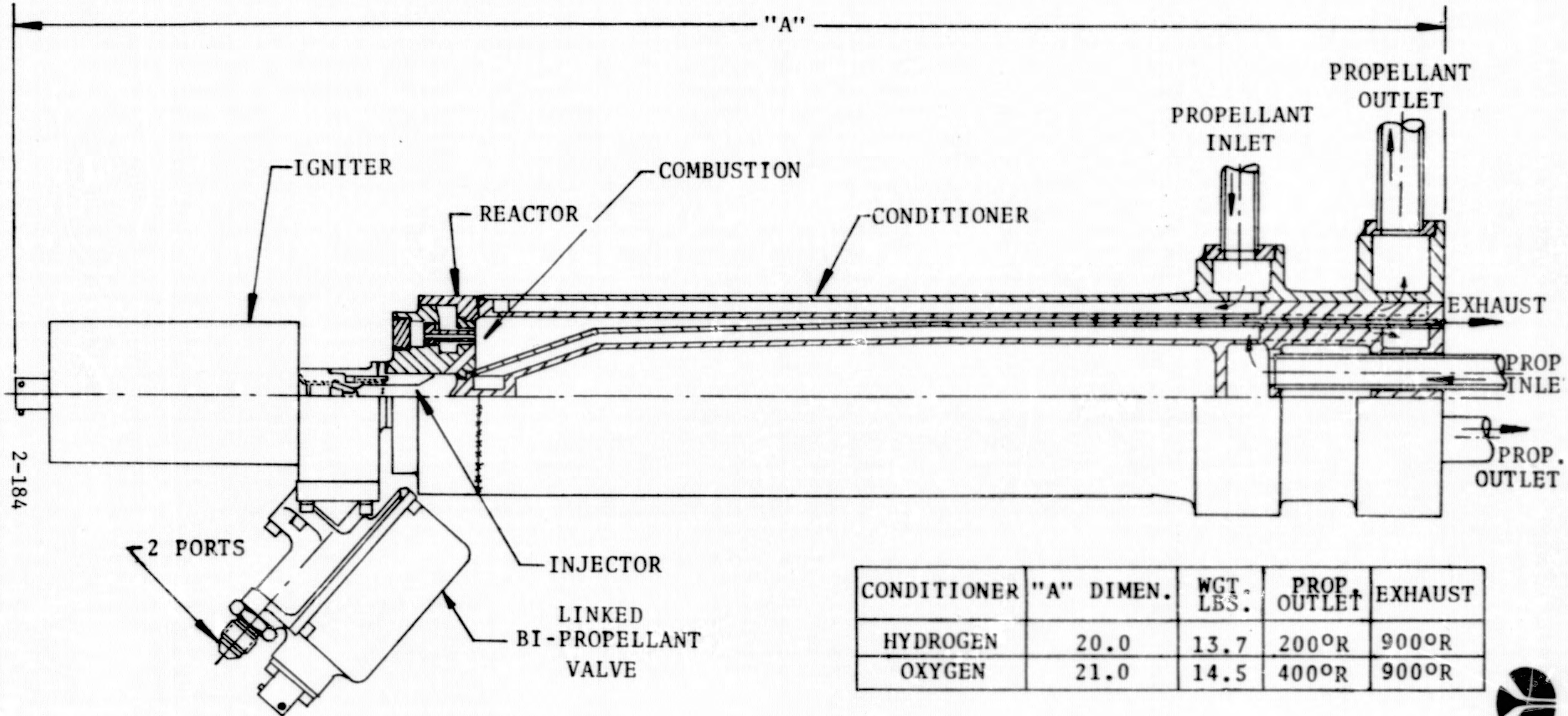


Figure 2.10-26 Configuration #3. Hydrogen System Gas Generator



CONDITIONER	"A" DIMEN.	WGT. LBS.	PROP. OUTLET	EXHAUST
HYDROGEN	20.0	13.7	200°R	900°R
OXYGEN	21.0	14.5	400°R	900°R

Figure 2.10-27 Heat Exchanger Assembly

flow in an annulus between the liquid propellants. The liquids flow through baffles counter to the hot gas flow (toward the hot gas generator) and then return parallel to flow (toward exit) and exit as gas at the required temperature near the point of entry. The gas generator assembly is similar to the design used for the turbopumps, but is integrated into the heat exchanger design.

The gas generator 3:1 mixture ratio was selected for the design since variations in inlet temperature would have a small effect on the combustion temperature. Figure 2.10-28 indicates the small effect a 250°R change GH_2 temperature would have on the 4500°R combustion temperature. This concept would be desirable for the turbopumps except that the turbines cannot tolerate such high temperatures. However, designing heat exchangers for this temperature is feasible and within the state-of-the-art.

The heat exchangers are designed to convert 0.5 LB per sec. of LH_2 and 1.5 LBS of LOX to 200°R GH_2 and 400°R GOX. The propellants required for these conversions are as follows:

	$\dot{W}_{GH_2}$ (lb/sec)	$\dot{W}_{GOX}$ (lb/sec)
GH_2 Heat Exchanger	0.0160	0.0480
GOX Heat Exchanger	0.0102	0.0306
	0.0262	0.0786

Flow Controllers. The variation in inlet conditions of the combustants (GOX and GH_2) to the gas generators for the turbopump and heat exchanger, and the thrusters results in fluctuations of the combustion temperatures. These fluctuations are more extreme for the lower mixture ratios. The problem is created by the variable propellant conditions due to accumulator blowdown, the space environment, and the time between operations.

If left uncontrolled, the resulting variations in combustion temperature would cause wide fluctuations in thruster, turbopump and heat exchanger performance. Component life studies indicate that low mixture ratio gas generators should be controlled within 200°R and thruster combustion temperatures within 500°R. Mixture ratio variations are the primary reason for the change in combustion temperature, and hence mixture ratio (mass) control for the gas is critical to assure component life. The effect of the hydrogen inlet temperature is far more pronounced at the low mixture ratios. Figure 2.10-29 shows the combustion temperature as a function of both hydrogen and oxygen inlet temperatures at the nominal 1:0 mixture ratio. At this low mixture ratio, the hydrogen inlet temperature is far more important than the oxygen temperature, and produces a degree-for-degree change in combustion temperature.

The use of devices (flow controllers) to control the mass flow to the gas generators for the turbopump and heat exchanger are contemplated. Due to the low mixture ratio (1:0) required to prevent damaging temperatures for the for the turbine, a more sophisticated device will be required to control the

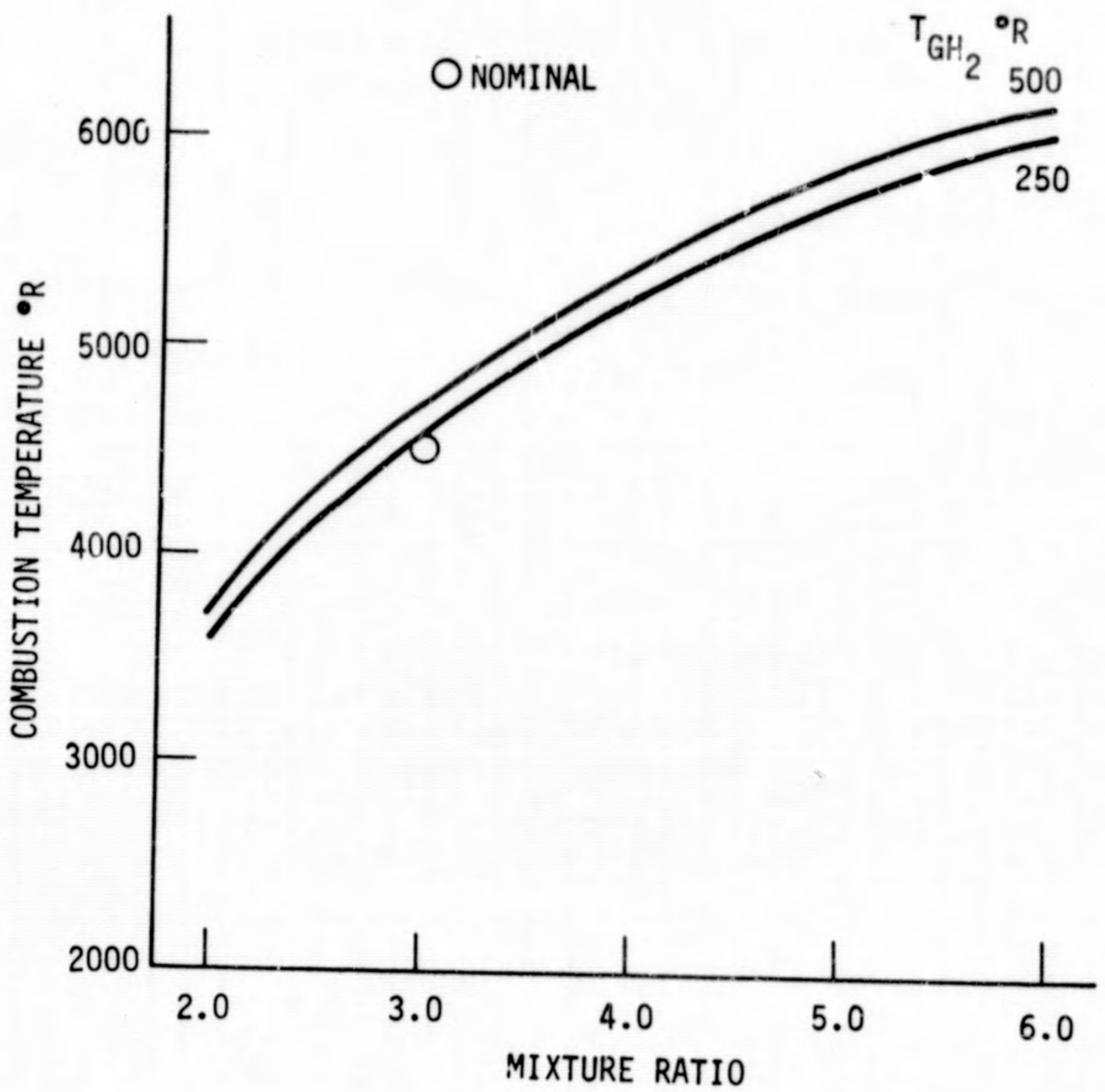


Figure 2.10-28. Combustion Temperatures

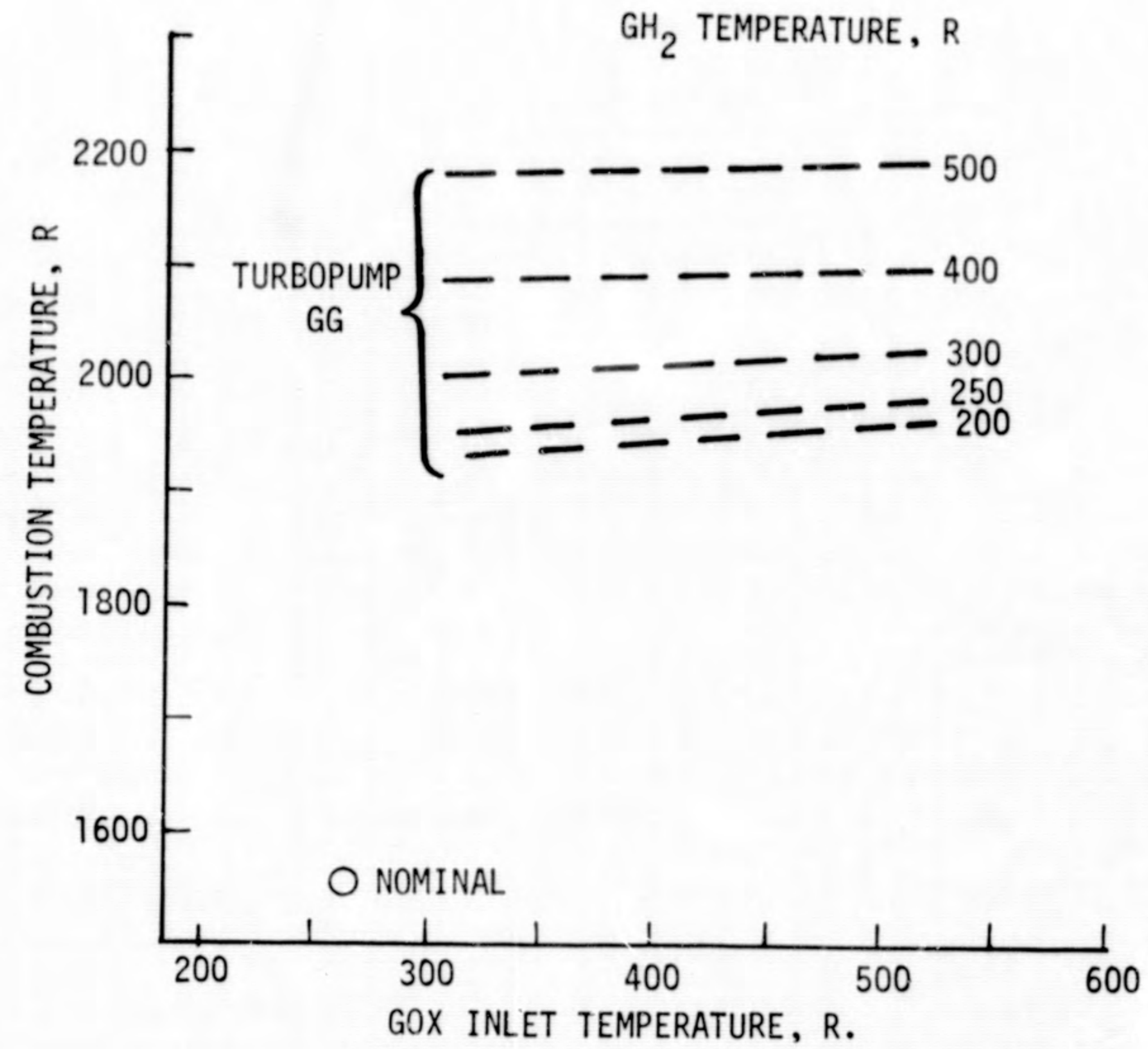


Figure 2.10-29 Turbopump GG Combustion Temperature vs Propellant Inlet Temperature

combustion temperatures within 200°R than for the heat exchangers or thrusters. The gas generator for the heat exchanger operates at a 3:1 mixture ratio while the thrusters operate at a 4:1 mixture ratio. Both are similarly sensitive to the inlet conditions of the combustants and mixture ratio as shown in Figure 2.10-30. It appears that a passive device would be satisfactory for the heat exchanger and the thruster. For this study, a flow control device was not planned for the thrusters. However, if the device is necessary, Rocketdyne's present studies indicate it would probably weigh two pounds. Consequently, a 28 LB. (14 thrusters) weight penalty could be incurred. Rocketdyne is presently studying flow controllers in the NASA-Lewis Research Study NA3-14390, "Investigation of Propellant Flow Control System." Following is a brief discussion of control methods.

Numerous control methods are possible to provide the required operating bands. However, they all contain three basic elements: sensors, logic, and a control mechanism. Since it is impossible (on a vehicle) to measure thrust and mixture ratio directly, any method must sense one or more of the following primary parameters: chamber pressure, chamber temperature, propellant inlet pressure, propellant inlet temperature, or propellant flow rate. These are the most common control parameters, since they have either a direct relationship or a reasonably good predictable relationship to the desired control conditions (thrust or mixture ratio).

The system logic can embody either open-loop or closed-loop control. The former method senses the system variables (inlet temperature and pressure) and causes control settings to be made which will provide the desired control condition (thrust and mixture ratio). However, no feedback of the actual control condition is incorporated. The advantage of an open-loop scheme is that it is capable of providing the proper control signals to the control mechanism before flow is initiated, thereby eliminating any start-up response lag. The alternative, closed-loop feedback of the controlled condition, does not possess this feature. However, it offers a simpler logic function. The logic for either approach may be either inherent (performed by the sensed parameters by an appropriate flow circuit) or external (performed by a controller, either electronic, mechanical, or fluidic). Inherent logic has the advantage of simplicity, however, it lacks the broader logic capability of an external controller.

The flow control mechanism, can be either an active device or a passive device. An active device requires an input signal which causes the movement of a control element. A passive device requires no input signal to perform its function. It utilizes temperature and pressure sensitive devices to control flow. A passive mechanism offers high reliability and long life, however, its functions are extremely limited in application.

Regulators. Quad regulators were selected to provide the APS with an operational attitude hold capability after the failure of a single APS component including a regulator. Therefore, considering the possibility of failing open or closed, the redundancy of the quad design was chosen. Each quad weighs 14 lbs. and consists of two branches each with two regulators in series. The inlet ports are 3/8" and the outlet ports are 5/8". Each branch was designed to flow 0.25 lbs/sec min. GH_2 at 200°R or 0.75 lbs/sec min. GOX at

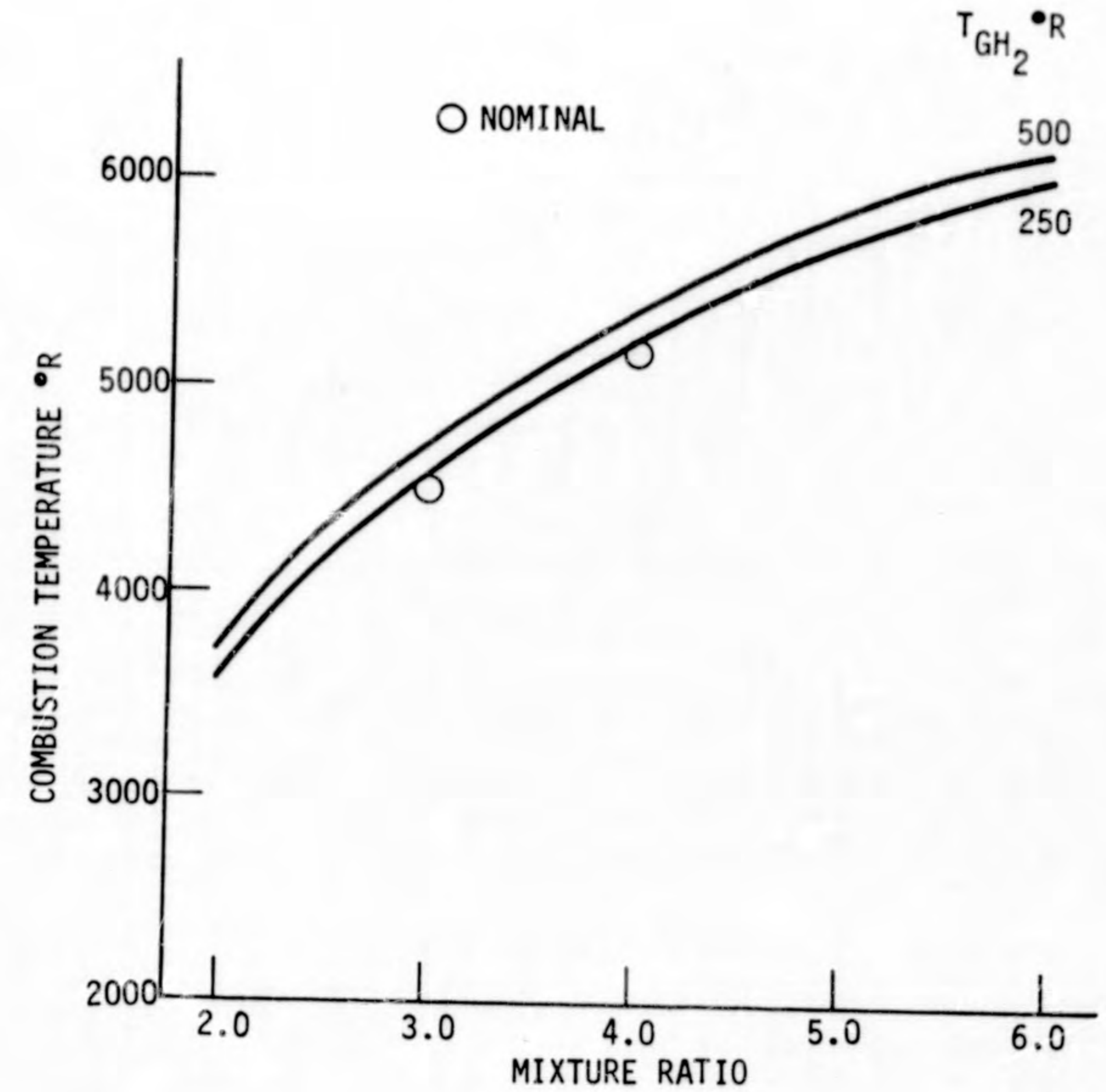


Figure 2.10-30 Thruster and Heat Exchanger Combustion Temperatures

$400^{\circ}R$ with inlet pressures of 1250 to 475 psia and an outlet of 375 ± 25 psia. Only one regulator in each branch functions under normal conditions. If it should fail open, then the second regulator would take over. The design is to incorporate pressure sensing to enable the APS control system to determine a failure has taken place when a regulator has failed open or closed. At this point, the control system would shutdown and isolate the propellant conditioning systems. It would also stop the flow of reactants to the fuel cell by moving the fuel cell/vent selector valve to the vent position. The control system would now limit the use of the APS stored gas in the accumulators to the thrusters for attitude hold.

Motor Operated Ball Valves. Motor operated ball valves were selected as isolation valves to minimize weight, and because fast operation is not necessary, the 115 VAC required for operation of these valves will be available from the GSE and Shuttle whenever the valves are to be operated. The use of AC rather than DC eliminates the brush arcing problems or need for case pressurization which are common to DC motors in vacuum operation.

For the TUG application, four 3/4" valves are required. Each valve weighs 2-1/2 lbs. and operates on 60 watts.

Other Components. The other components used in the APS will not be discussed. They are present state-of-the-art and have been successfully used in similar applications.

Component Tables. The physical, electrical and operating characteristics of the APS components are compiled in three tables:

Table 2.10-10	List of All APS Components with Brief Description of Operating Characteristics and Weight Estimates
Table 2.10-11	Summary of APS Component Weight Estimates
Table 2.10-12	APS Components Electrical Power Requirements

2.10.3 System Selection and Options

Because of study direction to use GOX and GH_2 for attitude control, no other system was considered.

2.11 INTERFACES

2.11.1 Panel Configuration

All Tug electric and fluid interfaces with the Orbiter have been combined into three panels, the LOX panel, the LH_2 panel and the aft panel. Figure 2.11-1 shows the location of the LOX and LH_2 panel in relation to the Orbiter. Figure 2.11-2 shows the orientation of the panels.

The LOX panel is parallel to the vehicle mold line and protrudes from the skin. It is approximately 18" X 24" and located at Station 249 with $\emptyset = 8^{\circ}$.

Table 2.10-10. List of All APS Components With Brief Description of Operating Characteristics and Weight Estimates

	Component	Qty	Description	Est. Max. Wt.
17L	Regulator, GOX, Quad Assy.	1	Quad arrangement, flow 1.5 lbs per sec 400°R GOX at 1250 -475 psia inlet and 375 ± 25 psia outlet, inlet port 3/8" and outlet port 5/8" for each branch.	14
17R	Regulator, GH_2, Quad Assy.	1	Quad arrangement, flow .5 lbs per sec 200°R GH_2 at 1250 -475 psia inlet and 375 ± 25 psia outlet, inlet port 3/8" and outlet port 5/8" for each branch.	14
16R	Burst Diaphragm, GH_2	1	1/8" ports, rupture at 1750 ±100 psid at 200°R with min. restriction .070" dia. and 1600 psid min. at 530°R.	½
16L	Burst Diaphragm, GOX	1	1/8" ports, rupture at 1750 ±100 psid at 400°R with min. restriction .070" dia. and 1600 psid min. at 530°R.	½
5R	Bypass Valve, LH_2	1	3/8" ports for flow and ¼" port for ref. pressure. Special valve designed to bypass pump output until pump outlet and reference pressures are equal, LH_2 pump flow is .5 lbs per sec. at 1250 psia.	2½
5L	Bypass Valve, LOX	1	3/8" ports for flow and ¼" port for ref. pressure. Special valve designed to bypass pump output until pump outlet and reference pressures are equal, LOX pump flow is 1.5 lbs per sec. at 1250 psia.	2½
26	Thruster, Assy.	10	70 LB_F, 250 psia chamber pressure, GOX 400°R and GH_2 200°R, expansion ratio 60:1, igniter and bi-propellant valve, MR4:1.	80
26	Thruster, Assy.	4	20 LB_F, 250 psia chamber pressure, GOX 400°R and GH_2 200°R, expansion ratio 60:1, igniter and bi-propellant valve, MR4:1.	4.1
18	Ball Valve, Motor Operated	4	3/4" ports flow .5 lbs per sec. GH_2 at 200°R and .5 lbs per sec. LH_2 at 1250 psia with 3 psid max; flow 1.5 lbs per sec. GOX at 400°R and 1.5 lbs per sec. LOX at 1250 psia with 3 psid max; 60 watts max at 115 VAC.	2½
27	2-Way Solenoid Valve	2	3/8" ports flow .06 lbs per sec. Min. GH_2 at 375 psia and 200°R, and .11 lbs per sec. min. GOX at 375 psia and 200°R, 10 psid max, 84 watts max at 28 VDC.	2½
19	3-Way Solenoid Valve	2	¼" ports flow .1 lbs per sec. Min. GH_2 at 375 psia and 200°R, and .26 lbs per sec. min. GOX at 375 psia and 400°R, 84 watts max at 28 VDC.	2½
6 20	2-Way Solenoid Valve	4	¼" ports, flow .1 lbs per sec. GH_2 at 375 psia and 200°R and .26 lbs per sec. GOX at 375 psia and 400°R, 84 watts max at 28 VDC.	2
24	Squib Valve	10	Bi-Valve with 3/8" ports, permanently closed when actuated, utilizes EBW and draws 1 amp max, 1 watt at 28 VDC.	1
24	Squib Valve	4	Bi-Valve with ¼" ports, permanently closed when actuated, utilizes EBW and draws 1 amp max, 1 watt at 28 VDC.	1
15L	Accumulator, GOX	1	2.4 Ft^3 sphere with two ports 3/4", material AL 2014-T6, RM. temp. yield 2060 psid, burst 2300 psid.	19
15R	Accumulator, GH_2	1	1.5 Ft^3 sphere with two ports 3/4", material 6AL-4V-Ti, RM. temp. yield 1650 psid, burst 1850 psid.	20
13L	Turbopump, LOX Assy.	1	0 NPSP inlet, centrifugal pump 3 stage + inducer, flow 1.5 lbs per sec. LOX at 1250 psid, ports 3/4" inlet and ½" outlet, turbine partial admission impulse inlet 270 psia and 1960°R, exhaust 13.5 psia and 900°R, igniter and gas generator.	18

Table 2.10-10. List of All APS Components With Brief Description of Operating Characteristics and Weight Estimates (Cont)

	Component	Qty	Description	Est. Max. Wt.
13R	Turbopump, LH_2 Assy.	1	0 NPSP inlet, centrifugal pump 2 stage + inducer, flow .5 lbs per sec. LH_2 at 1250 psid, ports 3/4" inlet and ½" outlet, turbine 2-row velocity compounded, inlet 270 psia and 1960°R, exhaust 13.5 psia and 900°R, igniter and gas generator.	11.2
14L	Heat Exchanger, GOX, Assy.	1	Baffled, single pass, integral igniter and gas generator, MR3:1, combustion temp. 4500°R, exhaust 900°R, flow in 1.5 lbs per sec LOX at 1250 psia and GOX out at 400°R and 1240 psia, 2 inlet ports ½", 2 outlet ports 5/8".	18
14R	Heat Exchanger, *Assy.*	1	Baffled, single pass, intergral igniter and gas generator, MR3:1, combustion temp. 4500°R, exhaust 900°R, flow in .5 lbs per sec. LH_2 at 1250 psig and GH_2 out at 200°R and 1240 psia, 2 inlet ports ½", 2 outlet ports 5/8".	13.7
10R	Flow Controller, Elec. Mech. sonic	1	Elec-Mech. device to regulate flow of GOX and GH_2 into gas generator at MR .97:1 and .036 lbs per sec. to operate LH_2 turbopump.	4
10L	Flow Controller, Elec. Mech. sonic	1	Elec.-Mech. device to regulate flow of GOX and GH_2 into gas generator at MR.97:1 and .027 lbs per sec. to operate LOX turbopump.	4
9R	Flow Controller, Passive	1	Passive device to regulate flow of GOX and GH_2 into gas generator at MR3:1 and .064 lbs per sec. to operate LH_2 heat exchanger.	2
9L	Flow Controller Passive	1	Passive device to regulate flow of GOX and GH_2 into gas generator at MR3:1 and .0408 lbs per sec. to operate LOX heat exchanger.	2
25	2-Way Solenoid Valve	1	1/8" ports, flow .0008 lbs per sec. GH_e at 50 psia and 530°R.	1½
4	Check Valve, Series	2	½" ports, flow .5 lbs per sec. LH_2 and 1.5 lbs per sec. LOX at 15 psid max, crack 2 psid min, arranged two in series.	1½
2	Check Valve	2	¼" ports, flow .365 lbs per sec. GH_2 at 1250 psia and 200°R and .95 lbs per sec. GOX at 1250 psia and 400°R.	½
1	Disconnect	2	½" ports, flow .365 lbs per sec. GH_2 at 1250 psia and 200°R and .95 lbs per sec. GOX at 1250 psia and 400°R, internal check valve in airborne half which is active only after separation.	2
28	Orifice	1	1/8" ports, flow .0002 lbs per sec. GH_e at 50 psia and 530°R.	½
21R	Orifice	1	¼" inlet and 3/8" outlet, flow .05 lbs per sec GH_2 at 375 psia and 200°R.	½
21L	Orifice	1	¼" inlet and 3/8" outlet, flow .1 was per sec GOX at 375 psia and 400°R.	½
22R	Orifice	1	1/8" ports, flow .0001 #/sec GH_2 at 375 psia and 200°R.	½
22L	Orifice	1	1/8" ports, flow .0008 #/sec GOX at 375 psia and 400°R.	½

Table 2.10-11. Summary of APS Component Weight Estimates

	LOX/GOX	LH_2/GH_2
Turbopumps Assy.	18	11.2
Heat Exchanger Assy.	14.5	13.7
Turbopump Flow Controller	4	4
Heat Exchanger Flow Cont.	2	2
Accumulator	19	20
Regulators	14	14
Check Valves	2	2
Disconnects	2	2
Burst Diaphragm	.5	.5
Bypass Valve	2.5	2.5
Orifices	1.5	1
Solenoid Valves	9.75	8.5
Motor OPTD Ball Valves	5	5
Thrusters (96.4 lbs Total Wt)	48.2	48.2
Squib Valves	7	7
Press. SWS.	4	4
Temp. SWS	1	1
Total Combined Weight 301.55 lbs.	154.95	146.60

Weight does not include instrumentation, exhaust system, lines, fittings, insulation, or component mountings.

Table 2.10-12. APS Components Electrical Power Requirements

Subsystem -			Operational Phase -				
			Power Required (in Watts)				
Component	Voltage/ Frequency Required	Regulation Required (Percent)	Max Demand Inrush/Peak Cryo	Operational Usage (Avg) AMB	Standby Usage (Avg)	Emergency Power	Duration (Duty Cycle)
Sol. Vlv 1/8" 2-way (1 valve)	28 VDC	N/A	42	21	0		LOX Pump OPN +5 SECS
MTR OPTD Ball 3/4" (4 valves)	115 VAC	N/A	60	30	0		2 operate in flt. twice
Sol. Vlv ¼" 3-way (2 valves)	28 VDC	N/A	84	42	0		Not Planned
Sol. Vlv 3/8" 2-way (2 valves)	28 VDC	N/A	84	42	0		Every time a GC is operated.
Bi-Propellant Vlv, GG (4 valves)	28 VDC	N/A	13	6	0		1570 sec. of OPN. each.
Igniter, GG (4 units)	28 VDC	N/A	2.5	2.5	0		20 secs. of OPN each.
Bi-Propellant Valve, thruster (4-20 LB_F 10-70 LB_F)	28 VDC	N/A	20	20	0	20	max. 6 simul.
Igniter, thruster (14 units)	28 VDC	N/A	28	28	0	28	max. 6 simul.
Squib Valve (14 units)	28 VDC	N/A	1	1	0	1	1 OPN not planned
Sol. Valve ¼"/2-way (4 valves)	28 VDC	N/A	84	42	0	84	2 not planned 2 prior to first start only
Flow Controller (2 assys.)	28 VDC	N/A	50	50	0		1 for LOX pump 1 for LH_2 pump on when pump on + start up

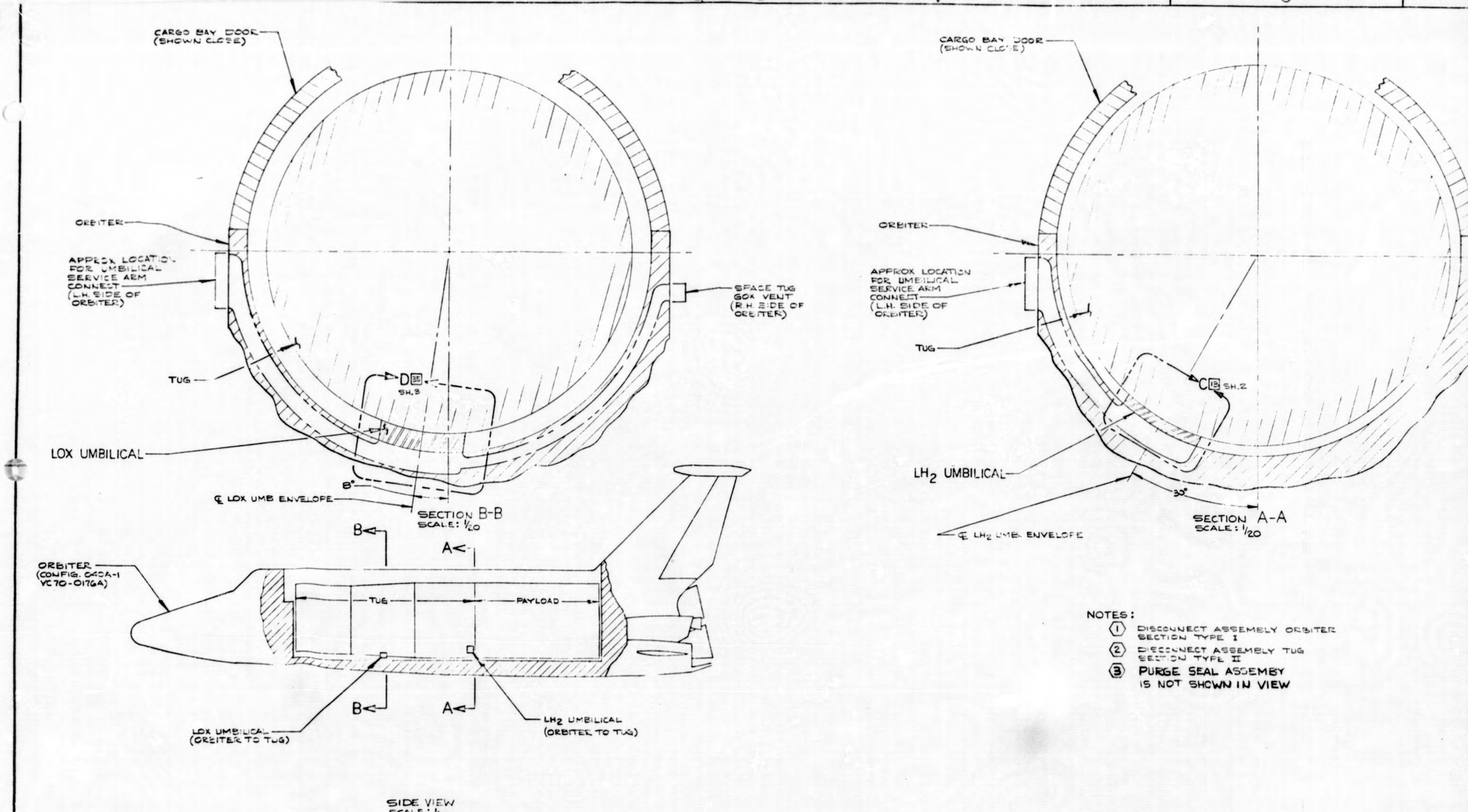


Figure 2.11

F-1

8 7 6 5 4 3

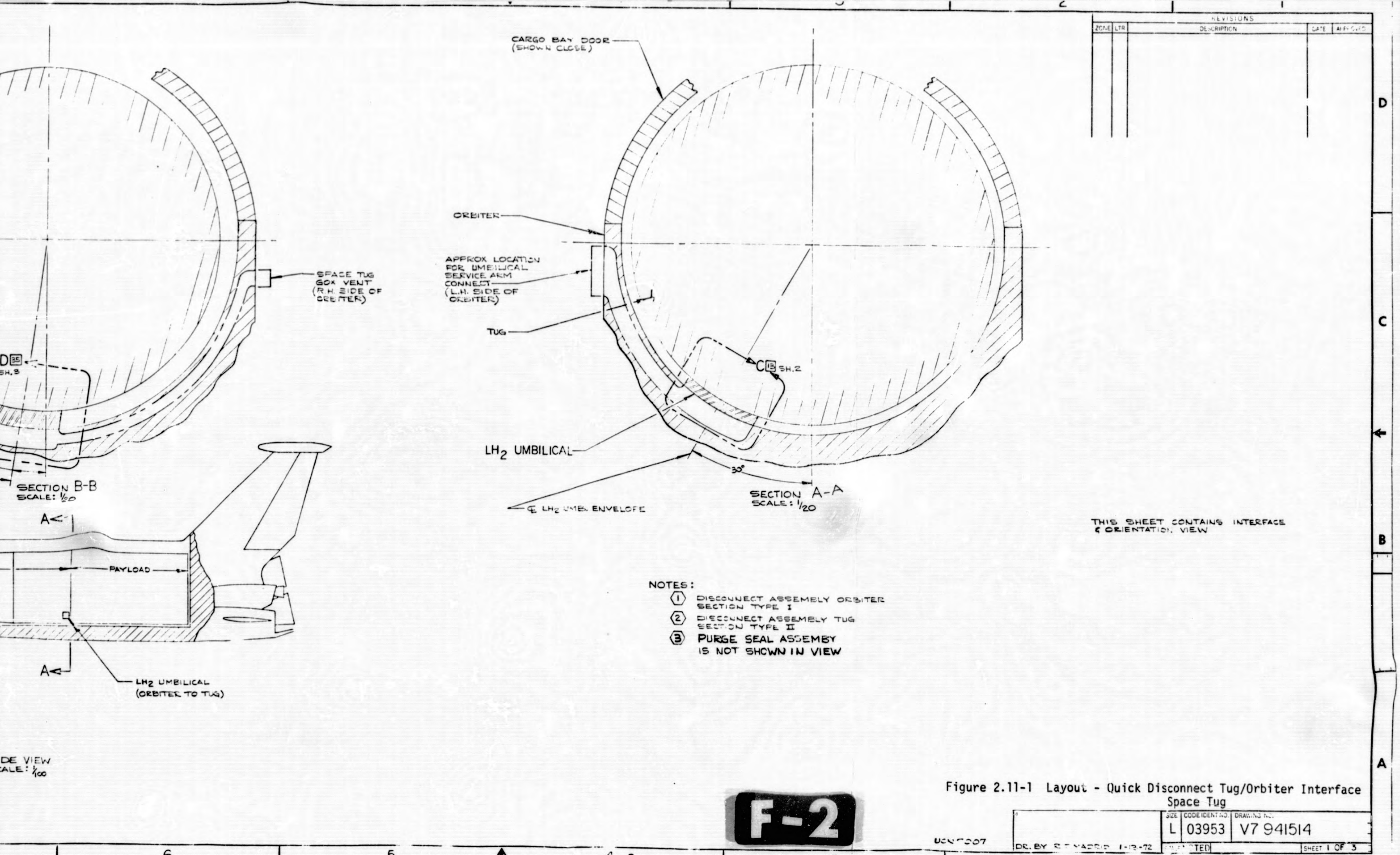


Figure 2.11-1 Layout - Quick Disconnect Tug/Orbiter Interface
Space Tug

2-195,2-196

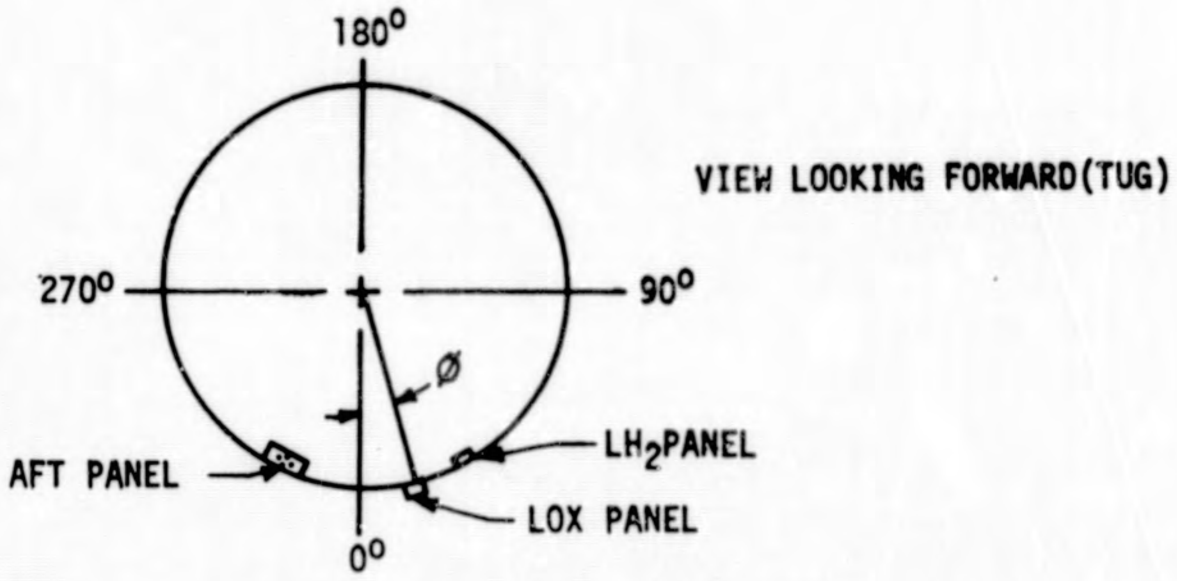


Figure 2.11-2 Panel Orientation Schematic

The panel is composed of five fluid and one electrical connection as shown in Table 2.11-1. The maximum operating pressures are noted. The panel configuration and details of the umbilical mechanism are shown in Figure 2.11-3.

The LH_2 panel is also parallel to the vehicle mold line but recessed into the skin. It is approximately 18" X 18" and located at Station 449 with Ø = 30°. It has four fluid and one electrical connections are indicated in Table 2.11-2. The panel configuration and details of the umbilical mechanism are shown in Figure 2.11-4.

The LOX panel was located at Ø = 8° in order not to interfere with the main structural member of the Orbiter wing. The LH_2 panel was located as far from the LOX panel as was possible. Ø = 30° is the maximum angle possible and still keep the actuator mechanism for the mating panel within the thick portion of the Orbiter wing.

The aft panel is in the plane of the Tug/Tug support structure interface (Station 130.5). It is facing aft at a radius of 66" and with an angle of Ø = 345°. It has two fluid and one electrical disconnects as shown in Table 2.11-3.

Matching movable umbilicals for the LH_2 and LOX panels will be provided as a part of the orbiter. Each umbilical contains two locating pins and the free-floating mating half of the fluid and electrical disconnects. An estimated misalignment of the panels of 0.5 inches can be accommodated. The

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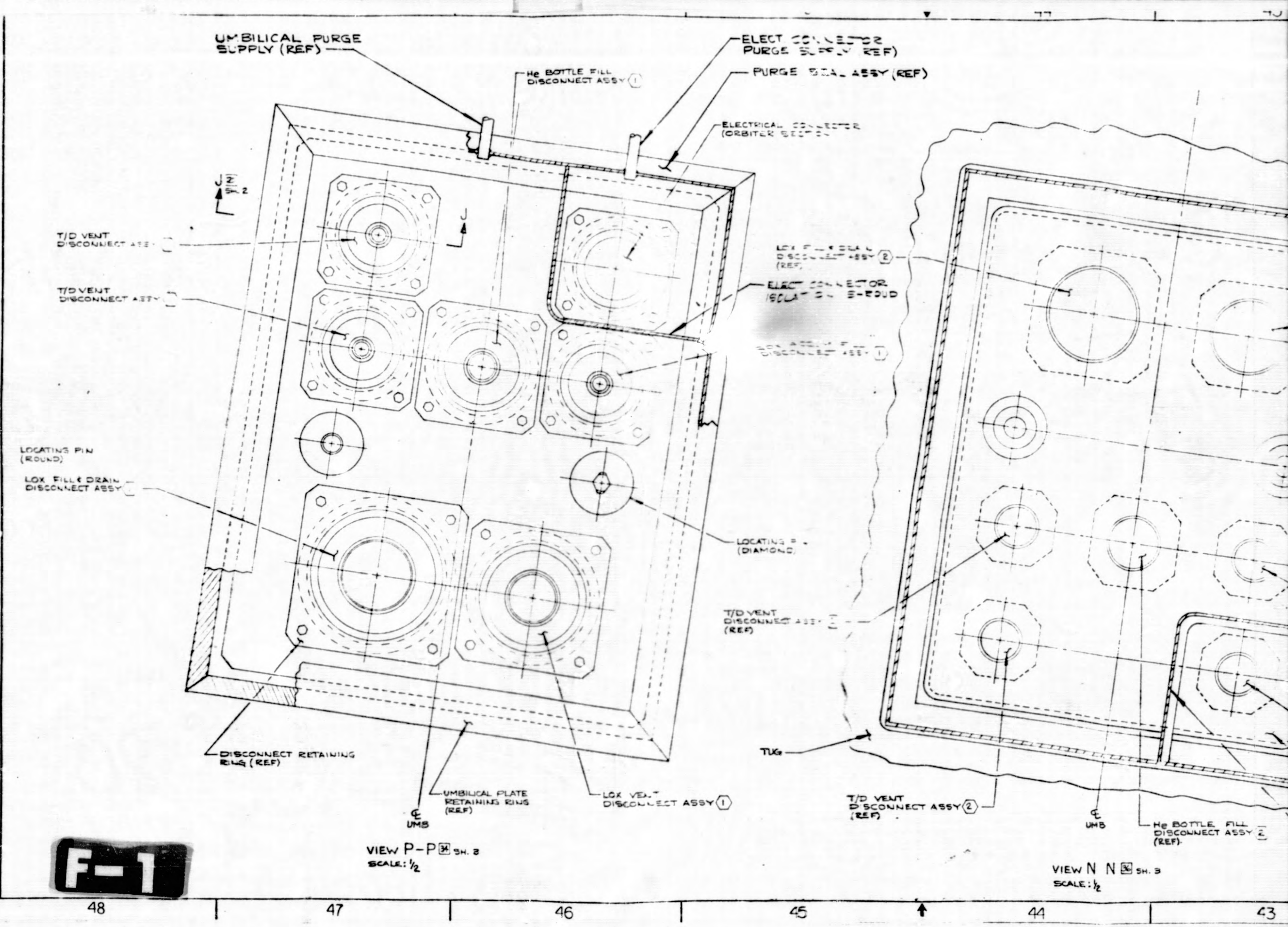
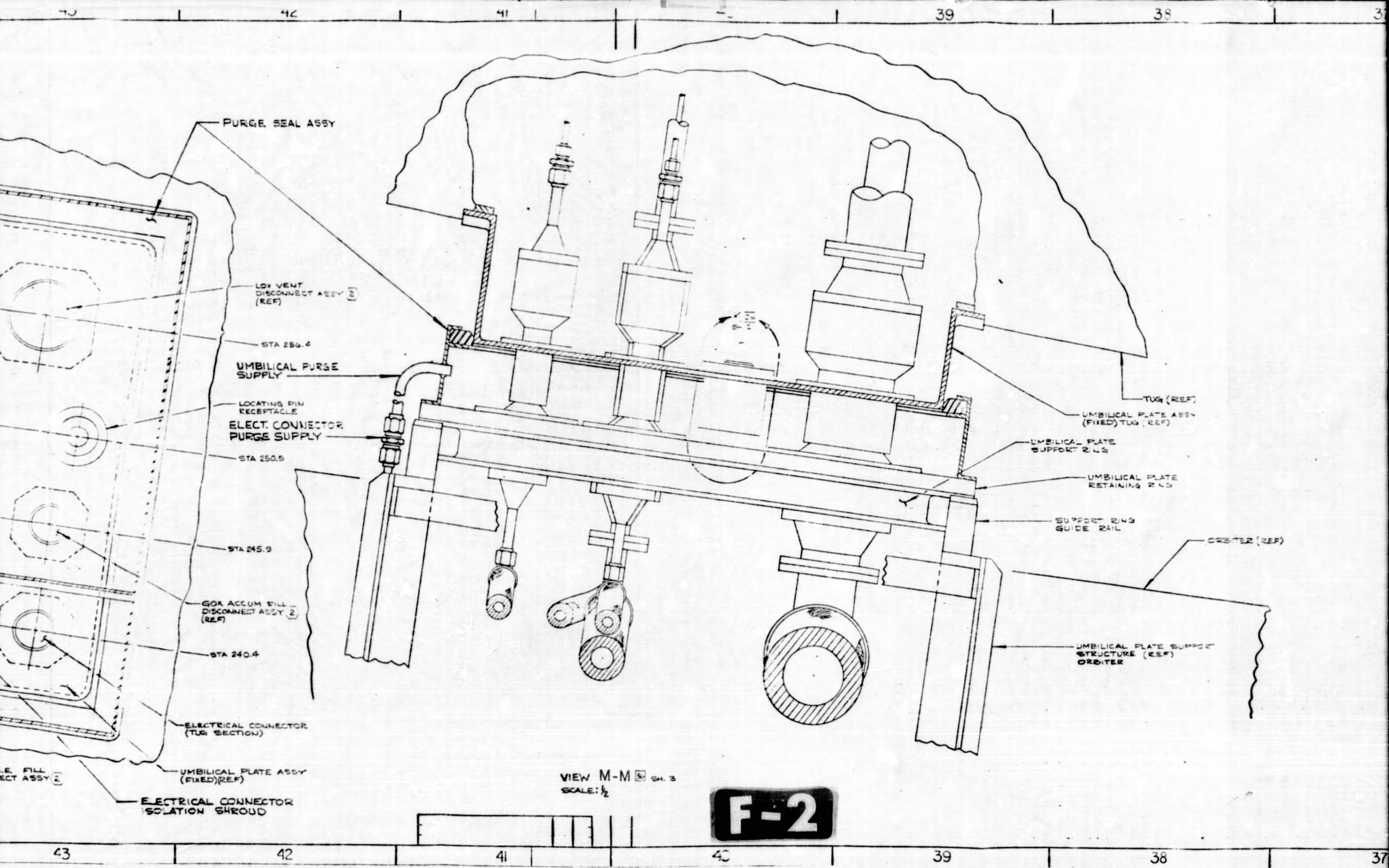


UMBILICAL PURGE SUPPLY (REF)
He BOTTLE FILL DISCONNECT ASSY 1
ELECT CONNECTOR PURGE SUPPLY (REF)
PURGE SEAL ASSY (REF)
ELECTRICAL CONNECTOR (ORBITER SECTION)
T/D VENT DISCONNECT ASSY
T/D VENT DISCONNECT ASSY
ELECT CONNECTOR ISOLATION SHROUD
LOCATING PIN (ROUND)
LOX FILL & DRAIN DISCONNECT ASSY 1
LOCATING PIN (DIAMOND)
DISCONNECT RETAINING RING (REF)
UMBILICAL PLATE RETAINING RING (REF)
LOX VENT DISCONNECT ASSY 1
℄ UMB
VIEW P-P 34 SH. 3
SCALE: 1/2
LOX FILL & DRAIN DISCONNECT ASSY 2 (REF)
T/D VENT DISCONNECT ASSY 2 (REF)
TUG
T/D VENT DISCONNECT ASSY 2 (REF)
℄ UMB
He BOTTLE FILL DISCONNECT ASSY 2 (REF)
VIEW N-N 34 SH. 3
SCALE: 1/2
F-1
48
47
46
45
44
43



VIEW M-M SH. 3
SCALE: 1/2

F-2

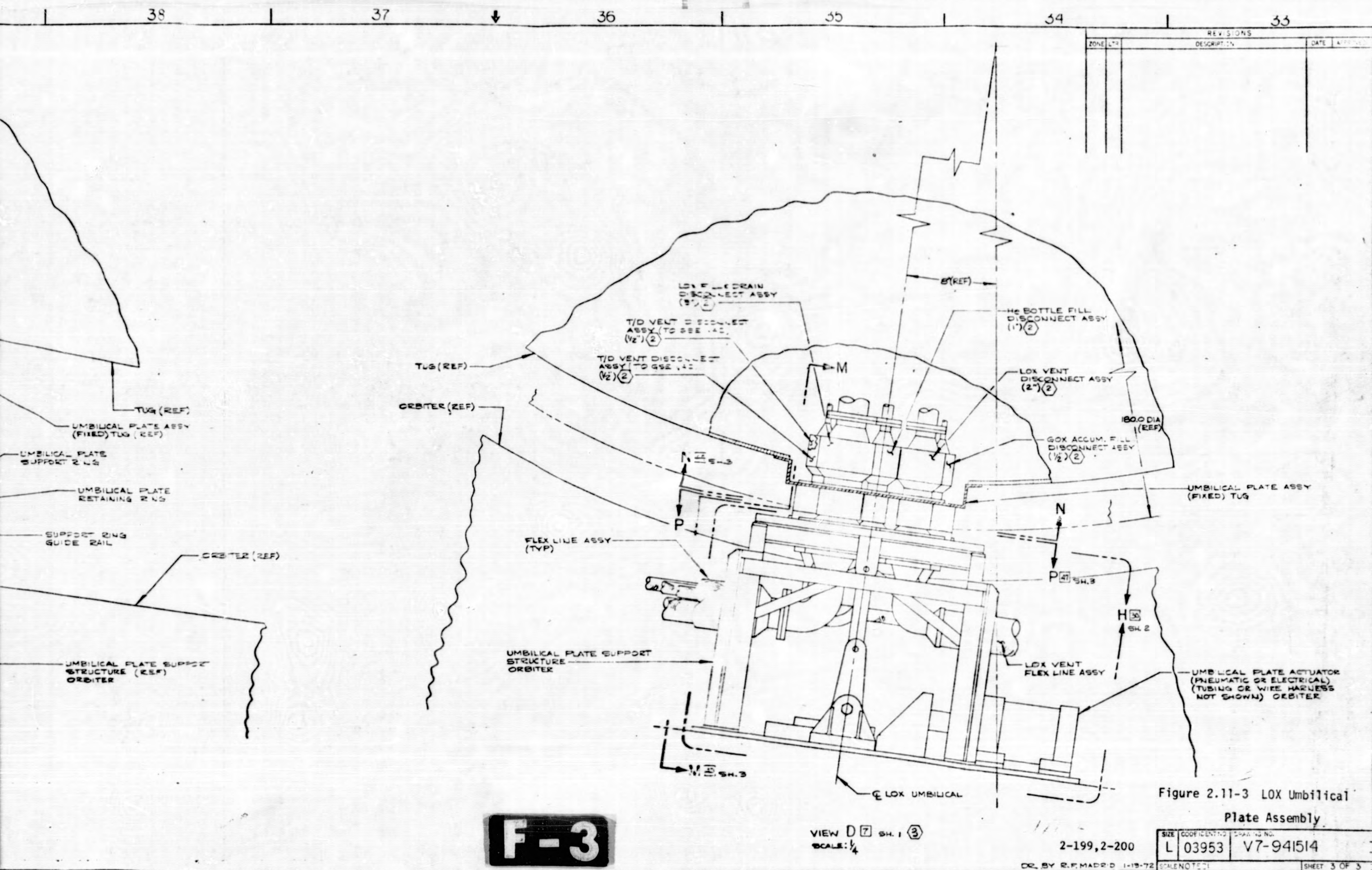
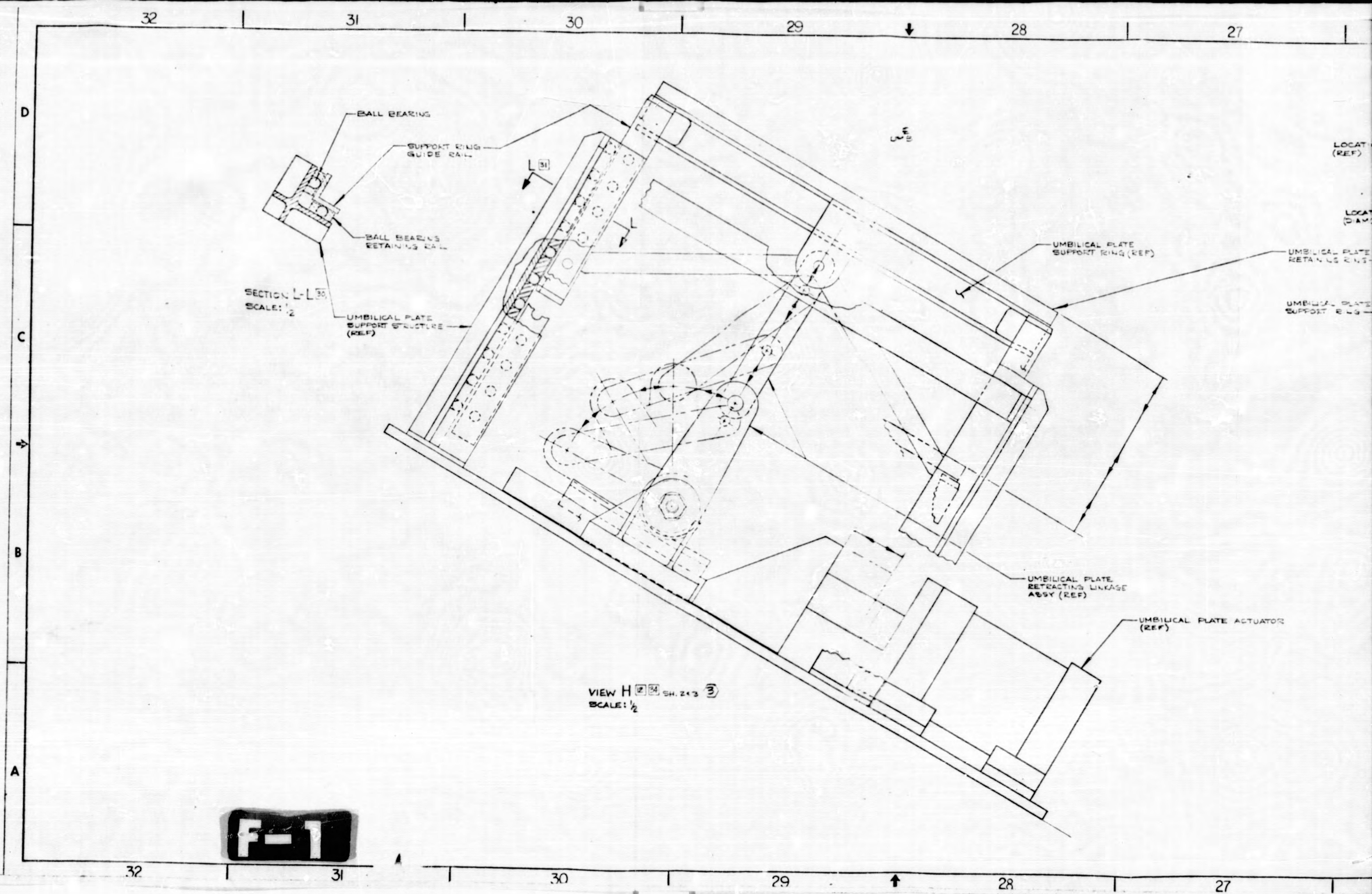


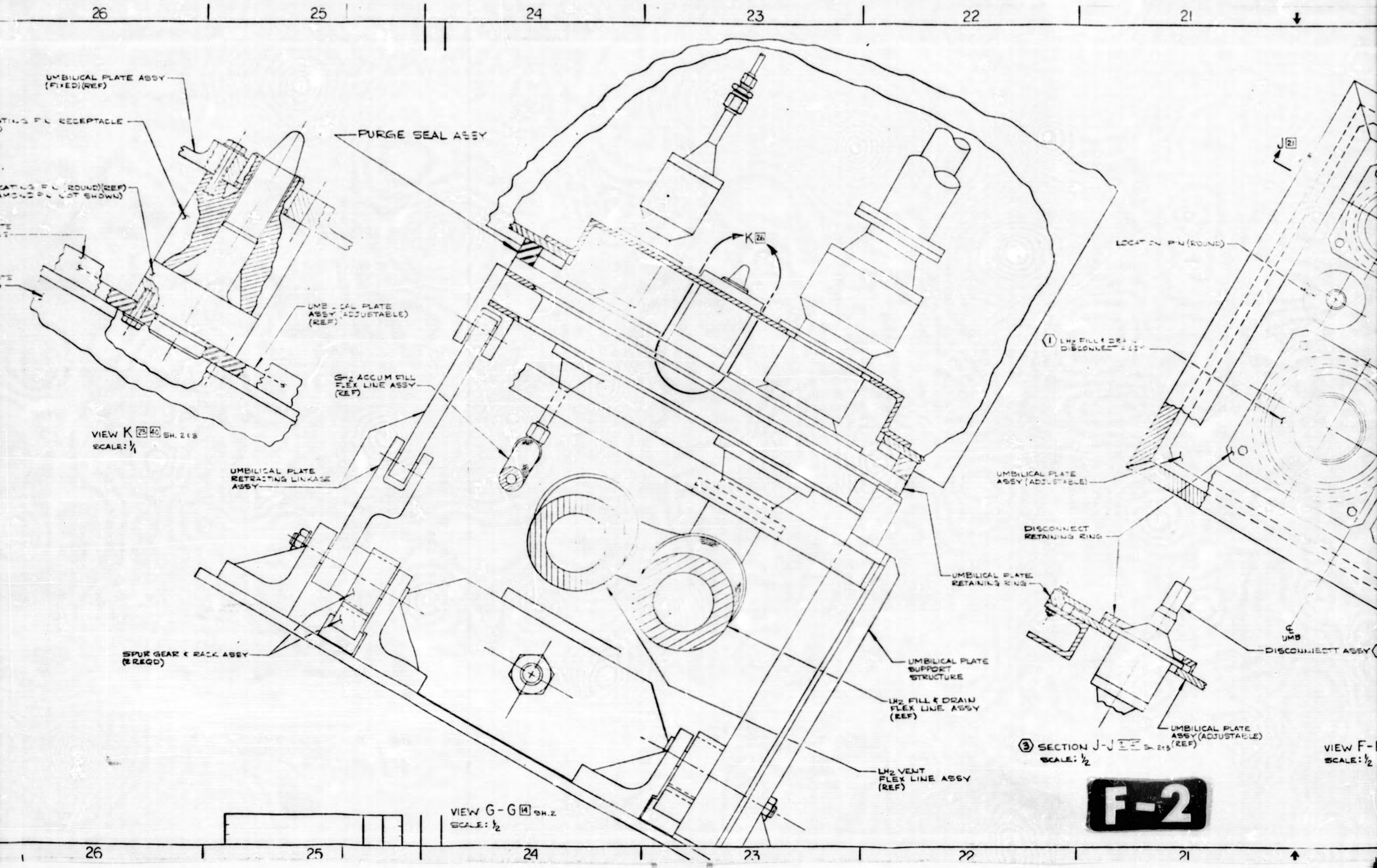
Figure 2.11-3 LOX Umbilical Plate Assembly

2-199, 2-200

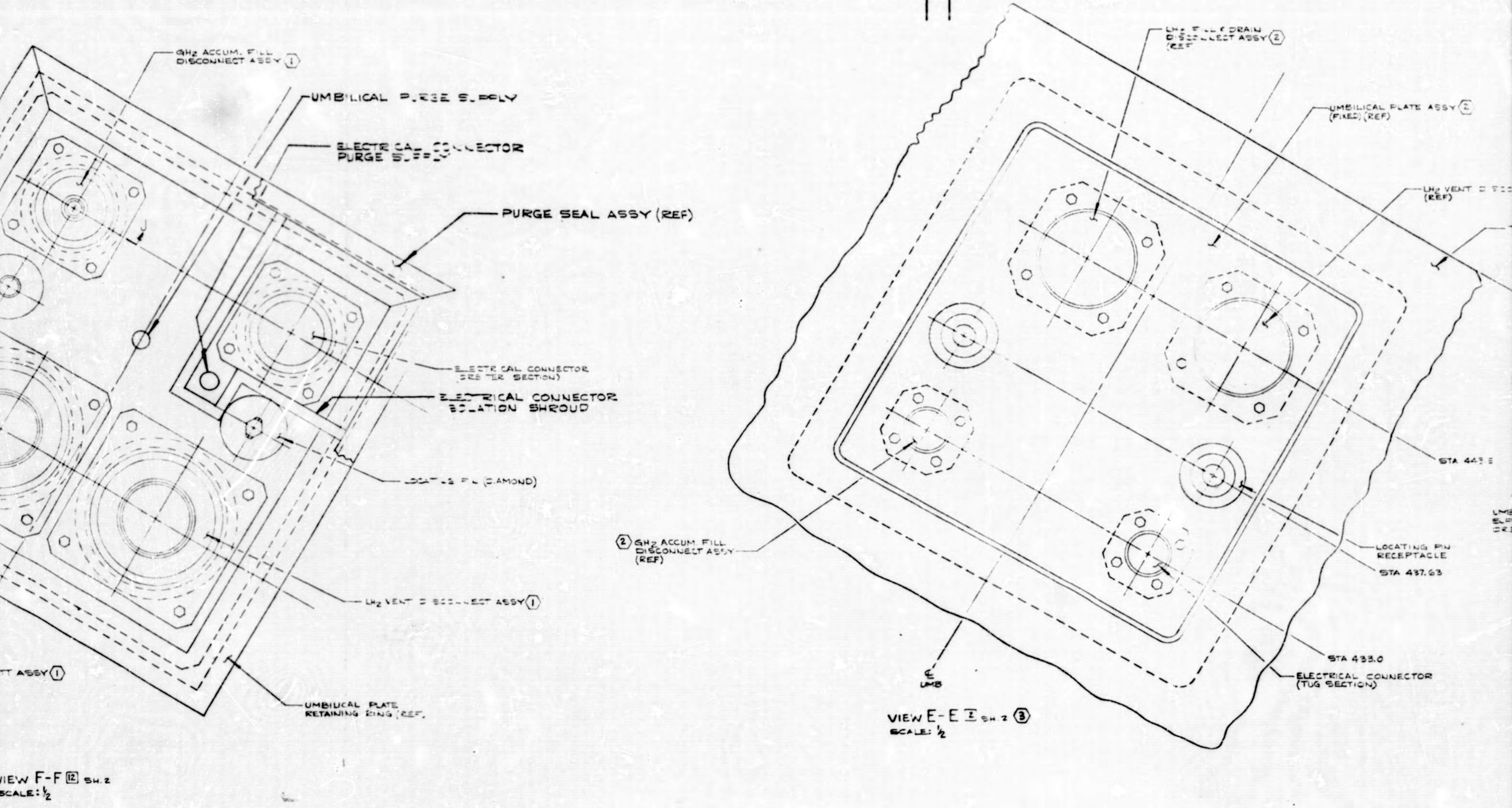
F-3


BALL BEARING
SUPPORT RING GUIDE RAIL
BALL BEARING RETAINING RAIL
SECTION L-L
SCALE: 1/2
UMBILICAL PLATE SUPPORT STRUCTURE (REF)
UMBILICAL PLATE SUPPORT RING (REF)
UMBILICAL PLATE RETRACTING LINKAGE ASSY (REF)
UMBILICAL PLATE ACTUATOR (REF)
VIEW H
SH. 2+3
SCALE: 1/2
32
31
30
29
28
27
D
C
B
A

F-1



26
25
24
23
22
21
UMBILICAL PLATE ASSY (FIXED)(REF)
PURGE SEAL ASSY
K
UMBILICAL PLATE ASSY (ADJUSTABLE) (REF)
VIEW K
SCALE: 1/1
GH2 ACCUM FILL FLEX LINE ASSY (REF)
UMBILICAL PLATE RETRACTING LINKAGE ASSY
SPUR GEAR & RACK ASSY (2 REQD)
UMBILICAL PLATE SUPPORT STRUCTURE
LH2 FILL & DRAIN FLEX LINE ASSY (REF)
LH2 VENT FLEX LINE ASSY (REF)
VIEW G-G 14 SH.2
SCALE: 1/2
J 21
LOCATION PIN (ROUND)
1 LH2 FILL & DRAIN DISCONNECT ASSY
UMBILICAL PLATE ASSY (ADJUSTABLE)
DISCONNECT RETAINING RING
UMBILICAL PLATE RETAINING RING
℄ UMB
DISCONNECT ASSY
UMBILICAL PLATE ASSY (ADJUSTABLE) (REF)
3 SECTION J-J
SCALE: 1/2
SCALE: 1/2
F-2


GH2 ACCUM. FILL DISCONNECT ASSY (1)
UMBILICAL PURGE SUPPLY
ELECTRICAL CONNECTOR PURGE SUPPLY
PURGE SEAL ASSY (REF)
ELECTRICAL CONNECTOR ISOLATION SHROUD
LOCATING PIN (DIAMOND)
LH2 VENT DISCONNECT ASSY (1)
UMBILICAL PLATE RETAINING RING (REF.
VIEW F-F SH. 2
SCALE: 1/2
LH2 FILL & DRAIN DISCONNECT ASSY (2) (REF
UMBILICAL PLATE ASSY (2) (FIXED) (REF)
(REF)
LOCATING PIN RECEPTACLE
STA 437.63
STA 433.0
ELECTRICAL CONNECTOR (TUG SECTION)
(2) GH2 ACCUM FILL DISCONNECT ASSY (REF)
℄ UMB
VIEW E-E SH. 2 (3)
SCALE: 1/2

F-3

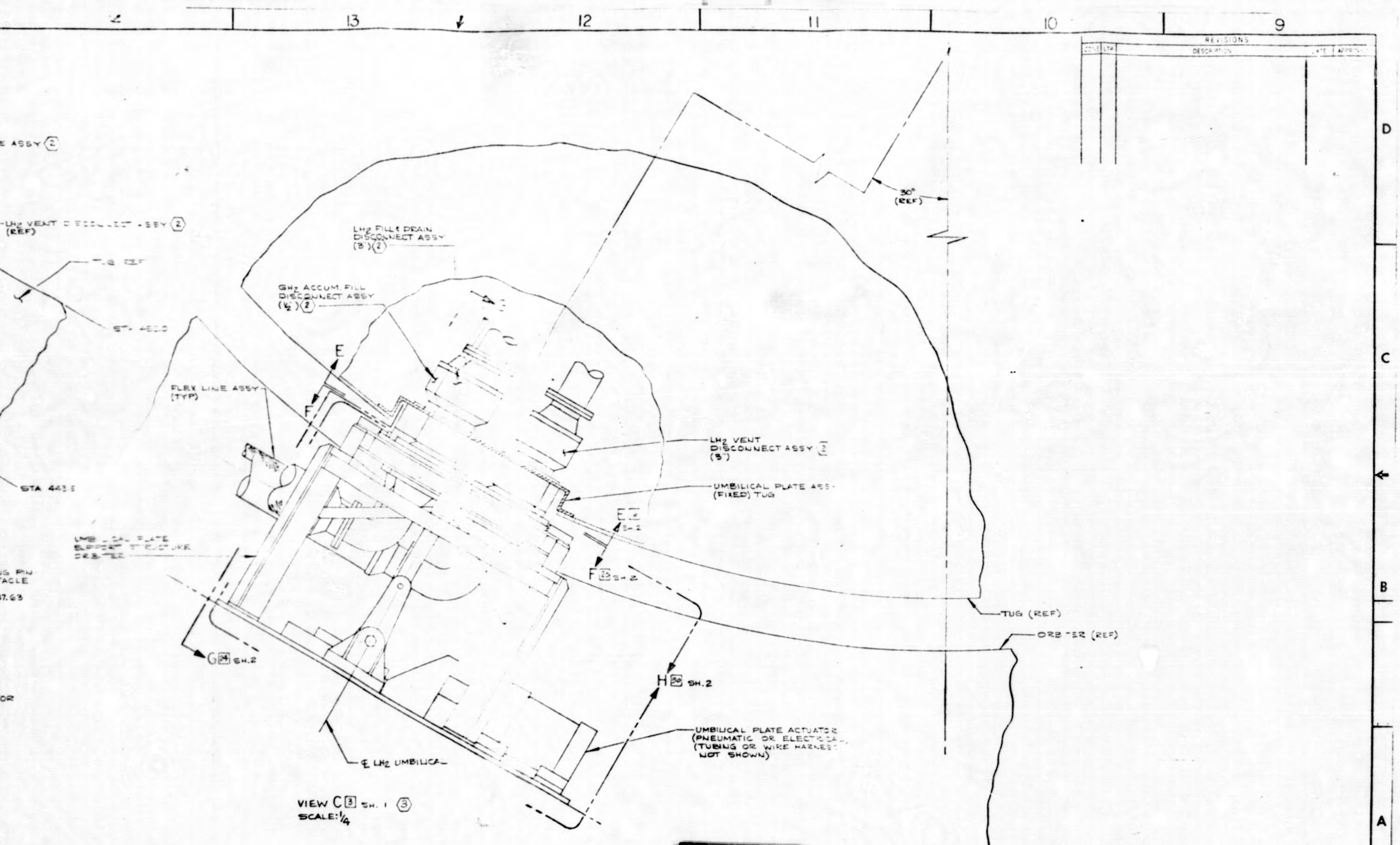


Figure 2.11-4 LH_2 Umbilical Plate Assembly

2-201,2-202

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-941514

SCALE NOTED — SHEET 2 OF 3

Table 2.11-1. LOX Panel Fluid and Electrical Connections

3"	-	LOX Fill and Drain	-	24 PSIA
2"	-	LOX Vent	-	24 PSIA
1/2"	-	APS LOX Accumulator Fill	-	1250 PSIA
1/2"	-	Thermodynamic Vent - LOX Tank	-	2 PSIA
1"	-	Helium Fill	-	3000 PSIA
Electrical Connector - 24 Shell Size				

Table 2.11-2. LH_2 Panel Fluid and Electrical Connections

3"	-	LH_2 Fill and Drain	-	24 PSIA
3"	-	LH_2 Vent	-	24 PSIA
1/2"	-	APS LH_2 Accumulator Fill	-	1250 PSIA
1/2"	-	Thermodynamic Vent - LH_2	-	2 PSIA
Electrical Connector - 24 Shell Size				

Table 2.11-3. Aft Panel Fluid and Electrical Connections

1"	-	Helium Purge	-	3000 PSIA
1/2"	-	Insulation Repressurization	-	15 PSIA
Electrical Connector - 24 Shell Size				

mating umbilicals move perpendicular to the Tug skin and are actuated by an electrically driven rack and pinion device. The capability for approximately nine inches of movement of the umbilical has been incorporated. This corresponds to an actuator movement of 3 inches. The umbilicals are guided at the four corners to minimize misalignment. Details of this mechanism are shown in Figures 2.11-3 and 2.11-4.

Panels similar in layout and size to the LOX and LH_2 panels will also be required on the orbiter skin for connection to the GSE when the Tug is within the cargo bay. Since no definite information was available for the configuration of the Orbiter, only the details of the panels on the Tug and the mating umbilicals within the Orbiter have been provided.

2.11.2 Disconnect Configuration

The disconnects utilized are similar to those used on the Saturn V Program except these do not have the positive latching "Dog & Collect" arrangement. The details of a typical fluid disconnect are shown in Figure 2.11-5. The details of the electrical disconnects will be found in Section 3.0 "Avionics Subsystems" of this volume.

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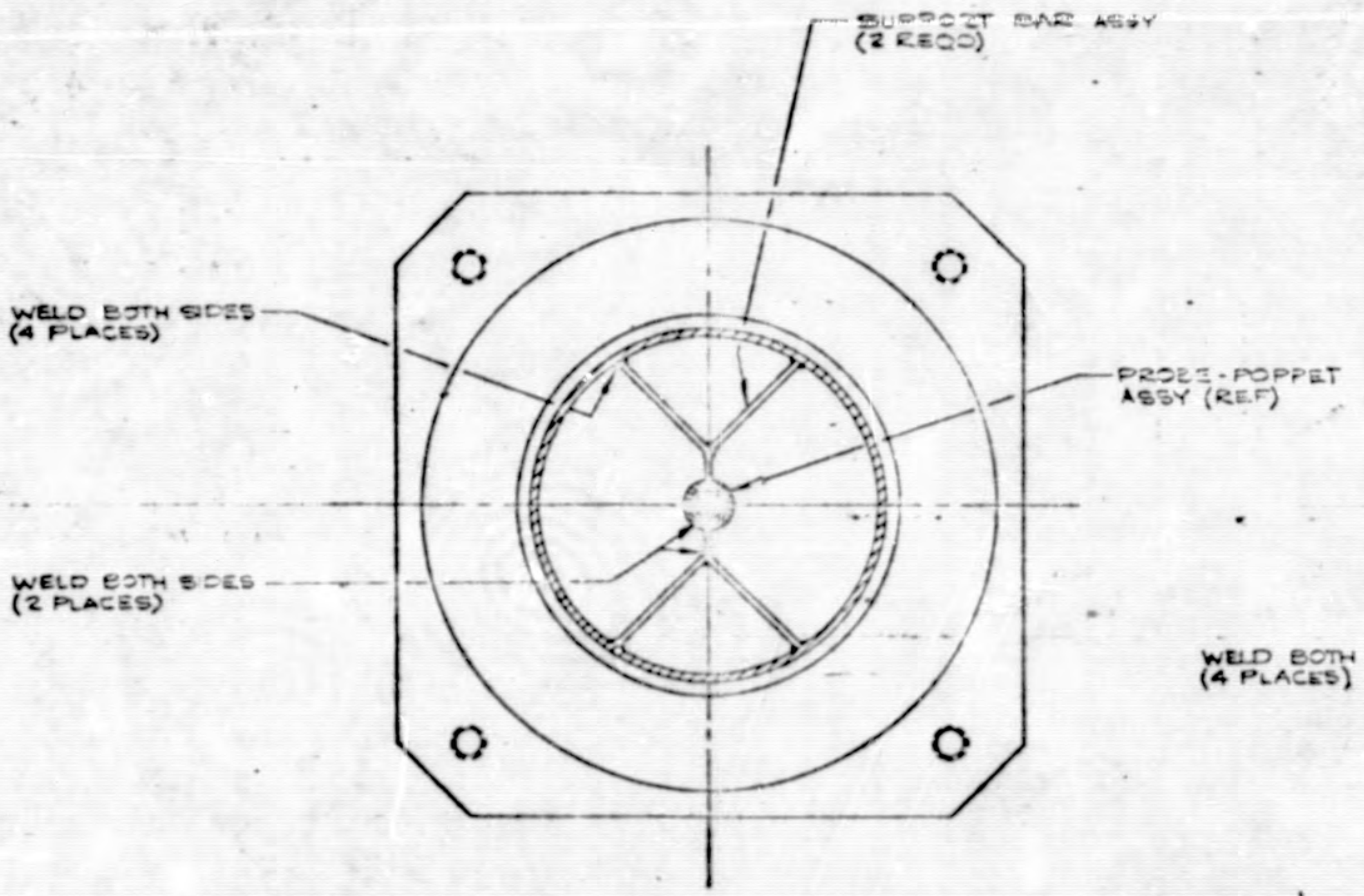
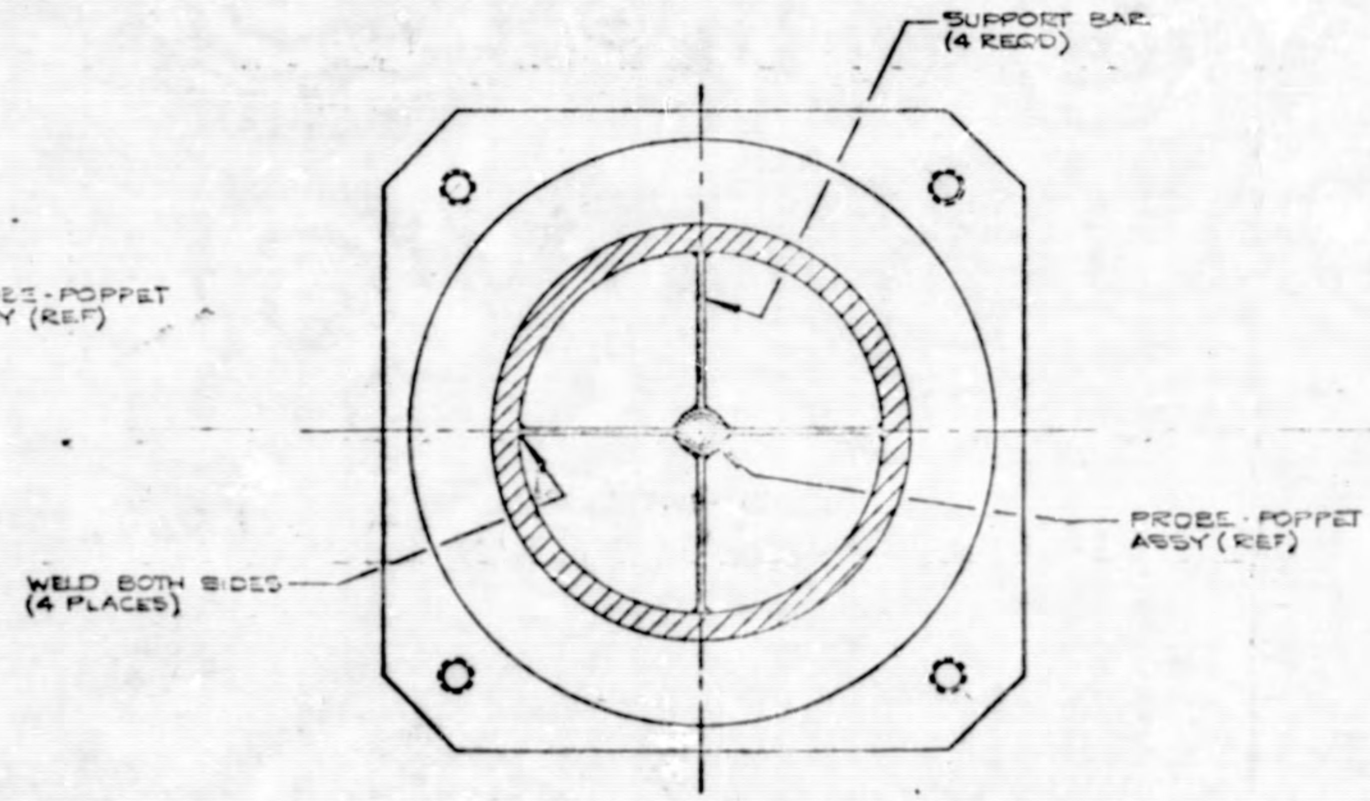
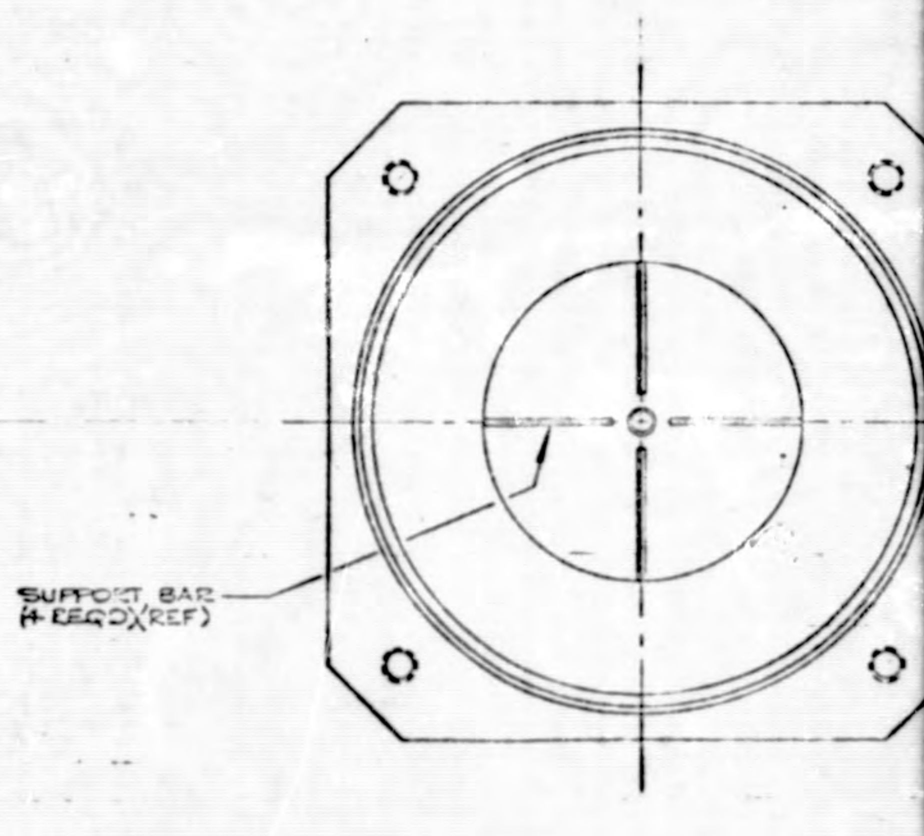


SECTION C-C $\boxed{4}$
SCALE: 1/1



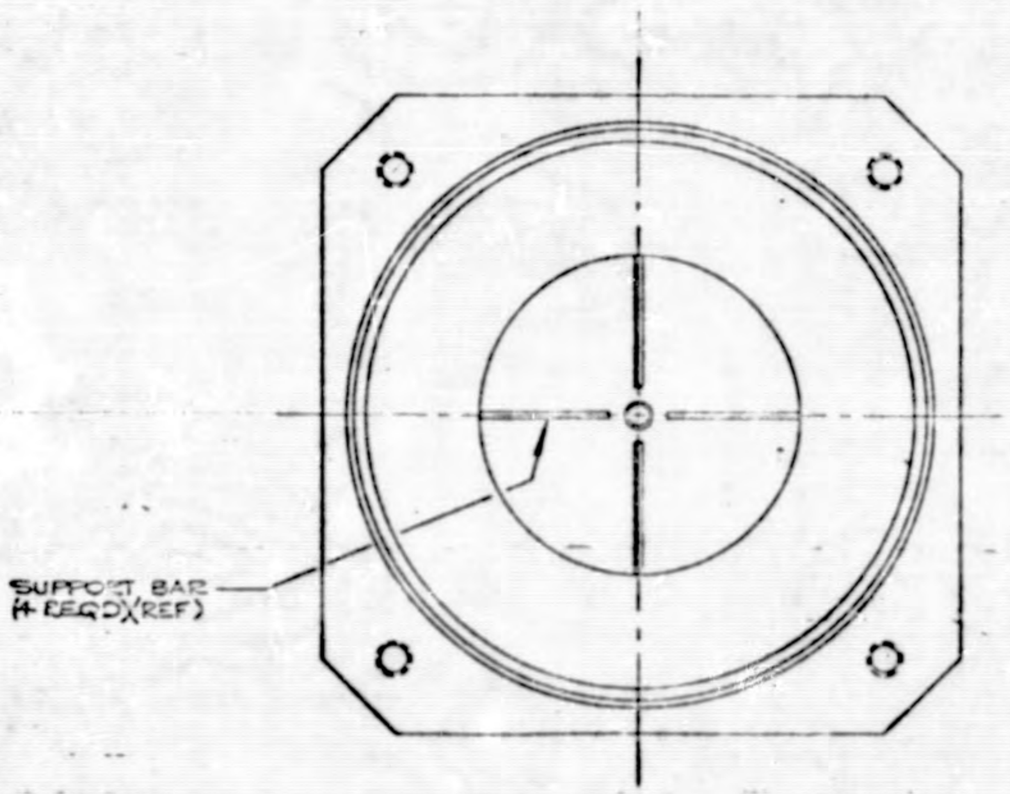
SECTION B-B $\boxed{5}$
SCALE: 1/1



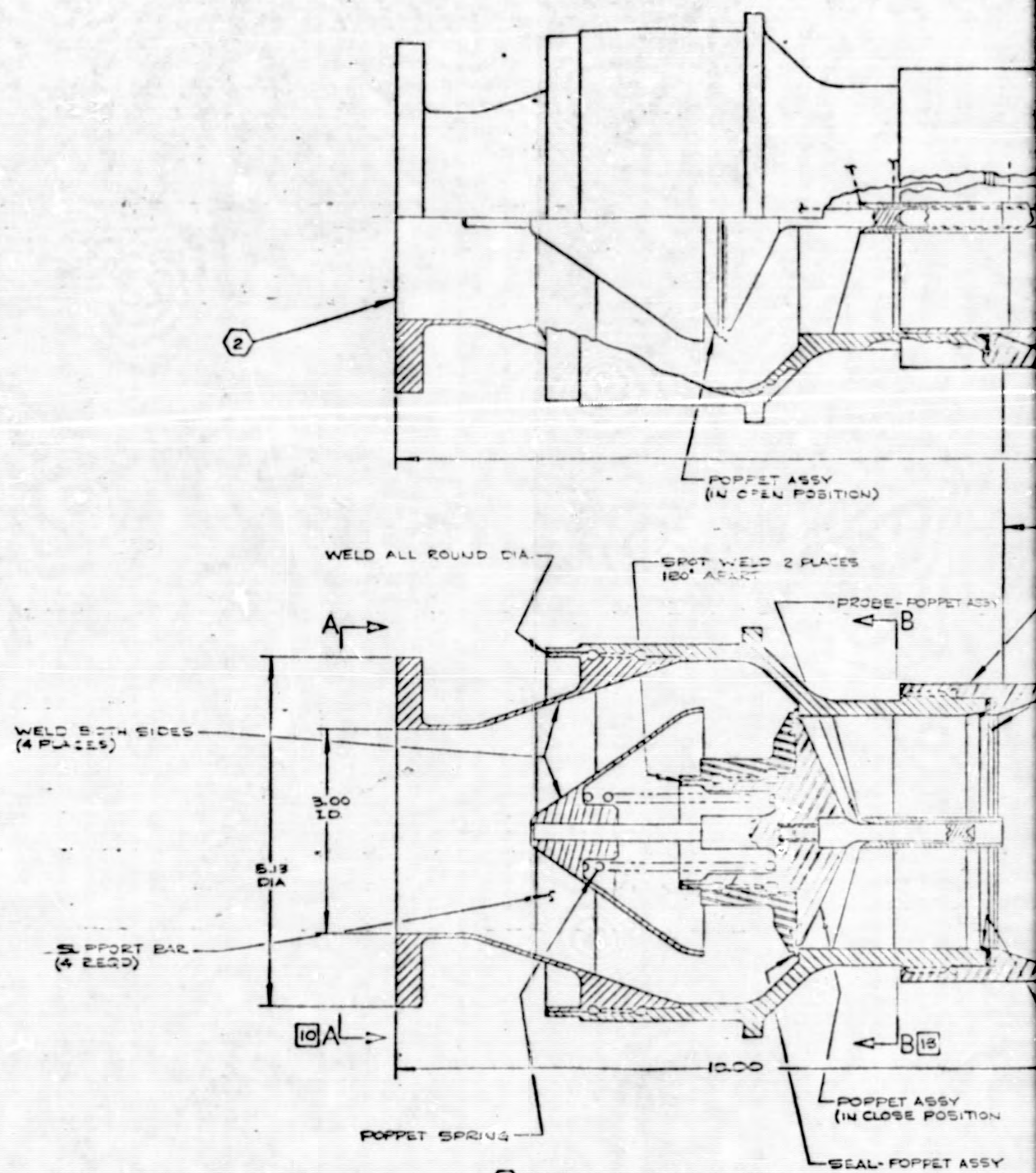
VIEW A-A $\boxed{7}$
SCALE: 1/1

F-1

11 10 9 8 7 6 5



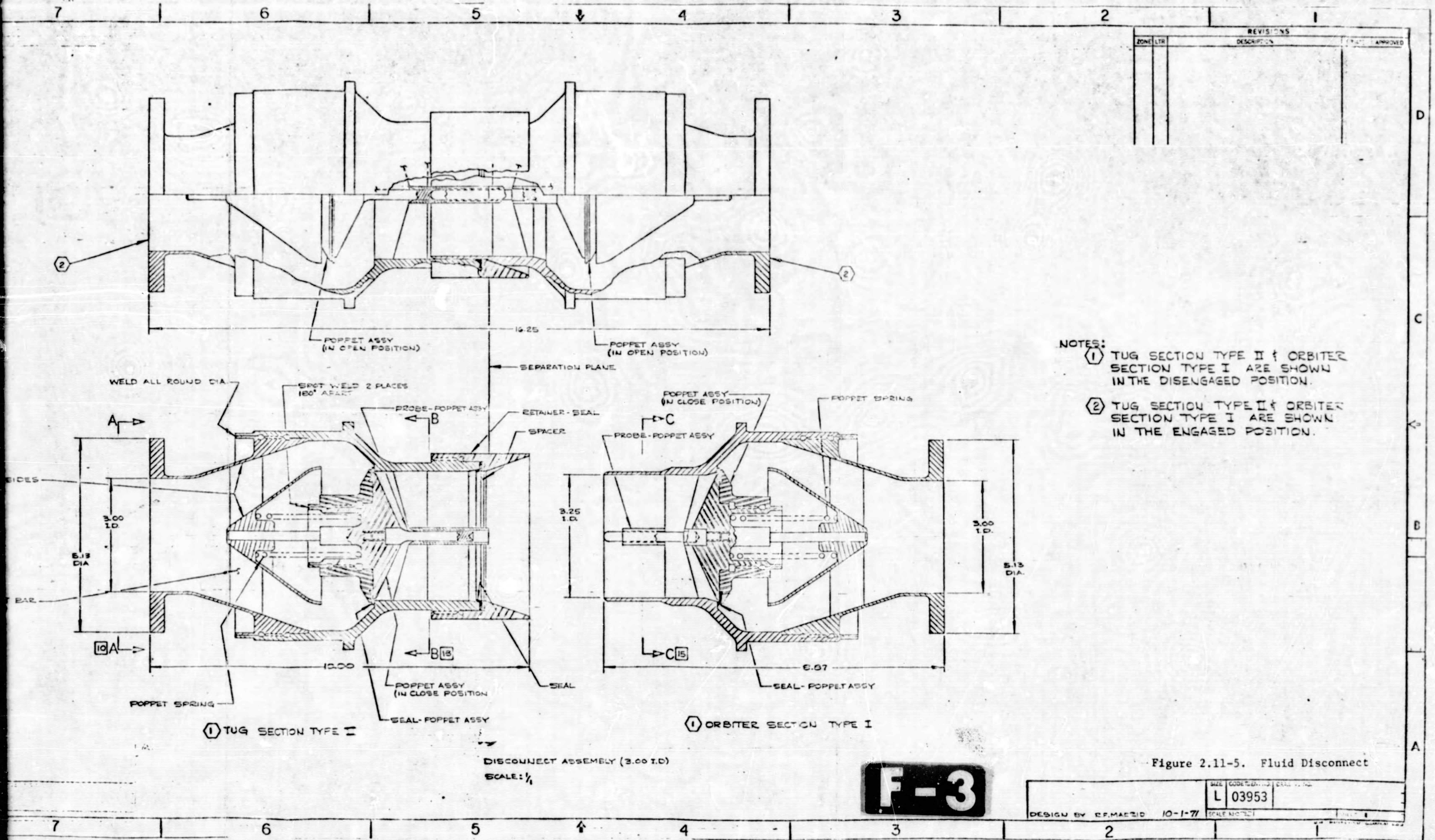
VIEW A-A [7]
SCALE: 1/1



(1) TUG SECTION TYPE I

F-2

11 10 9 8 7 6 5



NOTES:

1. TUG SECTION TYPE II & ORBITER SECTION TYPE I ARE SHOWN IN THE DISENGAGED POSITION.
2. TUG SECTION TYPE II & ORBITER SECTION TYPE I ARE SHOWN IN THE ENGAGED POSITION.

Figure 2.11-5. Fluid Disconnect

F-3

A dynamic (pressure sensitive) seal serves as the mating seal for the fluid disconnects. A highly polished probe engages the lip seal and any pressure increase provides a corresponding increase in the force applied between the lip seal and probe. This feature reduces leakage rates at the disconnect interface. Each half of the disconnect contains a shutoff poppet. After the probe engages the mating seal continued axial motion causes both poppets to move to a full open position allowing fluid flow. When the units are disengaged the poppets close prior to the probe disengaging from the mating seal. This feature eliminates hard poppet seating and reduces the amount of residual fluids trapped between the two halves of the disconnect.

2.11.3 Operation

For ground operations of the Tug outside the Orbiter, the LOX and LH_2 panels on the Tug skin provide the means of checking out and servicing the vehicle in the maintenance and refurbishment areas. For ground operations with the Tug inside the cargo bay, the interface with the GSE will be at the Orbiter skin.

Prior to deploying or removing the Tug from the cargo bay, the mating umbilicals of the Orbiter will be withdrawn from the panels at the Tug skin. The reverse operation is used when the Tug is returned to the cargo bay.

The disconnects on the aft panel are mated or demated with the matching disconnects on the support structure as the Tug is either attached or separated from the Orbiter. Misalignment of approximately the same magnitude as the LOX and LH_2 umbilicals is acceptable for the aft panel as it employs the same type of electrical and fluid disconnects.

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3.0 AVIONICS SUBSYSTEMS

The basic objective of the NR avionics design effort during the Tug Point Design Study was to define in detail a set of avionics subsystems possessing minimum weight characteristics and mission performance capability. The primary Tug mission is to deliver and retrieve geosynchronous orbit payloads. The Tug is an unmanned, ground-based vehicle that is itself delivered to and retrieved from low earth orbit by the Shuttle Orbiter.

The design developed during this study is based on the use of 1976 technology to achieve reduced component weights. To achieve reduced subsystem weights by minimizing the number of components, a simplex concept utilizing no redundancy beyond that required for fail-safe operation was implemented.

Control of the vehicle electrical, electronic, electro-optical, electro-mechanical, and electro-chemical components in the operational modes necessary to perform the Tug mission is provided by a single centralized general-purpose digital computer and remotely positioned digital multiplexing input/output terminals. This "data bus" approach results in a wire weight reduction while providing the capability for on-board checkout of vehicle systems. The Tug will be under computer control during all operational phases other than those occurring when the Tug is mated to the Shuttle Orbiter and when the Tug is docking with the payload. During those phases, control will be maintained by the Shuttle Orbiter or uplink override.

To enable the Tug to perform its mission, the avionics subsystem must be capable of performing the functions of: data management; guidance, navigation and control; rendezvous and docking; communications, instrumentation; and electrical power generation, conversion and distribution. The integration of those functions is shown in Figure 3.0-1 and discussed in the following paragraphs.

The Data Management Subsystem connects the vehicle subsystem components to the computer through the use of remotely located interface and data acquisition units. Intelligence exchange between the computer and these peripherals is in a coded digital format. A separate status and control panel is provided for Shuttle/Tug information transfer.

The Guidance, Navigation and Control Subsystem sensor outputs are transmitted via the remote data acquisition units to the computer where computations are made to determine vehicle position, velocity and attitude. This determination results in the proper sequencing of commands from the computer to the main engine and attitude control propulsion system. An independent back-up rate gyro and logic package is included to provide fail-safe capability.

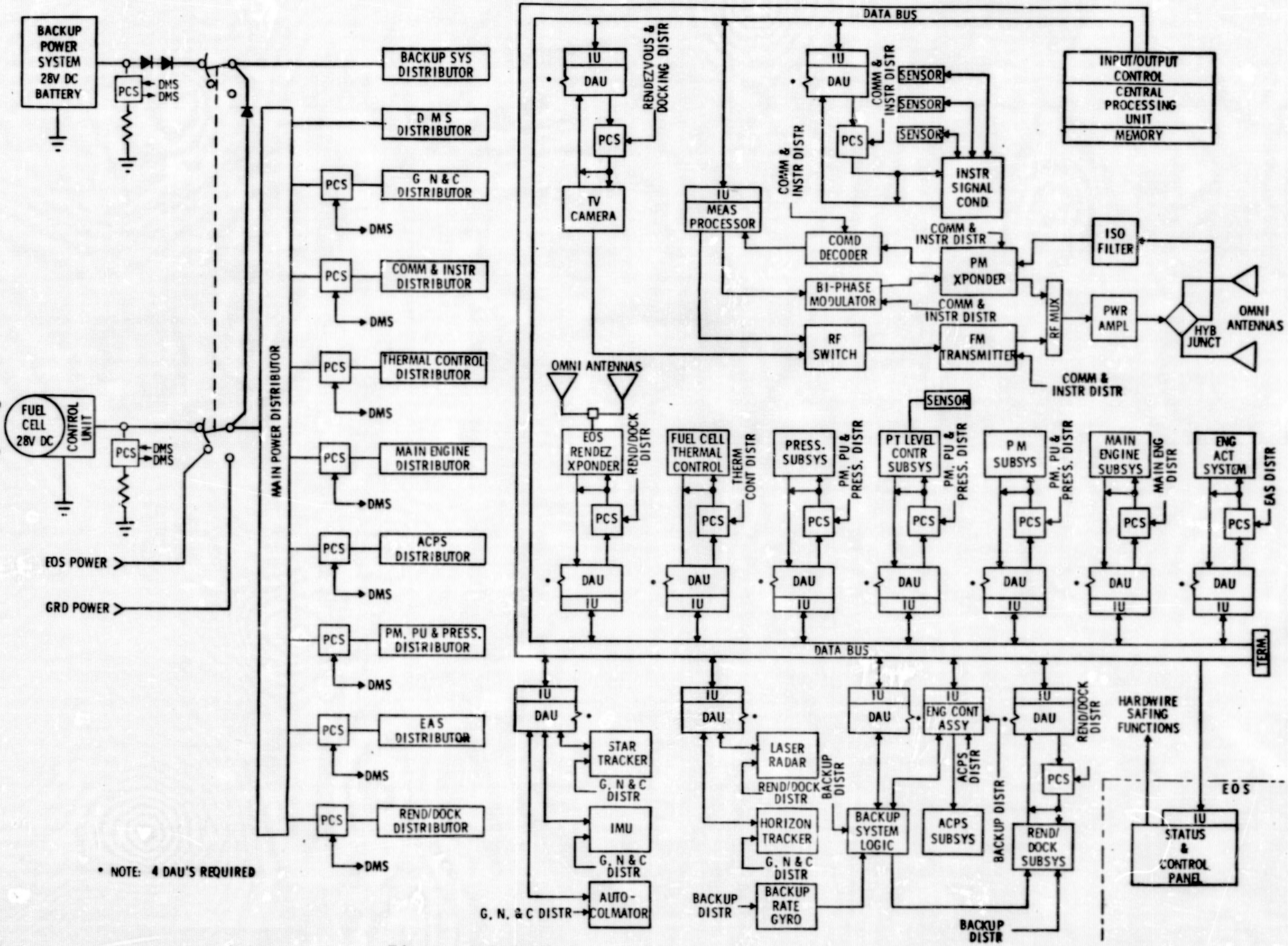


Figure 3.0-1 Avionics Subsystems Integrated Block Diagram

The Communication Subsystem interfaces with the computer through the use of a measurement processor and provides the capability for the Tug to communicate with both ground stations and the Shuttle. The Instrumentation Subsystem sensor signal conditioner outputs are interfaced directly into the remote data acquisition units to provide the computer measurements for determining system status.

The Rendezvous and Docking Subsystem is under control of the computer until final payload or Shuttle docking is initiated. For final payload docking, remote man control using TV is achieved; for final shuttle docking, the shuttle crew actively closes with the passively cooperative Tug.

The Electrical Power Generation, Conversion and Distribution Subsystem uses a 28 VDC fuel cell for primary power. A three position power transfer switch is used to connect fuel cell power, Shuttle power, or ground power to the Tug main power distributor. Computer controlled solid state switches distribute the power to the various subsystem load distributors. The design includes a 28 VDC battery to provide 30 minutes of vehicle attitude control in the event of fuel cell failure.

It should be pointed out that within this report, the level of detail reflected in each subsystem design resulted from a philosophy in approach that dictated the definition of each subsystem to the depth required to establish Tug mission performance capability and minimum weight objective achievement credibility. With such an approach, the level of subsystem design detail is directly related to the inherent complexity of the subsystem design (i.e.. more detail is required as complexity increases).

3.1 DATA MANAGEMENT

The Data Management Subsystem (DMS) provides the means of integrating, managing and controlling the various systems of the basically autonomous Tug vehicle. The subsystem design uses three major elements: (1) data management computer, (2) data bus distribution components, and (3) software. The functional relationships among components are shown in Figure 3.0-1,

The data management computer provides the centralized vehicle intelligence for control of subsystems and associated data. The data bus elements provide the communication link between the central computer and the vehicle subsystems, the Earth Orbital Shuttle (EOS) crew status and control panel, and ground via telemetry. The data bus elements consist of a twisted shielded pair transmission line, data acquisition units (DAU), interface units (IU), and measurement processor unit (MPU). The software element includes all computer programs for onboard checkout and flight sequencing of the vehicle subsystems.

The computer performs vehicle operational control in accordance with the programs contained in protected memory. In transmitting commands to the vehicle subsystems, the computer central processing unit identifies data to

be transmitted and initiates the outputing of data. The computer input/output processor outputs data to the data bus independently once the operation has been initiated. Data is placed on the data bus in a serial pulse code at a one MHz rate. The first word of the data transmission contains the address of the IU desired to receive the instructions contained in the message. If the command message requests data, the DAU routes the desired subsystem back to the computer via the IU and data bus. When the command message is an instruction to issue a subsystem control stimuli, the IU processes the data through to the DAU where it is decoded and the appropriate stimuli issued. At the time the stimuli is issued, it is self-tested and the go/no-go status fed back to the computer. The computer continues to monitor the function's operational parameters to verify proper system response to the command. If the desired response does not occur, action is taken to determine the cause and perform those actions required to continue system operation.

The subsystem provides the information and control necessary for the Communications Subsystem to transmit data to the ground. The MPU is loaded with the addresses of the desired data which is then retrieved each time it appears on the data bus. Continual and automatic update is provided to transmit current status to the ground upon preprogrammed command or as requested by the uplink.

The subsystem includes a status and control panel installed on the EOS to be utilized prior to Tug electrical demating for transmitting Tug status to the EOS crew, to transfer navigation state vector information, and to control critical Tug subsystems.

An onboard tape recorder is included in the subsystem to record the data stream from the main engine instrumentation system during main engine burns.

3.1.1 Requirements

The requirements influencing the design of the DMS hardware include mission requirements (functional and performance), and subsystem requirements (functional and performance). The requirements affecting the DMS software include computer storage requirements and computer speed requirements.

Mission Requirements

The primary requirement imposed upon the DMS by the Tug baseline mission is to provide a means of integrating, managing and controlling the various systems of the virtually autonomous Tug vehicle. This includes the mechanization required to perform the centralized functions utilized in vehicle sequencing and control, subsystem configuration management, and data management.

A significant mission oriented requirement was identified in the NASA Study Plan. That requirement is that the Tug be designed as a fail-safe vehicle, defined as one which has no failure mode which could cause the payload to be destroyed or a mode which could cause an unsafe situation for the EOS or its crew.

Subsystem Requirements

The NASA Study Plan established certain subsystem requirements as follows:

1. The avionics system baseline should include a digital multiplexing technique to interconnect the subsystems with the data management. This system is also utilized as a ground and on-orbit checkout scheme.

2. The Tug and payload will have their detailed checkouts performed prior to installation in the EOS. Only limited functional tests will be performed at the pad and in orbit while the Tug/payload are attached to the EOS. These functional checkouts will be self-tests by Tug and payload and will represent only a go/no-go status to the EOS crew via hardline or radio frequency (RF) link.

3. Provisions are to be made for monitoring of Tug critical functions for EOS crew safety by the EOS crew at all times the payload (Tug) is attached to the EOS.

*4. The DMS should have digital interface units with 32 channel capability.

5. The Tug will be provided a navigation update from the EOS prior to EOS/Tug separation.

6. The Tug will be designed for autonomous operation except in areas where remote control will significantly reduce weight without compromising operational effectiveness.

* The 32 channel digital interface unit capability was assumed to be a minimum capability for this study.

Software Requirements

The Tug flight software is composed of two basic elements: the Executive Control Program and the Subsystem Management Program. The Executive Control Program controls the sequence and order of execution of all programmed functions in the Subsystem Management Program.

The Executive Control Program is a group of interacting subprograms whose primary purpose is to control the sequencing and scheduling of the execution of individual subsystem program modules to meet the operational requirements of the Tug mission. It examines interrupts and priority tables to determine order of module execution and provides a means of transferring operational control to and from the program modules. The design and organization of the Executive Program is based upon the use of a set of master queue/system tables by which the application requirements of any given mission can be specified. The Executive Program by definition must control the flow of information, manage the available resources, and provide for flexibility and ease of use.

The Subsystem Management Program will provide for all computer controlled subsystem functions not provided for by the Executive Program. The program will perform all calculations and logical decisions required for each phase of the Tug system operation. The program will utilize information and common subprograms to the maximum extent consistent with program modularity concepts. The program services offered by the executive will be utilized in the execution of the subsystem application programs. The following functions as a minimum are included in these programs.

- Vehicle control under all mission conditions and phases
- All program initialization
- Powered and unpowered flight navigation
- Guidance
- Attitude Reference and Control
- Control of Main Propulsion and Auxiliary Propulsion
- Mission Sequencing

Computer Speed and Memory Requirements Analysis

The computer speed and memory requirements for the Tug were derived from the NR performed Expendable Second Stage study (ESS) and Shuttle Phase B study results and other existing computer programs with new estimates made for requirements peculiar to the Tug. To obtain these results the major requirement areas were broken into detail, with number of instructions and number of operations per second being established for each area. Using this information along with the following assumptions, the computer speed and memory requirements were identified.

- 50 instructions out of every 100 are half word 16 bit instructions.
- Data words are generally assumed to be half words except where 32 bit words are required for Guidance and Navigation.
- Total storage is expressed in 16 bit words.
- Speed requirements are identified in equivalent adds per second divided by 1000 (KADS).
- A fifteen percent contigency factor is included for storage and speed.
- The storage and speed estimates shown represent worst case analysis.

Tables 3.1-1 and 3.1-2 represent the preliminary storage and speed requirements needed to perform the software function herein identified for the Tug.

3.1.2 Subsystem Trades

Several trade studies were performed prior to and during the establishment of the DMS design concept. These trades were made to optimize the design within the guidelines of the NASA Study Plan. The trades included aspects of data traffic transmission techniques, computer allocations and capabilities, and hardware configurations. Conclusions for each of the studies made are summarized in the following sections.

Data Traffic Transmission Techniques

Results of the Tug study concerning hardware versus data bus systems and the review of similar studies conducted on other programs indicate the data bus sytem to be a lighter weight system. The major drawback to selecting the data bus approach on past programs has been the high risk factor associated with utilizing an unproven technique. In addition to the risk factor, the high initial development cost has placed an excessive burden on program cost. However, with advancements which are being made as new programs utilize data bus concepts to accomplish additional tasks, both the risk and cost factors are being reduced to an acceptable level. Thus, for the Tug design time frame, the data bus concept represents the lightest weight, the latest technology, and the most effective approach available to the Tug avionics system.

Centralized vs Dedicated Computer for Guidance, Navigation and Control (GN&C)

The analysis conducted to evaluate the feasibility of providing dedicated computational capability for various functions of the Tug avionics system resulted in the selection of a centralized system as the most practical approach. With the baselining of a simplex system, the computer load associated with redundancy management was reduced to a level where it was not effective to consider it a specific function. This along with establishing a requirement of minimum on-board checkout capability reduced the fault isolation programming requirement to a point where it became a normal function of vehicle sequencing and control. With the reduction in these two major functions, the primary task of the DMS computer became performance of GN&C functions. The computer load associated with vehicle sequencing and data management is not large enough to warrant dedicated computer hardware nor does it significantly increase the computer computational rate when compared with that of the GN&C functions. Thus it is not necessary to provide dedicated computational capabilities to the various vehicle functions in general and GN&C in particular.

Computer Selection

Because of the study limited time span, it was necessary to limit the evaluation of computer candidates to a minimum number of typical units. As a

Table 3.1-1. Computer Storage Requirements

(16 Bit Words)

Function	No. of Words
Executive	
Program Control	2600
Miscellaneous Service Routines	3100
Events Management	4400
Subsystem	
Status	3000
Vehicle Sequencing	7800
System Tables	3800
Data Management	3100
Guidance, Navigation and Flight Control	
Guidance, Navigation. and Control Computational Checks	3000
Incremental Propagation of Trajectories and Error Transition Matrices (Including Gravity Potential Function)	2700
Large Step Propagation of Trajectories	500
Determination of Required Velocity (Lambert Trajecting)	1300
Determination of Velocity to be Gained (Lambert Trajecting)	1100
Computation of Special Functions (Battin's Universal Orbit Equation)	500
Velocity Vector Perturbation	900
Star Catalog	200
Line of Sight to and Angle from the Sun	200
Star Tracker Pointing Commands	1400
Horizon Sensor Data Processing	500
Star Data Processing	900
Kalman Estimation	3000
Measurement Incorporation and State Vector Update	1500
Attitude Computation	500
Attitude Control	1600
Local Vertical Computations	500
Covariance Matrix From Error Transition Matrix	500
Injection Burn Determination	1400
Plane Change Burn Determination	900
Incorporation of Accelerometer Data	300
Cross Product Steering Including Thrust Vector Control	1900
Laser Radar Positioning Commands	700
Laser Radar Data Processing	500
Relative Position and Velocity During Docking	700
15% Margin	8200
Total	63,200

Table 3.1-2. Computer Speed Requirements

Program Segment	Equivalent Adds Per Second
Executive	15 K
Subsystem Control	16 K
Attitude Reference Computations	200 K
Attitude Control	6 K
Rate of Velocity To Be Gained	6 K
Cross Product Steering and TVC	16 K
Incorporation of Accelerometer Data	6 K
Powered Trajectory Nav. Computation	9 K
15 Margin Included	41 K
Total	316 K

result, three basic computer models were considered: (1) IBM Model AP-1 General Purpose Digital Computer, (2) Autonetics Model DM216 General Purpose Digital Computer, and (3) CDC Model 469 General Purpose Digital Computer. References 3.1-1, 3.1-2. and 3.1-3 identify the design concepts for the three candidate computers. Of the three candidates, only number (1) would require no development for the Tug application.

From an operational standpoint, considering performance capability only, the IBM AP-1 computer is the most desirable unit. This computer contains the largest repertoire of instructions and provides the largest capability for data manipulation and storage. However, it also represents the highest power consumption and is the heaviest machine. Because of its weight, the AP-1 computer was not chosen as the baseline computer.

The two mini-computers, Autonetics Model DM216 and the CDC Model 469, represent scaled down models of the AP-1 machine from a capabilities standpoint. However, the weight and power consumption of the mini-computers represent an extensive advantage over the AP-1 computer.

The mini-computers both are of the fixed point arithmetic design where the AP-1 computer offers a programmable option. A review of the Tug calculation requirement indicates either mode is acceptable. The fixed point mode was accepted for baselining primarily because it required less memory storage, it satisfied the program requirements and no additional requirements were placed upon the candidate mini-computer configurations. However, it should be noted that by the 1976 design time frame of the Tug vehicle the floating/fixed point option capability of the AP-1 computer may also be available in the smaller mini-computers and this option could be added to the baseline to lessen the complexity factor associated with programming a fixed point machine.

A 32 bit word length was baselined over a 16 bit word length because of the inefficiency in memory addressing that the 16 bit word imposes. It is often necessary to circumvent this limitation by the use of indirect addressing or double-word instructs both of which consume extra time and space. In addition, a 32 bit word length reduces the need for multiple precision data handling. This is particularly true in the large number of calculations associated with the Tug. Reference 3.1-4 provides the results of a detailed study covering this subject.

The DM216 computer was baselined over the Model 469 computer because of its larger repertoire of instructions, its direct memory addressing capability, and its 32 bit word capability.

This study did not eliminate the possibility of utilizing the CDC Model 469 computer. However, it did indicate that at this point in the definition of system design the CDC computer was marginal for satisfying program requirements. With more detailed definition of the program requirements, it may prove that this smaller machine is acceptable and could be included as the baseline unit.

Data Bus Hardware Selection

The study conducted to establish data bus hardware for baseline usage for the Tug DMS encompassed various techniques and types. These included the Data and Control Management concepts and hardware proposed in the NR Shuttle Phase B study (References 3.1-5 and 3.1-6 provide detailed definitions of hardware and concepts), the hardwired switch selector and data multiplexing techniques utilized of the Saturn Program, and the data bus concepts and hardware being developed for the B-1 Program. The study resulted in the selection of the data bus hardware being developed for the B-1 Program (Reference 3.1-7 provides a detailed description of the hardware and concepts). This hardware design has incorporated many of the capabilities specified in the Shuttle Phase B study. Its basic design utilized self verification as a design driver and represents hardware under development as opposed to hardware proposed for development. With a large portion of the development being accomplished prior to Tug usage, it represents cost effective hardware. Also, since this hardware is being developed for a long life program, its designed life time and reliability should be beneficial to the Tug program.

3.1.3 Subsystem Operation

The DMS provides the processing and/or computing functions required for operation and checkout of the Tug on-board systems. To accomplish these functions, the following capabilities are provided:

A. Configuration and Sequence Control - This function will provide control of subsystem configurations and sequencing of subsystem operations and and modes for both normal and contingent operations. In abnormal cases, initiation will be implemented by uplink data from the EOS or ground. Under normal conditions, all operations will be automatically initiated by preprogrammed vehicle mission sequences or as a result of the checkout and fault isolation function.

B. Guidance and Navigation - This function will process the appropriate guidance and navigation algorithms for each mission phase.

C. Flight Control - This function receives data from the guidance and navigation function and processes it to generate appropriate vehicle maneuvering commands to the vehicle control subsystems.

D. Data Storage - This function provides for core storage of flight and ground programs, critical flight data, system performance data, and vehicle configuration status.

E. Data Acquisition and Distribution - This function provides for the intercommunication of measurements, subsystem outputs and control signals between the DMS and other vehicle subsystem elements.

F. Checkout and Fault Isolation - This function will maintain continued monitoring of vehicle subsystem performance. It shall determine the subsystem functional status and in the event of a failure, initiate those actions necessary to continue system operation.

G. Self-Test and Status - This function shall provide verification of proper DMS operation and initiate system reconfiguration when required.

H. Recording - This function provides the capability of recording engine performance data on magnetic tape during each of the main engine burns.

Command and Response Functions

A preliminary analysis of the Tug subsystems control needs resulted in the identification of the commands and responses listed in Table 3.1-3. The listing is divided into a command and response section for each vehicle subsystem requiring an interface with the DMS. The selection of a specific DAU for outputing each command and inputing each response as shown in the table represents a preliminary effort taken to verify adequacy of the number of DAU's established for the baseline. A rigorous optimization of input/output channelization to minimize wire runs and separate control for redundant components was not possible within the study time frame.

Table 3.1-3 identifies the DAU's as A1, A2, F1, and F2. The location on the vehicle of each is as follows (Reference Figure 3.8-1):

A1 - Located on aft skirt Panel A1.
A2 - Located on aft skirt Panel A4.
F1 - Located on forward skirt Panel F3.
F2 - Located on forward skirt Panel F6.

The serial - digital commands and responses are identified as S/D in Table 3.8-1 while analog responses are identified by the letter A.

Table 3.1-3. Command and Response List

MAIN PROPULSION

DAU	COMMANDS
A1(S/D)	DATA BUS COMMAND DATA
	1) START
	2) THRUST
	3) MIXTURE RATIO
	4) SHUTDOWN

DAU	RESPONSES
A1(S/D)	DATA BUS RESPONSE DATA
	1) KEY ENGINE PARAMETERS
	2) CONTROL SYSTEM COMPONENT FAILURE INFORMATION
A1(S/D)	TAPE RECORDER DATA

PROPELLANT FEED, FILL & DRAIN

DAU	COMMANDS	
F1	LH_2 PREVALVE	OPEN
F1	LH_2 PREVALVE	CLOSE
A1	LOX PREVALVE	OPEN
A1	LOX PREVALVE	CLOSE
F1	LH_2 AUX PROP TANK T/D VENT CONTROL VALVE	OPEN
F1	LH_2 MPS & APS FEEDLINE T/D VENT CONTROL VALVE	OPEN
F1	LH_2 MPS & APS FEEDLINE T/D VENT OUTLET CONTROL VALVE	OPEN
A1	LOX AUX PROP TANK T/D VENT CONTROL VALVE	OPEN
A1	LOX APS FEEDLINE T/D VENT CONTROL VALVE	OPEN
A1	LOX MPS FEEDLINE T/D VENT CONTROL VALVE	OPEN
F1	LH_2 AUX PROP TANK VENT CONTROL VALVE	OPEN
A1	LOX AUX PROP TANK VENT CONTROL VALVE	OPEN

DAU	RESPONSES		
F1	LH_2 FILL & DRAIN VALVE NO. 1	OPEN PCS	ON/OFF
F1	LH_2 FILL & DRAIN VALVE NO. 1	CLOSE PCS	ON/OFF
F1	LH_2 FILL & DRAIN VALVE NO. 1	OPEN	
F1	LH_2 FILL & DRAIN VALVE NO. 1	CLOSED	
F2	LH_2 FILL & DRAIN VALVE NO. 2	OPEN PCS	ON/OFF
F2	LH_2 FILL & DRAIN VALVE NO. 2	CLOSE PCS	ON/OFF

Table 3.1-3. Command and Response List (Cont)

PROPELLANT FEED, FILL & DRAIN (Cont)

DAU	RESPONSES (Cont)		
F2	LH_2 FILL & DRAIN VALVE NO. 2	OPEN	
F2	LH_2 FILL & DRAIN VALVE NO. 2	CLOSED	
A1	LOX FILL & DRAIN VALVE NO. 1	OPEN PCS	ON/OFF
A1	LOX FILL & DRAIN VALVE NO. 1	CLOSE PCS	ON/OFF
A1	LOX FILL & DRAIN VALVE NO. 1	OPEN	
A1	LOX FILL & DRAIN VALVE NO. 1	CLOSED	
A2	LOX FILL & DRAIN VALVE NO. 2	OPEN PCS	ON/OFF
A2	LOX FILL & DRAIN VALVE NO. 2	CLOSE PCS	ON/OFF
A2	LOX FILL & DRAIN VALVE NO. 2	OPEN	
A2	LOX FILL & DRAIN VALVE NO. 2	CLOSED	
F1	LH_2 PREVALVE OPEN PCS	ON/OFF	
F1	LH_2 PREVALVE CLOSE PCS	ON/OFF	
F1	LH_2 PREVALVE OPEN		
F1	LH_2 PREVALVE CLOSED		
A1	LOX PREVALVE OPEN PCS	ON/OFF	
A1	LOX PREVALVE CLOSE PCS	ON/OFF	
A1	LOX PREVALVE OPEN		
A1	LOX PREVALVE CLOSED		
F1	LH_2 AUX PROP TANK T/D VENT CONTROL VALVE OPEN PCS	ON/OFF	
F1	LH_2 MPS & APS FEEDLINE T/D VENT CONTROL VALVE OPEN PCS	ON/OFF	
F1	LH_2 MPS & APS FEEDLINE T/D VENT OUTLET CONTROL VALVE OPEN PCS	ON/OFF	
A1	LOX AUX PROP TANK T/D VENT CONTROL VALVE OPEN PCS	ON/OFF	
A1	LOX APS FEEDLINE T/D VENT CONTROL VALVE OPEN PCS	ON/OFF	
A1	LOX MPS FEEDLINE T/D VENT CONTROL VALVE OPEN PCS	ON/OFF	
F1	LH_2 AUX PROP TANK FILL VALVE OPEN		
F1	LH_2 AUX PROP TANK FILL VALVE CLOSED		
A1	LOX AUX PROP TANK FILL VALVE OPEN		
A1	LOX AUX PROP TANK FILL VALVE CLOSED		
F1	LH_2 AUX PROP TANK VENT CONTROL VALVE OPEN PCS	ON/OFF	
A1	LOX AUX PROP TANK VENT CONTROL VALVE OPEN PCS	ON/OFF	

Table 3.1-3. Command and Response List (Cont)

PRESSURIZATION

DAU	COMMANDS	
F1	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN
F1	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 1	CLOSE
F2	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN
F2	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 2	CLOSE
A1	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN
A1	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 1	CLOSE
A2	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN
A2	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 2	CLOSE

DAU	RESPONSES	
F1	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN PCS ON/OFF
F1	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 1	CLOSE PCS ON/OFF
F1	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN/CLOSED
F2	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN PCS ON/OFF
F2	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 2	CLOSE PCS ON/OFF
F2	LH_2 TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN/CLOSED
A1	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN PCS ON/OFF
A1	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 1	CLOSE PCS ON/OFF
A1	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 1	OPEN/CLOSED
A2	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN PCS ON/OFF
A2	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 2	CLOSE PCS ON/OFF
A2	LOX TANK PREPRESSURIZATION CONTROL VALVE NO. 2	OPEN/CLOSED

Table 3.1-3. Command and Response List (Cont)

PROPELLANT ORIENTATION

DAU	COMMANDS	
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS POWER ON	
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE WET ON	
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY ON	
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN ON	
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS POWER ON	
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE WET ON	
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY ON	
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN ON	

DAU	RESPONSES	
F2	LH_2 AUX PROP TANK LEVEL SENSOR NO. 1 DRY	
F2	LH_2 AUX PROP TANK LEVEL SENSOR NO. 1 OPEN	
F2	LH_2 AUX PROP TANK LEVEL SENSOR NO. 2 DRY	
F2	LH_2 AUX PROP TANK LEVEL SENSOR NO. 2 OPEN	
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS POWER ON PCS	ON/OFF
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE WET PCS	ON/OFF
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY PCS	ON/OFF
F2	LH_2 AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN PCS	ON/OFF
A2	LOX AUX PROP TANK LEVEL SENSOR NO. 1 DRY	
A2	LOX AUX PROP TANK LEVEL SENSOR NO. 1 OPEN	
A2	LOX AUX PROP TANK LEVEL SENSOR NO. 2 DRY	
A2	LOX AUX PROP TANK LEVEL SENSOR NO. 2 OPEN	
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS POWER ON PCS	ON/OFF
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE WET PCS	ON/OFF
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY PCS	ON/OFF
A2	LOX AUX PROP TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN PCS	ON/OFF

Table 3.1-3. Command and Response List (Cont)

SAFING & VENTING

DAU	COMMANDS	
F1	LH_2 TANK RELIEF VALVE NO. 1 LOW MODE	ON
F1	LH_2 TANK RELIEF VALVE NO. 1 LOW MODE	OFF
F2	LH_2 TANK RELIEF VALVE NO. 2 LOW MODE	ON
F2	LH_2 TANK RELIEF VALVE NO. 2 LOW MODE	OFF
A1	LOX TANK RELIEF VALVE NO. 1 LOW MODE	ON
A1	LOX TANK RELIEF VALVE NO. 1 LOW MODE	OFF
A2	LOX TANK RELIEF VALVE NO. 2 LOW MODE	ON
A2	LOX TANK RELIEF VALVE NO. 2 LOW MODE	OFF
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1	OPEN
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1	CLOSE
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2	OPEN
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2	CLOSE
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1	OPEN
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1	CLOSE
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2	OPEN
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2	CLOSE

DAU	RESPONSES	
F1	LH_2 TANK RELIEF VALVE NO. 1 LOW MODE ON PCS	ON/OFF
F1	LH_2 TANK RELIEF VALVE NO. 1 LOW MODE OFF PCS	ON/OFF
F1	LH_2 TANK RELIEF VALVE NO. 1 OPEN	
F1	LH_2 TANK RELIEF VALVE NO. 1 CLOSED	
F2	LH_2 TANK RELIEF VALVE NO. 2 LOW MODE ON PCS	ON/OFF
F2	LH_2 TANK RELIEF VALVE NO. 2 LOW MODE OFF PCS	ON/OFF
F2	LH_2 TANK RELIEF VALVE NO. 2 OPEN	
F2	LH_2 TANK RELIEF VALVE NO. 2 CLOSED	
A1	LOX TANK RELIEF VALVE NO. 1 LOW MODE ON PCS	ON/OFF
A1	LOX TANK RELIEF VALVE NO. 1 LOW MODE OFF PCS	ON/OFF
A1	LOX TANK RELIEF VALVE NO. 1 OPEN	
A1	LOX TANK RELIEF VALVE NO. 1 CLOSED	
A2	LOX TANK RELIEF VALVE NO. 2 LOW MODE ON PCS	ON/OFF
A2	LOX TANK RELIEF VALVE NO. 2 LOW MODE OFF PCS	ON/OFF
A2	LOX TANK RELIEF VALVE NO. 2 OPEN	
A2	LOX TANK RELIEF VALVE NO. 2 CLOSED	
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1 OPEN PCS	ON/OFF
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1 CLOSE PCS	ON/OFF
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1 OPEN	
F1	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 1 CLOSED	

Table 3.1-3. Command Response List (Cont)

SAFING & VENTING (Cont)

DAU	RESPONSES (Cont)	
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2 OPEN PCS	ON/OFF
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2 CLOSE PCS	ON/OFF
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2 OPEN	
F2	GH_2 NONPROPULSIVE VENT CONTROL VALVE NO. 2 CLOSED	
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1 OPEN PCS	ON/OFF
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1 CLOSE PCS	ON/OFF
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1 OPEN	
A1	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 1 CLOSED	
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2 OPEN PCS	ON/OFF
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2 CLOSE PCS	ON/OFF
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2 OPEN	
A2	GOX NONPROPULSIVE VENT CONTROL VALVE NO. 2 CLOSED	
F1	LH_2 TANK VENT SELECTOR VALVE NO. 1 OPEN PCS	ON/OFF
F1	LH_2 TANK VENT SELECTOR VALVE NO. 1 CLOSE PCS	ON/OFF
F1	LH_2 TANK VENT SELECTOR VALVE NO. 1 OPEN	
F1	LH_2 TANK VENT SELECTOR VALVE NO. 1 CLOSED	
F2	LH_2 TANK VENT SELECTOR VALVE NO. 2 OPEN PCS	ON/OFF
F2	LH_2 TANK VENT SELECTOR VALVE NO. 2 CLOSE PCS	ON/OFF
F2	LH_2 TANK VENT SELECTOR VALVE NO. 2 OPEN	
F2	LH_2 TANK VENT SELECTOR VALVE NO. 2 CLOSED	
A1	LOX TANK VENT SELECTOR VALVE NO. 1 OPEN PCS	ON/OFF
A1	LOX TANK VENT SELECTOR VALVE NO. 1 CLOSE PCS	ON/OFF
A1	LOX TANK VENT SELECTOR VALVE NO. 1 OPEN	
A1	LOX TANK VENT SELECTOR VALVE NO. 1 CLOSED	
A2	LOX TANK VENT SELECTOR VALVE NO. 2 OPEN PCS	ON/OFF
A2	LOX TANK VENT SELECTOR VALVE NO. 2 CLOSE PCS	ON/OFF
A2	LOX TANK VENT SELECTOR VALVE NO. 2 OPEN	
A2	LOX TANK VENT SELECTOR VALVE NO. 2 CLOSED	
F1	LH_2 TANK VENT VALVE OPEN PCS	ON/OFF
F1	LH_2 TANK VENT VALVE CLOSE PCS	ON/OFF
F1	LH_2 TANK VENT VALVE OPEN	
F1	LH_2 TANK VENT VALVE CLOSED	
A1	LOX TANK VENT VALVE OPEN PCS	ON/OFF
A1	LOX TANK VENT VALVE CLOSE PCS	ON/OFF
A1	LOX TANK VENT VALVE OPEN	
A1	LOX TANK VENT VALVE CLOSED	
F2	LH_2 TANK HELIUM PURGE CONTROL VALVE OPEN PCS	ON/OFF
F2	LH_2 TANK HELIUM PURGE CONTROL VALVE OPEN	
F2	LH_2 TANK HELIUM PURGE CONTROL VALVE CLOSED	
A2	LOX TANK HELIUM PURGE CONTROL VALVE OPEN PCS	ON/OFF
A2	LOX TANK HELIUM PURGE CONTROL VALVE OPEN	
A2	LOX TANK HELIUM PURGE CONTROL VALVE CLOSED	

Table 3.1-3. Command and Response List (Cont)

PROPELLANT MANAGEMENT

COMMANDS

DAU	
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS POWER ON
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE WET ON
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY ON
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN ON
A2	LOX TANK LEVEL SENSOR CONTROLLERS POWER ON
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE WET ON
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY ON
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN ON

RESPONSES

DAU		
F2	LH$_2$ TANK LEVEL SENSOR NO. 1 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 1 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 2 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 2 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 3 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 3 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 4 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 4 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 5 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 5 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 6 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 6 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 7 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 7 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 8 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 8 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR NO. 9 DRY	
F2	LH$_2$ TANK LEVEL SENSOR NO. 9 OPEN	
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS POWER ON PCS	ON/OFF
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE WET PCS	ON/OFF
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY PCS	ON/OFF
F2	LH$_2$ TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN PCS	ON/OFF
A2	LOX TANK LEVEL SENSOR NO. 1 DRY	
A2	LOX TANK LEVEL SENSOR NO. 1 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 2 DRY	
A2	LOX TANK LEVEL SENSOR NO. 2 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 3 DRY	
A2	LOX TANK LEVEL SENSOR NO. 3 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 4 DRY	
A2	LOX TANK LEVEL SENSOR NO. 4 OPEN	

Table 3.1-3. Command and Response List (Cont)

PROPELLANT MANAGEMENT (Cont)

RESPONSES (Cont)

DAU		
A2	LOX TANK LEVEL SENSOR NO. 5 DRY	
A2	LOX TANK LEVEL SENSOR NO. 5 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 6 DRY	
A2	LOX TANK LEVEL SENSOR NO. 6 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 7 DRY	
A2	LOX TANK LEVEL SENSOR NO. 7 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 8 DRY	
A2	LOX TANK LEVEL SENSOR NO. 8 OPEN	
A2	LOX TANK LEVEL SENSOR NO. 9 DRY	
A2	LOX TANK LEVEL SENSOR NO. 9 OPEN	
A2	LOX TANK LEVEL SENSOR CONTROLLERS POWER ON PCS	ON/OFF
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE WET PCS	ON/OFF
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE DRY PCS	ON/OFF
A2	LOX TANK LEVEL SENSOR CONTROLLERS SIMULATE OPEN PCS	ON/OFF

THRUST VECTOR CONTROL

COMMANDS

DAU	
A1	TVC PUMP MOTOR START
A1	TVC PUMP MOTOR STOP
A1	TVC HEATER ON

RESPONSES

DAU		
A1	TVC PUMP MOTOR START PCS	ON/OFF
A1	TVC PUMP MOTOR STOP	
A1	TVC HEATER ON PCS	ON/OFF

Table 3.1-3. Command and Response List (Cont)

AUXILIARY PROPULSION

DAU	COMMANDS
A1	LH_2 SYSTEM BLEED VALVE OPEN
A1	LH_2 SYSTEM BLEED VALVE CLOSE
A2	LOX SYSTEM BLEED VALVE OPEN
A2	LOX SYSTEM BLEED VALVE CLOSE
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO FUEL CELL
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO VENT
A2	GOX FUEL CELL/APS SELECTOR VALVE TO FUEL CELL
A2	GOX FUEL CELL/APS SELECTOR VALVE TO VENT
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE OPEN
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE CLOSE
A2	GAS GENERATOR GOX SYSTEM CONTROL VALVE OPEN
A2	GAS GENERATOR GOX SYSTEM CONTROL VALVE CLOSE
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE
A1	GH_2 VENT VALVE OPEN
A1	GH_2 VENT VALVE CLOSE
A2	GOX VENT VALVE OPEN
A2	GOX VENT VALVE CLOSE
A1	LH_2 HEAT EXCHANGER GAS GENERATOR IGNITER ON
A1	LH_2 TURBOPUMP GAS GENERATOR IGNITER ON
A2	LOX HEAT EXCHANGER GAS GENERATOR IGNITER ON
A2	LOX TURBOPUMP GAS GENERATOR IGNITER ON
A1	GH_2 REGULATOR OUTPUT PRESSURE SWITCH POWER ON
A1	GH_2 ACCUMULATOR PRESSURE SWITCH POWER ON
A2	GOX REGULATOR OUTPUT PRESSURE SWITCH POWER ON
A2	GOX ACCUMULATOR PRESSURE SWITCH POWER ON
A1	LH_2 TURBOPUMP BYPASS VALVE OPEN
A1	LH_2 TURBOPUMP BYPASS VALVE CLOSE
A2	LOX TURBOPUMP BYPASS VALVE OPEN
A2	LOX TURBOPUMP BYPASS VALVE CLOSE

Table 3.1-3. Command and Response List (Cont)

AUXILIARY PROPULSION (Cont)

DAU	COMMANDS (Cont)	
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE OPEN	
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE CLOSE	

DAU	RESPONSE	
A1	GH_2 ISOLATION VALVE OPEN PCS	ON/OFF
A1	GH_2 ISOLATION VALVE CLOSE PCS	ON/OFF
A1	GH_2 ISOLATION VALVE OPEN	
A1	GH_2 ISOLATION VALVE CLOSED	
A2	GOX ISOLATION VALVE OPEN PCS	ON/OFF
A2	GOX ISOLATION VALVE CLOSE PCS	ON/OFF
A2	GOX ISOLATION VALVE OPEN	
A2	GOX ISOLATION VALVE CLOSED	
A1	LH_2 FEED ISOLATION VALVE OPEN PCS	ON/OFF
A1	LH_2 FEED ISOLATION VALVE CLOSE PCS	ON/OFF
A1	LH_2 FEED ISOLATION VALVE OPEN	
A1	LH_2 FEED ISOLATION VALVE CLOSED	
A2	LOX FEED ISOLATION VALVE OPEN PCS	ON/OFF
A2	LOX FEED ISOLATION VALVE CLOSE PCS	ON/OFF
A2	LOX FEED ISOLATION VALVE OPEN	
A2	LOX FEED ISOLATION VALVE CLOSED	
A1	LH_2 SYSTEM BLEED VALVE OPEN PCS	ON/OFF
A1	LH_2 SYSTEM BLEED VALVE CLOSE PCS	ON/OFF
A1	LH_2 SYSTEM BLEED VALVE OPEN	
A1	LH_2 SYSTEM BLEED VALVE CLOSED	
A2	LOX SYSTEM BLEED VALVE OPEN PCS	ON/OFF
A2	LOX SYSTEM BLEED VALVE CLOSE PCS	ON/OFF
A2	LOX SYSTEM BLEED VALVE OPEN	
A2	LOX SYSTEM BLEED VALVE CLOSED	
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO FUEL CELL PCS	ON/OFF
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO VENT PCS	ON/OFF
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO FUEL CELL	
A1	GH_2 FUEL CELL/APS SELECTOR VALVE TO VENT	
A2	GOX FUEL CELL/APS SELECTOR VALVE TO FUEL CELL PCS	ON/OFF
A2	GOX FUEL CELL/APS SELECTOR VALVE TO VENT PCS	ON/OFF
A2	GOX FUEL CELL/APS SELECTOR VALVE TO FUEL CELL	
A2	GOX FUEL CELL/APS SELECTOR VALVE TO VENT	
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE OPEN PCS	ON/OFF
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE CLOSE PCS	ON/OFF
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE OPEN	
A1	GAS GENERATOR GH_2 SYSTEM CONTROL VALVE CLOSED	

Table 3.1-3. Command and Response List (Cont)

AUXILIARY PROPULSION (Cont)

DAU	RESPONSE (Cont)	
A2	GAS GENERATOR GOX SYSTEM CONTROL VALVE OPEN PCS	ON/OFF
A1	GAS GENERATOR GOX SYSTEM CONTROL VALVE CLOSE PCS	ON/OFF
A2	GAS GENERATOR GOX SYSTEM CONTROL VALVE OPEN	
A2	GAS GENERATOR GOX SYSTEM CONTROL VALVE CLOSED	
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN PCS	ON/OFF
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE PCS	ON/OFF
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN	
A1	LH_2 HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSED	
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN PCS	ON/OFF
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE PCS	ON/OFF
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN	
A1	LH_2 TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSED	
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN PCS	ON/OFF
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE PCS	ON/OFF
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN	
A2	LOX HEAT EXCHANGER GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSED	
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN PCS	ON/OFF
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSE PCS	ON/OFF
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE OPEN	
A2	LOX TURBOPUMP GAS GENERATOR BIPROPELLANT CONTROL VALVE CLOSED	
A1	GH_2 VENT VALVE OPEN PCS	ON/OFF
A1	GH_2 VENT VALVE CLOSE PCS	ON/OFF
A1	GH_2 VENT VALVE OPEN	
A1	GH_2 VENT VALVE CLOSED	
A2	GOX VENT VALVE OPEN PCS	ON/OFF
A2	GOX VENT VALVE CLOSE PCS	ON/OFF
A2	GOX VENT VALVE OPEN	
A2	GOX VENT VALVE CLOSED	
A1	GH_2 REGULATOR OUTPUT PRESSURE SWITCH POWER PCS	ON/OFF
A1	GH_2 REGULATOR OUTPUT PRESSURE SWITCH	ON/OFF
A1	GH_2 ACCUMULATOR PRESSURE SWITCH POWER PCS	ON/OFF
A1	GH_2 ACCUMULATOR PRESSURE SWITCH	ON/OFF
A2	GOX REGULATOR OUTPUT PRESSURE SWITCH POWER PCS	ON/OFF
A2	GOX REGULATOR OUTPUT PRESSURE SWITCH	ON/OFF
A2	GOX ACCUMULATOR PRESSURE SWITCH POWER PCS	ON/OFF
A2	GOX ACCUMULATOR PRESSURE SWIITCH	ON/OFF
A1	LH_2 HEAT EXCHANGER GAS GENERATOR IGNITER PCS	ON/OFF
A2	LOX HEAT EXCHANGER GAS GENERATOR IGNITER PCS	ON/OFF
A1	LH_2 TURBOPUMP GAS GENERATOR IGNITER PCS	ON/OFF
A2	LOX TURBOPUMP GAS GENERATOR IGNITER PCS	ON/OFF

Table 3.1-3. Command and Response List (Cont)

AUXILIARY PROPULSION (Cont)

DAU	RESPONSE (Cont)	
A1	LH_2 TURBOPUMP BYPASS VALVE OPEN PCS	ON/OFF
A1	LH_2 TURBOPUMP BYPASS VALVE CLOSE PCS	ON/OFF
A1	LH_2 TURBOPUMP BYPASS VALVE OPEN	
A1	LH_2 TURBOPUMP BYPASS VALVE CLOSED	
A2	LOX TURBOPUMP BYPASS VALVE OPEN PCS	ON/OFF
A2	LOX TURBOPUMP BYPASS VALVE CLOSE PCS	ON/OFF
A2	LOX TURBOPUMP BYPASS VALVE OPEN	
A2	LOX TURBOPUMP BYPASS VALVE CLOSED	
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE OPEN PCS	ON/OFF
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE CLOSE PCS	ON/OFF
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE OPEN	
A2	LOX PUMP SEAL HELIUM SUPPLY VALVE CLOSED	

Table 3.1-3. Command and Response List (Cont)

DATA MANAGEMENT		
DAU		COMMANDS
A2	TAPE RECORDER START	
A2	TAPE RECORDER STOP	
DAU		RESPONSES
A2	TAPE RECORDER START PCS	ON/OFF
A2	TAPE RECORDER STOP PCS	ON/OFF

G,N & C	
DAU	COMMANDS
F1(S/D)	GIMBALED STAR TRACKER COMMAND DATA
F2	HORIZON TRACKER A TRACKING-MODE COARSE ON
F2	HORIZON TRACKER B TRACKING-MODE COARSE ON
F2	HORIZON TRACKER C TRACKING-MODE COARSE ON
F2	HORIZON TRACKER D TRACKING-MODE COARSE ON
F2	HORIZON TRACKER A TRACKING-MODE PRECISION ON
F2	HORIZON TRACKER B TRACKING-MODE PRECISION ON
F2	HORIZON TRACKER C TRACKING-MODE PRECISION ON
F2	HORIZON TRACKER D TRACKING-MODE PRECISION ON
F2	HORIZON TRACKER A SEARCH ON
F2	HORIZON TRACKER B SEARCH ON
F2	HORIZON TRACKER C SEARCH ON
F2	HORIZON TRACKER D SEARCH ON
F2	HORIZON TRACKER A CALIBRATION ON
F2	HORIZON TRACKER B CALIBRATION ON
F2	HORIZON TRACKER C CALIBRATION ON
F2	HORIZON TRACKER D CALIBRATION ON
F2	HORIZON TRACKER READOUT ENABLE ON
F2	HORIZON TRACKER ALTERNATE ROLL NO. 1 ON
F2	HORIZON TRACKER ALTERNATE ROLL NO. 2 ON
F1	IMU DATA REQUEST NO. 1 ON
F1	IMU DATA REQUEST NO. 2 ON
F1	IMU STATUS ENABLE ON
A1	BACKUP STABILIZATION SYSTEM COMPUTER INHIBIT ON
F2	HORIZON TRACKER DEPLOY ON
F2	HORIZON TRACKER RETRACT ON

Table 3.1-3. Command and Control List (Cont)

G, N & C (Cont)

DAU	RESPONSES
F2(S/D)	GIMBALED STAR TRACKER OUTPUT & TEST DATA
F2	HORIZON TRACKER A ALARM LOGIC ON/OFF
F2	HORIZON TRACKER B ALARM LOGIC ON/OFF
F2	HORIZON TRACKER C ALARM LOGIC ON/OFF
F2	HORIZON TRACKER D ALARM LOGIC ON/OFF
F2	HORIZON TRACKER A SUN-PRESENT LOGIC SIGNAL ON/OFF
F2	HORIZON TRACKER B SUN-PRESENT LOGIC SIGNAL ON/OFF
F2	HORIZON TRACKER C SUN-PRESENT LOGIC SIGNAL ON/OFF
F2	HORIZON TRACKER D SUN-PRESENT LOGIC SIGNAL ON/OFF
F2 (S/D)	HORIZON TRACKER OUTPUT NO. 1
F2 (S/D)	HORIZON TRACKER OUTPUT NO. 2
F1(S/D)	IMU X DATA OUTPUT
F1(S/D)	IMU Y DATA OUTPUT
F1(S/D)	IMU Z DATA OUTPUT
F1 (S/D)	IMU STATUS DATA OUTPUT
F1(A)	AUTOCOLLIMATOR PITCH OUTPUT
F1(A)	AUTOCOLLIMATOR ROLL OUTPUT
F2	HORIZON TRACKER DEPLOY PCS ON/OFF
F2	HORIZON TRACKER RETRACT PCS ON/OFF
F2	HORIZON TRACKER DEPLOYED
F2	HORIZON TRACKER RETRACTED

RENDEZVOUS & DOCKING

DAU	COMMANDS
F2(S/D)	SCANNING LASER RADAR INPUT
F1	RF SWITCH TO OPERATIONAL
F1	RF SWITCH TO CHECKOUT

DAU	RESPONSES
F2(S/D)	SCANNING LASER RADAR OUTPUT
F1	RF SWITCH TO OPERATIONAL PCS ON/OFF
F1	RF SWITCH TO CHECKOUT PCS ON/OFF

Table 3.1-3. Command and Control List (Cont)

COMMUNICATIONS

DAU	COMMANDS
F1	RF SWITCH TO DATA MANAGEMENT
F1	RF SWITCH TO TELEVISION
F1	FM TRANSMITTER OUTPUT ON
F1	TRANSPONDER TRANSMITTER A ON
F1	TRANSPONDER TRANSMITTER B ON
F1	TRANSPONDER RECEIVER A ON
F1	TRANSPONDER RECEIVER B ON
F1	POWER AMPLIFIER A ON
F1	POWER AMPLIFIER B ON

DAU	RESPONSES	
F1	RF SWITCH TO DATA MANAGEMENT PCS	ON/OFF
F1	RF SWITCH TO TELEVISION PCS	ON/OFF
F1	TRANSPONDER TRANSMITTER A	ON/OFF
F1	TRANSPONDER TRANSMITTER B	ON/OFF
F1	TRANSPONDER RECEIVER A	ON/OFF
F1	TRANSPONDER RECEIVER B	ON/OFF
F1	POWER AMPLIFIER A	ON/OFF
F1	POWER AMPLIFIER B	ON/OFF

FUEL CELL

DAU	COMMANDS
A1	WATER PURGE VALVE OPEN
A1	STEAM PURGE VALVE OPEN
A1	WATER STORAGE ISOLATION VALVE OPEN
A1	SECONDARY WATER FILL VALVE OPEN
A1	GAS SEPARATOR VENT VALVE OPEN
A1	BY-PRODUCT WATER CHAMBER PURGE VALVE OPEN
A1	CONDENSER TEMP CONTROL VALVE OPEN
A1	GH_2 PURGE CONTROL VALVE OPEN
A1	O_2 PURGE CONTROL VALVE OPEN
A1	FREON PUMP START
A1	FREON PUMP STOP
A1	FUEL CELL GH_2 REACTANT CONTROL VALVE OPEN
A1	FUEL CELL GOX REACTANT CONTROL VALVE OPEN

Table 3.1-3. Command and Control List (Cont)

FUEL CELL	
DAU	RESPONSES
A1	WATER PURGE VALVE OPEN PCS ON/OFF
A1	WATER PURGE VALVE OPEN/CLOSED
A1	STEAM PURGE VALVE OPEN PCS ON/OFF
A1	STEAM PURGE VALVE OPEN/CLOSED
A1	WATER STORAGE ISOLATION VALVE OPEN PCS ON/OFF
A1	WATER STORAGE ISOLATION VALVE OPEN/CLOSED
A1	SECONDARY WATER FILL VALVE OPEN PCS ON/OFF
A1	SECONDARY WATER FILL VALVE OPEN/CLOSED
A1	GAS SEPARATOR VENT VALVE OPEN PCS ON/OFF
A1	GAS SEPARATOR VENT VALVE OPEN/CLOSED
A1	BY-PRODUCT WATER CHAMBER PURGE VALVE OPEN PCS ON/OFF
A1	BY-PRODUCT WATER CHAMBER PURGE VALVE OPEN/CLOSED
A1	CONDENSER TEMP CONTROL VALVE OPEN PCS ON/OFF
A1	CONDENSER TEMP CONTROL VALVE OPEN/CLOSED
A1	GH_2 PURGE CONTROL VALVE OPEN PCS ON/OFF
A1	GH_2 PURGE CONTROL VALVE OPEN/CLOSED
A1	02 PURGE CONTROL VALVE OPEN PCS ON/OFF
A1	02 PURGE CONTROL VALVE OPEN/CLOSED
A1	FUEL CELL GH_2 REACTANT CONTROL VALVE OPEN PCS ON/OFF
A1	FUEL CELL GH_2 REACTANT CONTROL VALVE OPEN/CLOSED
A1	FUEL CELL GOX REACTANT CONTROL VALVE OPEN PCS ON/OFF
A1	FUEL CELL GOX REACTANT CONTROL VALVE OPEN/CLOSED
A1	FREON PUMP START PCS ON/OFF
A1	FREON PUMP STOP PCS ON/OFF

Table 3.1-3. Command and Response List (Cont)

ELECTRICAL POWER DISTRIBUTION	
DAU	COMMANDS
F1	G,N&C DISTRIBUTOR PCS ON
F1	COMMUNICATIONS & INSTRUMENTATION DISTRIBUTOR PCS ON
F2	THERMAL CONTROL DISTRIBUTOR PCS ON
A2	MAIN PROPULSION SYSTEM DISTRIBUTOR PCS NO. 1 ON
A2	MAIN PROPULSION SYSTEM DISTRIBUTOR PCS NO. 2 ON
A1	APS DISTRIBUTOR PCS ON
A1	MECHANICAL SYSTEMS DISTRIBUTOR PCS ON
A1	TVC DISTRIBUTOR PCS ON
F2	RENDEZVOUS & DOCKING DISTRIBUTOR PCS ON
A1	INVERTER PCS ON
A2	BATTERY LOAD RESISTOR PCS ON
A2	FUEL CELL LOAD RESISTOR PCS ON
F2	TAPE RECORDER PCS ON
F1	TV CAMERA PCS ON
F1	TV CAMERA LIGHTS PCS ON
F2	LASER RADAR PCS ON
F1	RENDEZVOUS TRANSPONDER PCS ON
F1	FM TRANSMITTER PCS ON
A2	AFT DATA ACQUISITION UNIT NO. 1 PCS ON
A1	AFT DATA ACQUISITION UNIT NO. 2 PCS ON
A2	FORWARD DATA ACQUISITION UNIT NO. 1 PCS ON
A1	FORWARD DATA ACQUISITION UNIT NO. 2 PCS ON
F2	MEASUREMENT PROCESSOR UNIT PCS ON
A1	APS GH_2 FLOW CONTROLLER PCS ON
A2	APS GOX FLOW CONTROLLER PCS ON
DAU	RESPONSES
F1	G,N&C DISTRIBUTOR PCS ON/OFF
F1	COMMUNICATIONS & INSTRUMENTATION DISTRIBUTOR PCS ON/OFF
F2	THERMAL CONTROL DISTRIBUTOR PCS ON/OFF
A2	MAIN PROPULSION SYSTEM DISTRIBUTOR PCS NO. 1 ON/OFF
A2	MAIN PROPULSION SYSTEM DISTRIBUTOR PCS NO. 2 ON/OFF
A1	APS DISTRIBUTOR PCS ON/OFF

Table 3.1-3. Command and Response List (Cont)

ELECTRICAL POWER DISTRIBUTION (Cont)	
DAU	RESPONSES (Cont)
A1	MECHANICAL SYSTEMS DISTRIBUTOR PCS ON/OFF
A1	TVC DISTRIBUTOR PCS NO. 1 ON/OFF
A1	TVC DISTRIBUTOR PCS NO. 2 ON/OFF
F2	RENDEZVOUS & DOCKING DISTRIBUTOR PCS NO. 1 ON/OFF
F2	RENDEZVOUS & DOCKING DISTRIBUTOR PCS NO. 2 ON/OFF
A1	INVERTER PCS ON/OFF
A2	BATTERY LOAD RESISTOR PCS ON/OFF
A2	FUEL CELL LOAD RESISTOR PCS ON/OFF
F2	TAPE RECORDER PCS ON/OFF
F1	TV CAMERA PCS ON/OFF
F1	TV CAMERA LIGHTS PCS ON/OFF
F2	LASER RADAR PCS ON/OFF
F1	RENDEZVOUS TRANSPONDER PCS ON/OFF
F1	FM TRANSMITTER PCS ON/OFF
A2	AFT DATA ACQUISITION UNIT NO. 1 PCS ON/OFF
A1	AFT DATA ACQUISITION UNIT NO. 2 PCS ON/OFF
A2	FORWARD DATA ACQUISITION UNIT NO. 1 PCS ON/OFF
A1	FORWARD DATA ACQUISITION UNIT NO. 2 PCS ON/OFF
F2	MEASUREMENT PROCESSOR UNIT PCS ON/OFF
A1	APS GH_2 FLOW CONTROLLER PCS ON/OFF
A2	APS GOX FLOW CONTROLLER PCS ON/OFF
A2	POWER TRANSFER SWITCH TO EOS
A2	POWER TRANSFER SWITCH TO TUG
A2	POWER TRANSFER SWITCH TO GSE

PURGE BAG	
DAU	COMMANDS
F1	LH_2 TANK INSULATION PURGE VALVE OPEN
F1	LOX TANK INSULATION PURGE VALVE OPEN
F1	RELIEF VALVE NO. 1 PRESSURE SWITCH PCS ON
F1	RELIEF VALVE NO. 2 PRESSURE SWITCH PCS ON
F1	INTERTANK/INSULATION SELECTOR VALVE TO INTERTANK
F1	INTERTANK/INSULATION SELECTOR VALVE TO INSULATION
F1	VENT VALVE NO. 1 OPEN
F1	VENT VALVE NO. 2 OPEN

Table 3.1-3. Command and Response List (Cont)

DAU	PURGE BAG (Cont) — RESPONSES
F1	LH_2 TANK INSULATION PURGE VALVE PCS ON/OFF
F1	LH_2 TANK INSULATION PURGE VALVE OPEN/CLOSED
F1	LOX TANK INSULATION PURGE VALVE PCS ON/OFF
F1	LOX TANK INSULATION PURGE VALVE OPEN/CLOSED
F1	RELIEF VALVE NO. 1 PRESSURE SWITCH PCS ON/OFF
F1	RELIEF VALVE NO. 1 PRESSURE SWITCH ON/OFF
F1	RELIEF VALVE NO. 2 PRESSURE SWITCH PCS ON/OFF
F1	RELIEF VALVE NO. 2 PRESSURE SWITCH ON/OFF
F1	INTERTANK/INSULATION SELECTOR VALVE TO INTERTANK PCS ON/OFF
F1	INTERTANK/INSULATION SELECTOR VALVE TO INSULATION PCS ON/OFF
F1	INTERTANK/INSULATION SELECTOR VALVE TO INTERTANK
F1	INTERTANK/INSULATION SELECTOR VALVE TO INSULATION
F1	VENT VALVE NO. 1 PCS ON/OFF
F1	VENT VALVE NO. 1 OPEN ON/OFF
F1	VENT VALVE NO. 1 CLOSED
F1	VENT VALVE NO. 2 PCS ON/OFF
F1	VENT VALVE NO. 2 OPEN ON/OFF
F1	VENT VALVE NO. 2 CLOSED

DAU	TUG/PAYLOAD DOCKING — COMMANDS
F1	TUG/PAYLOAD LATCHES ENGAGE
F1	TUG/PAYLOAD LATCHES DISENGAGE
F1	TUG/PAYLOAD PROBES EXTEND
F1	TUG/PAYLOAD PROBES RETRACT

DAU	RESPONSES
F1	TUG/PAYLOAD LATCHES ENGAGE PCS ON/OFF
F1	TUG/PAYLOAD LATCHES DISENGAGE PCS ON/OFF
F1	TUG/PAYLOAD LATCH NO. 1 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 2 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 3 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 4 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 5 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 6 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 7 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 8 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 9 ENGAGED/DISENGAGED

Table 3.1-3 Command and Response List (Cont)

TUG/PAYLOAD DOCKING (Cont)	
DAU	RESPONSES (Cont)
F1	TUG/PAYLOAD LATCH NO. 10 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 11 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD LATCH NO. 12 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 13 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 14 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 15 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 16 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 17 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 18 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 19 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 20 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 21 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 22 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 23 ENGAGED/DISENGAGED
F2	TUG/PAYLOAD LATCH NO. 24 ENGAGED/DISENGAGED
F1	TUG/PAYLOAD PROBE NO. 1 EXTENDED
F1	TUG/PAYLOAD PROBE NO. 1 RETRACTED
F1	TUG/PAYLOAD PROBE NO. 2 EXTENDED
F1	TUG/PAYLOAD PROBE NO. 2 RETRACTED
F1	TUG/PAYLOAD PROBE NO. 3 EXTENDED
F1	TUG/PAYLOAD PROBE NO. 3 RETRACTED
F1	TUG/PAYLOAD PROBES EXTEND PCS ON/OFF
F1	TUG/PAYLOAD PROBES RETRACT PCS ON/OFF

3.1.4 Component Characteristics

The component design utilized in the DMS relies heavily upon technology to be developed for the Air Force/NR B-1 Program in the 1972 through 1976 time span. With the exception of the computer, status and control panel and on-board tape recorder, the DMS components will be modified versions of hardware now defined by procurement specifications generated for the B-1 Program by the Los Angeles Division of NR. The modifications will consist of: (1) repackaging the components to operate with passive thermal control (the B-1 units are designed to accept forced air cooling), (2) redesigning the components' power supplies to be compatible with a 28 VDC power source (the B-1 units use AC power for operation), and (3) altering the number of input and output channels to provide component optimization based on actual Tug requirements. These modifications were taken into consideration during the estimation of component weights. The basic building blocks (i.e., integrated circuit chips) that will be used in the components represent state-of-the-art technology. However, the utilization of these circuits in the particular combinations necessary to satisfy the Tug design requirements has never been achieved. The development of the basic components by the B-1 Program should greatly reduce the overall cost impact to the Tug Program in as much as the added Tug-particular requirements will represent a relatively minor development effort. Functional implementaton of the design concept does not constitute a high risk factor. Also, the risk associated with the development of components capable of meeting minimum weight and power requirements is not considered to be significant.

The computer specified in the DMS is modeled after the DM216, a unit now under development by the Autonetics Division of NR. The DM216 is a general purpose machine that will possess the necessary performance characteristics while meeting the basic requirement of minimum weight. The basic DM216 will have the inherent capability to be customized to a particular application, thereby offering a low development risk to the Tug Program.

The DMS status and control panel will be an item requiring the integration and packaging of existing components which is not viewed as a significant effort. The unit will essentially be an input/output terminal for the Tug avionics system with design emphasis placed on operational simplification of the man/machine interface.

The on-board tape recorder selected for the baseline configuration represents an existing design. Use of advanced (1976) technology was not pursued in the selection of the tape recorder. The low weight of the selected unit made this unnecessary.

Computer

The processor utilized in the baseline system is a stored-program general-purpose digital computer designed to meet real-time high-speed applications. Processing rates in excess of 400,000 equivalent adds per second are attained by optimizng the machine organization and instruction set

(68 instructions), using metal oxide semiconductors and large scale integrated circuit (MOS/LSI) building blocks and plated wire memory, and using a 1 microsecond core storage.

Three data handling modules make up the computer: (1) the central processing unit (CPU), (2) input-output controller (I/OC), and (3) the main memory. Figure 3.1-1 shows the basic data handling modules of the computer.

Operating power for the data handling modules is supplied by the vehicle electrical power system. Power on and system reset commands initialize the computer by resetting and clearing all registers, control logic, and interfaces in the CPU and I/OC. An instruction counter is set to a predefined storage location. Reset occurs when power is applied, or when an external system reset signal is received.

For a power off command, the CPU receives a power off interrupt (POI) signal from the power supply. At the end of the storage cycle, during which the POI signal appears, the CPU inhibits the start of the next storage cycle and sends a signal back to the power supply that it is acceptable to shut down power. This procedure ensures no loss of storage information during power off condition for both transient turn off and normal power down.

Central Processor Unit

The CPU is the functional heart of the computer where all data processing and control functions are performed. Contained in the CPU are the facilities for addressing main memory, fetching and storing information, processing arithmetic and logic data, sequential instruction execution, processing interrupts, and initiating communication between main memory and the I/OC. The basic functional sections of the CPU are timing, general registers, working registers, arithmetic-logic unit, control, interrupt logic, and I/OC and main memory interfaces.

The main memory has a 1 microsecond cycle time with a 16 bit parallel readout. A parity bit is associated with each half-word (16 bits) of the storage word. The parity bits are generated whenever a store operation takes place and are checked for odd parity whenever a read operation takes place. An attempt to store in a location which is protected will not alter the contents of the location and will interrupt the CPU. The CPU has the ability to directly address up to 65,536 (16 bit) memory words.

The basic building blocks utilized in the construction of the CPU are such that the machine may be structured with a 16 bit, 24 bit, or 32 bit word lengths with only minimal changes in the number of MOS devices. For the Tug point design, the 32 bit word length has been selected.

Input-Output Controller

The I/OC provides the means of communicating with devices external to the computer. I/OC operation proceeds simultaneously and independently after

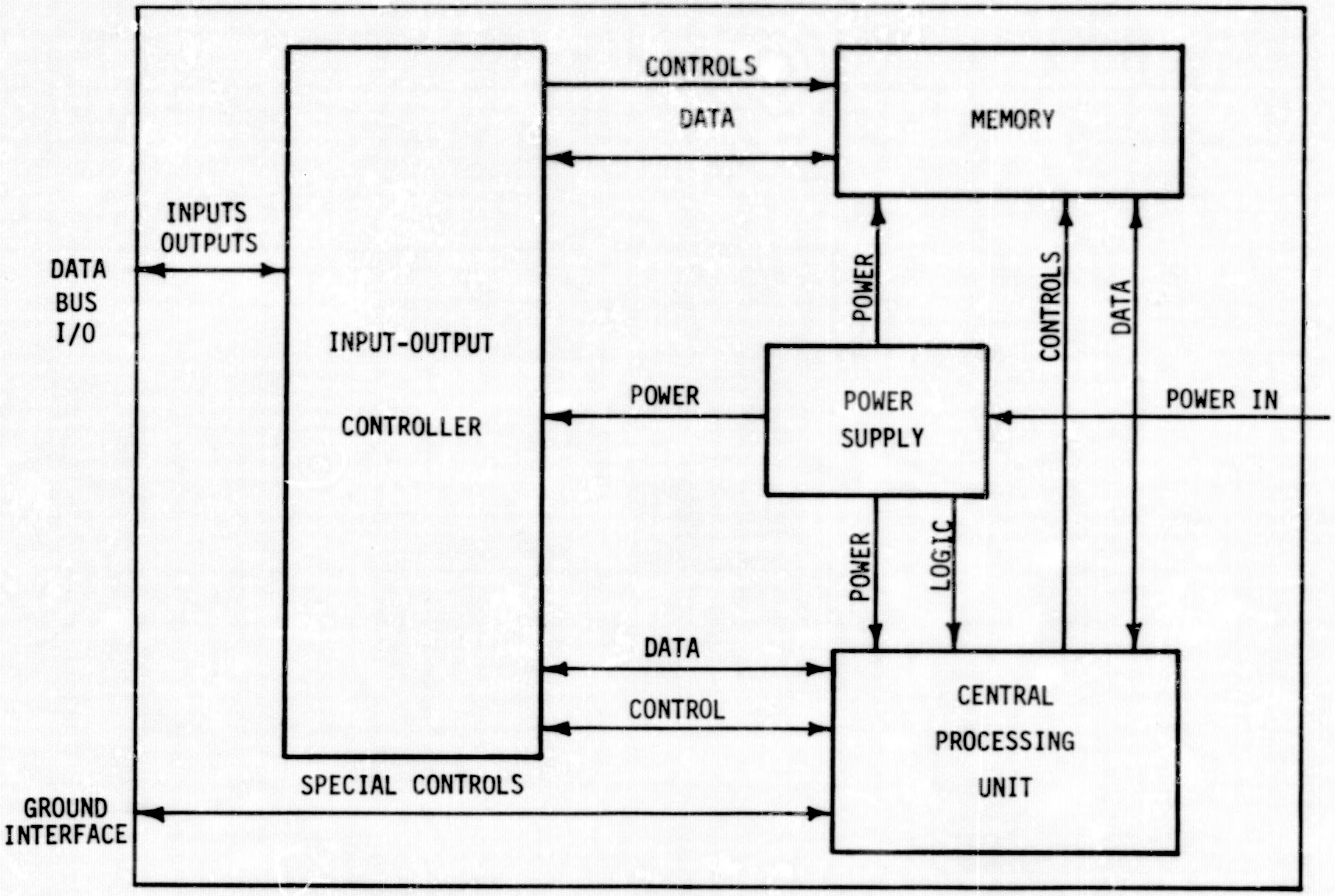


Figure 3.1-1 Digital Computer Functional Block Diagram

program control initiation by the CPU. The I/OC communicates with associated peripherals over the data bus.

The serial data bus connects the I/OC to a maximum of 31 peripherals. In the receive mode the I/OC accepts the data from the data bus, detects and decodes the incoming information by using the Manchester code to derive clocking information, and to convert the data to the signals required by the computer.

Input-output operations are initiated by an initializing command from the CPU. The I/OC contains the necessary logic to proceed from the initialing command through a complete input-output subprogram which is stored in main memory.

Main Memory

The baseline computer configuration has 64K words of main storage. Each word contains 16 bits each. Main memory has the capability of being accessed for read or write operations by the CPU and I/OC. Memory accesses are honored upon completion of the unfinished memory cycle except when simultaneous requests are received in which case a priority scheme is exercised.

The physical and performance characteristics of the computer are summarized in Table 3.1-4.

Interface Unit

The IU provides a serial digital input/output interface with the data bus, and a parallel digital input/output interface for data transfer to/from DAU's or other peripherals. The IU is a single plug-in assembly packaged within the DAU or any other peripheral equipment directly interfaced with the IU. The IU includes a data receiver/transmitter, a clock generator, and the necessary logic to provide control signals to interpret control instructions and regulate the operation of the IU/peripheral interface as directed by the control computer. The functional block diagram depicting operation of the IU within the DAU is as shown in Figure 3.1-2.

The IU transmits and receives data in the form of a message which will contain one to sixteen words, as below.

CONTROL WORD	DATA WORD NO. 1	DATA WORD NO. n	DEAD (1) TIME OF ANY DURATION	CONTROL WORD	
					ETC.

MESSAGE LENGTH OF 1 TO 16 WORDS

(1) NEXT MESSAGE MAY FOLLOW IMMEDIATELY OR THERE MAY BE A DEAD TIME OF ANY DURATION.

Table 3.1-4. Computer Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	**Characteristics**
Weight	26 lb.
Size	11 x 6.5 x 8.3 in.
Power	60 watts
Voltage/Current	28 VDC ±10%/2.1 amps
Operating Temperature Range	$-65^{o}F$ to $+130^{o}F$
Installation	Standard avionic equipment mounting
Reliability MTBF	20,000 hrs.

PERFORMANCE CHARACTERISTICS	
Parameter	**Performance**
Type	Stored program, parallel general purpose digital computer
Number System	Binary, fixed point, two's complement
Organization	Conventional
Component Technology	4 Phase, P-channel MOS/LSI
Data Word Length	16 or 32 bits, including sign
Instruction Word Length	16 bits and 32 bits
Memory Organization	8192, 16 bit words of plated wire (see below for options) half word (16 bits) or full word (32) - random access - non-destructive readout - non-volatile
Memory Addressing	65,536 words directly addressable - displacement addressing also provided
Memory Speed	0.500 μsec cycle time (16 bit)
Basic Clock Rate	4.0 MHz
Information Bit Rate/Time Register	1 MHz/1.00 μsec

Table 3.1-4. Computer Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS (Cont)	
Parameter	**Performance**
Register Complement	Accumulator - 16 or 32 bits Lower accumulator - 16 or 32 bits Program counter - 16 bits General register file - 16 registers, each 16 bits in length, each addressable, used for - indexing - displacement addressing - temporary storage Arithmetic status register - 4 bits
Interrupts	External - 8 interrupts (expandable) - cyclical priority; programmable masking - interrupt suspension capability Internal - 4 interrupts
Instruction Repertoire	64 instructions exclusive of interrupts and I/O commands - indexing - indirect addressing - typical execution times Add: 1.25 μsec (register) Add: 2.5 μsec (main memory, indexed) Multiply: 12.50/22.50 μsec Divide: 23.75/43.75 μsec
Input/Output	Inputs: - one buffered 16 or 32 bit parallel output channel - one 6 bit address register

The first word of each message is a control word which may be followed by a maximum of fifteen data words or a minimum of zero data words. The IU operates in the receive mode at all times, except when requested to transmit data. When the transmit mode is requested, the IU remains in the transmit mode until all data words requested have been transmitted. After the last bit of the last word of the message has been transmitted, the IU switches to the receive mode within 1 microsecond.

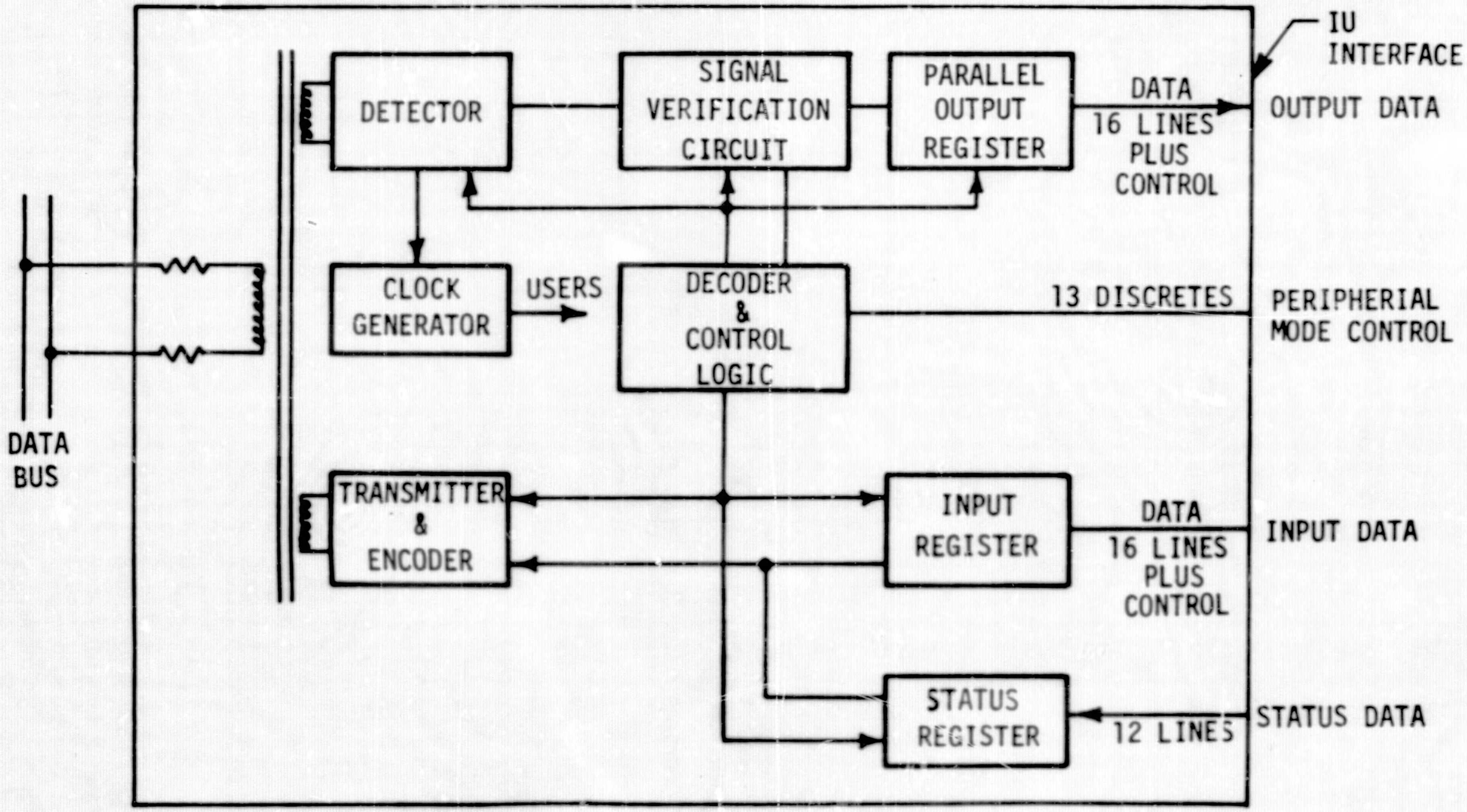


Figure 3.1-2 Interface Unit Functional Block Diagram

The basic functions performed by the IU in the receive mode are: to accept from the data bus data signals in the specified format, detect and decode the incoming signals using the Manchester code to derive clocking information, convert the data to signals required by the peripheral, and to accept and generate the control signals required to control the IU peripheral interface.

The IU contains the following to perform the basic functions when processing data from the data bus:

- Receiver/data detector
- Serial to parallel register
- Word counter
- Control register and reset logic
- Address decoder
- Mode code register decoder
- Status register
- Circuitry required to output data words to peripherals
- Take data control signals
- Clock generator

The basic functions performed by the IU in the transmit mode are to accept parallel digital data signals from the peripheral, to convert the signals to the specified format, to use the internal IU clock to transmit the signals, and to accept and generate the control signals required to control the IU peripheral input interface. When the computer commands the IU to transmit data, there will be a 2.0 microsecond minimum, 10.0 microseconds maximum dead time before the IU transmitter is permitted to turn on.

The IU includes the following to perform the basic functions when processing data from the peripherals:

- Circuitry required to input data words from the peripheral.
- A parallel-to-serial register which accepts 16 bit parallel words, and outputs 16 bit serial words.
- A send-data control signal to be used in conjunction with the input data lines and an inputing-data control signal.
- The circuitry required to generate a test bit.

- A parity generator which generates odd "ones" parity.
- The circuitry to generate the control field bits.
- A sync field generator.
- A Manchester encoder which encodes the output signals to be placed upon the data bus.
- A power amplifier capable of supplying the output signal required to meet the transmitter output requirements.
- A transmitter switch which will isolate the transmitter from the data bus during non-transmitting modes of operating.
- A 16-bit status register.
- A "reset" signal.
- A self-test capability.
- A clock generator which will be used to control timing, sequence of events, and the bit rate of the transmitter. The clock at the interface will be 1.0 MHz.
- A logic reset signal.

The physical and performance characteristics of the IU are summarized in Table 3.1-5.

Data Acquisition Unit

The DAU utilizes the capabilities provided by the IU for the accomplishing of its interface with the data bus. Each DAU has an IU as an integral part of its assembly. The DAU functions as a distribution point for stimuli to the vehicle subsystem and as a programmable multiplexing device for the gathering of subsystem output data.

The DAU performs the function of sampling analog, discrete, and serial digital signals from the vehicle subsystem and processing this data for subsequent transfer to the control computer on command. The DAU also provides a means of outputting discrete and serial digital signals to the vehicle subsystem under direct computer control. The DAU is capable of being programmed by the control computer via the data bus with up to 15 instructions. The instruction may call for any combination of stimuli generation or measurement sampling. The DAU is capable of randomly selecting any of its analog measurement inputs discrete inputs or serial digital inputs. Figure 3.1-3 is a functional block diagram for the DAU.

Table 3.1-5. Interface Unit Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	1.75 lb.
Size	0.5 x 6 x 5 in.
Power	Provided by using peripheral
Voltage/Current	Compatible with using peripheral
Operating Temperature Range	$-65^{\circ}F$ to $+160^{\circ}F$
Non-operating Temperature Range	$-80^{\circ}F$ to $+203^{\circ}F$
Installation	Unit utilized as plug-in cord in using peripheral
Reliability MTBF	4000 hrs.
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Data Format	Manchester 11 Bi-phase-L at a rate of 1.0 megabits/second
Logic Levels	Logic "1" + 4.0 ±1.5 volts Current 5.0 ma minimum Rise and Fall Time 100 nanoseconds

In normal operation the DAU accepts a message from the computer via the data bus and IU. The message received may contain up to 15 sequential instructions. After the last word of the message is received by the DAU input buffer, the DAU will automatically (within 2 microseconds) begin to process the instructions.

The processing of instructions begins with the first received and continues sequentially until all instructions are processed. The instructions may call for any combination of input and output signal processing. The frequency from one instruction to the next is controlled by the input buffer control logic. When the last instruction of a given message is read and the requested operation performed, the DAU will wait until the next message is received. When the input buffer is processing data an active signal is maintained in the status register of the IU. When the input buffer completes processing all the instructions, the action signal is removed from the IU status register.

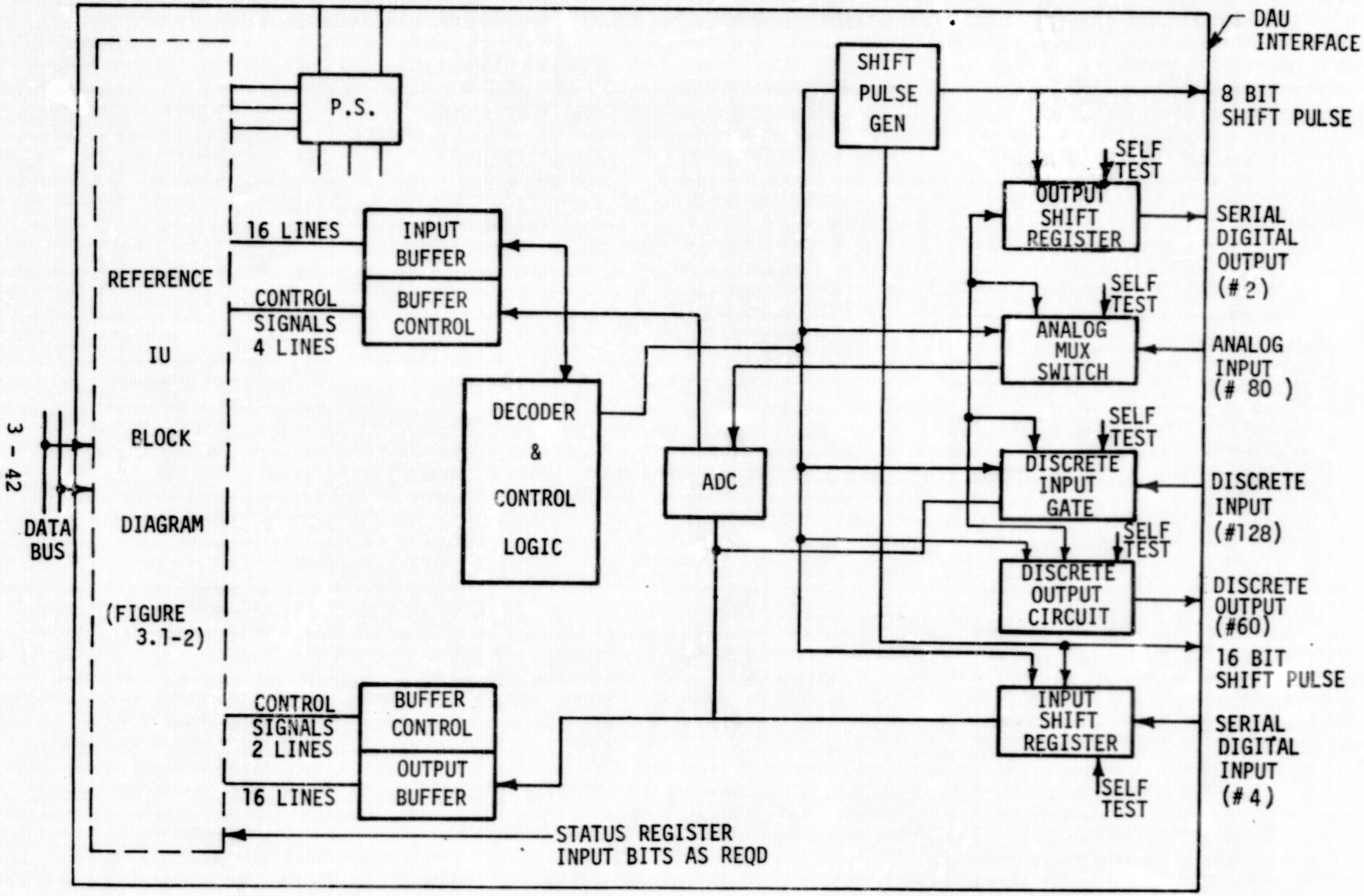


Figure 3.1-3 Data Acquisition Unit Functional Block Diagram

When an instruction calls for outputting an analog measurement from the DAU to the computer, the desired measurement is routed to the analog-to-digital converter (ADC) which converts the analog signal to its equivalent digital signal word. The output of the ADC is stored in the output buffer. The data is relayed to the computer via the IU and data bus upon command of the control computer. The total processing time required to decode the computer instruction, process the analog and store the data in the output register is 20 microseconds. The ADC output is 11 bits plus one sign bit in 2's complement (bi-polar). The overall DAU analog measurement absolute accuracy is +0.1 percent of full scale.

If the instructions read from the input buffer request processing a discrete measurement, the instruction is decoded to establish the desired measurement. The 16 preassigned discrete signals comprising the discrete word containing the desired measurement are gated to the DAU output buffer and outputted to the computer on command. The process time required to process instructions from the input buffer and storing the corresponding discrete in the output buffer is 5 microseconds.

If the instructions read from the input buffer calls for inputing serial digital signals the DAU selects the serial input channel and shifts in a 16 bit serial digital data field from the vehicle subsystem. A 16 bit shift signal generated by the DAU is used to shift out the 16 bit data field from the subsystem output register. The DAU shifts the digital data into the DAU input register with the same shift signal sent to the vehicle subsystem. The digital data will be stored in the output buffers. The processing time required to read the instruction from the input register and shifting in and storing the corresponding data in the output buffer is less than 20 microseconds.

The output register of the DAU is capable of storing up to 15 inputs. The inputs consist of up to 16 bits each. In the transmit mode the output buffer will store and output the processed data in the same sequence that the corresponding instruction is stored in the input buffer. The data stored in the output signal buffer will be outputed in parallel to the IU in the same sequence that data was stored into the output buffer. A data ready signal will be wired to the status register to indicate when the DAU has data for outputing. The output buffer is capable of transmitting processed data prior to the input buffer completing its processing.

The DAU generates discrete output signals in the same method as that used in processing an input signal. The computer instruction calling for a discrete output is processed to identify the desired output channel. The DAU gates that output channel on. The discrete signal will remain a logic "1" until commanded by the computer to reset the function. At the time of generating the stimuli the output channel will be subjected to a self-test and the status of that self-test will be indicated in the least significant bit of the corresponding word in the output register.

If the output is to be digital the data is shifted out serial with an 8 bit shift signal generated by the DAU. The shift signal is also outputed

to the user for shifting the serial digital address into the user input buffer. The processing time required to read the instruction and to output the 8 bits of serial data is less than 10 microseconds.

The DAU has the following input/output signal capacity:

Input Signals

Analog input signals	80
Discrete input signals	128
Digital serial signals	4

Output Signals

Discrete output signals	60
Digital serial signals	2

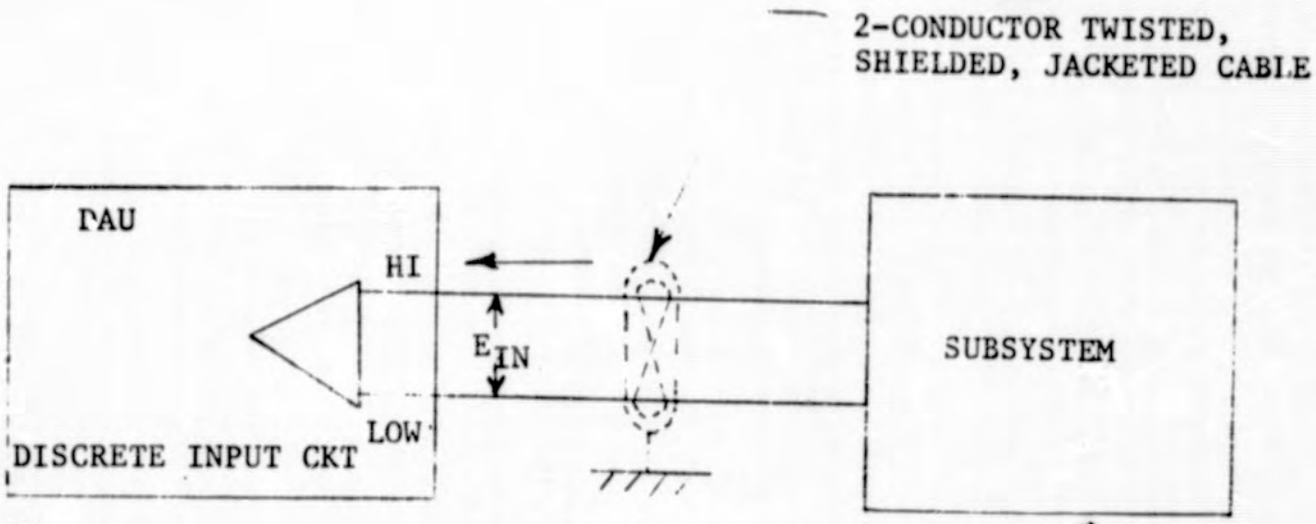
FIGURES 3.1-4, 3.1-5 and 3.1-6 identify signal parameters.

The DAU contains the necessary built-in test hardware to detect all failures that can affect DAU parameters and isolate all such failures to the equipment itself. The go/no-go performance evaluation of test results are such that a failure output or indication is obtained for all out-of-tolerance conditions of the DAU. In the event of failure, the capability to isolate that failure within the DAU is provided by the self-test. The type of testing is selected in accordance with the following restraints: (1) continuous testing or monitoring which can be performed simultaneous with operational use and where such testing does not degrade operational performance, (2) iterative testing not under computer control which will not degrade equipment operational performance.

Test results are available to the IU status register. When the equipment is energized, those parameters which are continuously functioned are monitored to form a go/no-go status available at the IU status register interface and are readable by program control. The DAU performs the self test listed below and provides the results of each individual test as indicated.

SELF TEST REQUIREMENTS

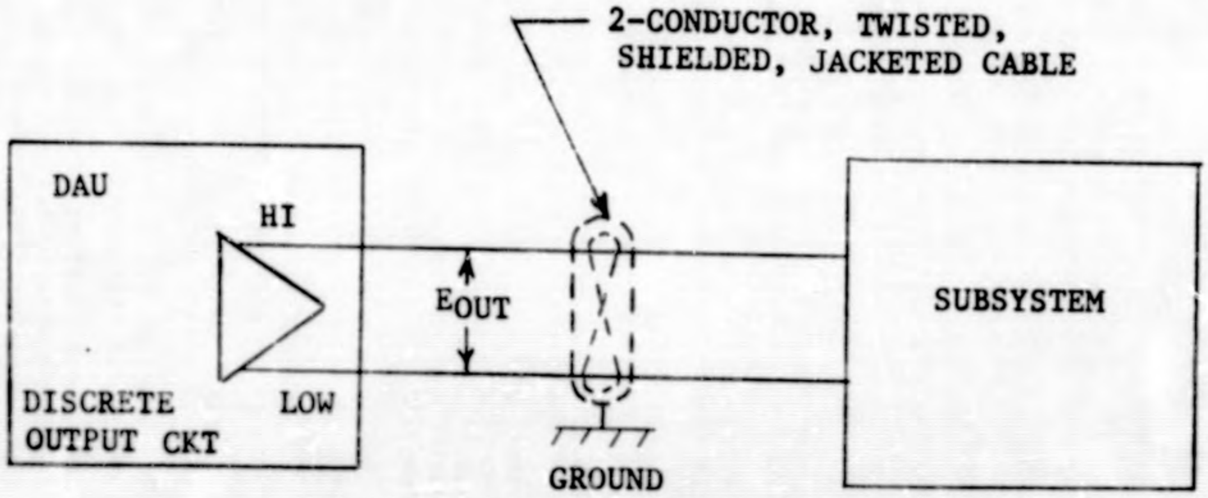
Item Test No. Performed	Primary Performance Requirements: Verified by Test	Test Result Signal
1. Serial Word (ea circuit)	Test word sent to DAU Serial Digital output circuits but inhibited to the Tug subsystem.	Test word looped through serial output channels of DAU and sent back to the computer.



TYPE:	Differential Input (E_{IN})	
INPUT VOLTAGE:	+2.5V to -6.0V represent a logic "1" (Differential)* Modifiable to +14 to +28. 0.0V to ±2.0V represents a logic "0" (Differential). When the "HI" input is a logic "1" with respect to the "LOW" input, the DAU discrete output indicates a logic "1". When the "HI" input is a logic "0" with respect to the "LOW" input, the DAU discrete output indicates a logic "0". When the discrete input circuit is ipen (not connected) the DAU discrete output indicates a logic "0".	
OVER VOLTATE PROTECTION:	± 30V DC or AC Peak to Peak (Differential or Common Mode)	
INPUT IMPEDANCE:	Differential	100 kilo-ohm minimum*
	Common mode to DAU Logic return	100 kilo-ohm minimum*
	With DAU power off, input circuit failure or over-voltage input (Differential and Common Mode)	20 kilo-ohm minimum*
COMMON MODE:	Common mode voltage ± 10 volts	
COMMON MODE REJECTION:	A common mode voltage of ± 10 volts DC to 1.0 KHz shall not cause a logic "1" or logic "0" to be misinterpreted.	
FREQUENCY:	DC to 5 millisecond pulse Noise filtering for pulses less than 1.0 millisecond wide.	

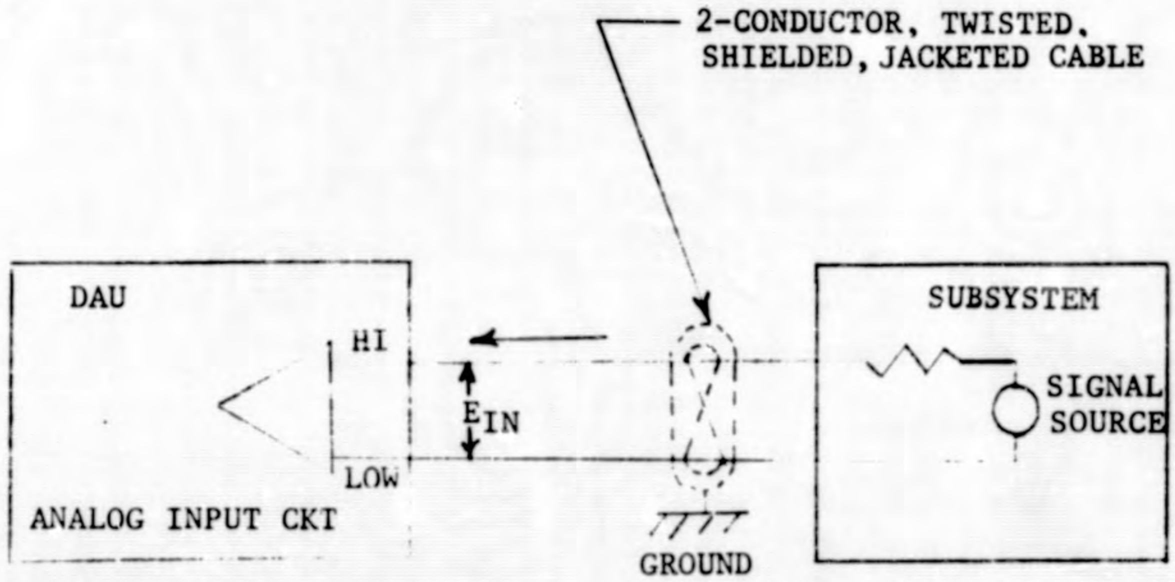
* with a logic "1" or a logic "0" input.

Figure 3.1-4. Discrete Input Circuit Characteristics



TYPE:	Differential Output (E_{OUT})
OUTPUT VOLTAGE:	Logic "1" shall be + 5 $\pm$ 1.0V Logic "0" shall be 0 $\pm$ 0.5V
SHORT PROTECTION:	The output circuit will not be damaged when subjected to shorts to ground or a voltage of plus or minus 10 volts DC or AC peak to peak, line to ground.
OUTPUT CURRENT CAPABILITY:	The output circuits are capable of supplying 5 milliamperes current minimum at plus 4.0 volts, minimum. The output circuit is capable of driving no greater than 50 feet of two conductor cable having a distributed capacitance of 25 pico-farads per foot. The output circuit is capable of sinking 10 milliamperes of current in the digital zero state plus 0.5 volts, maximum.
RISE AND FALL TIMES:	0.1 to 1.0 μseconds, the rise and fall time is measured between the nominal zero and plus 5 volt limits.
COMMON MODE:	Common mode voltage range $\pm$ 10 volts, common mode isolation of 100 kilo-ohm-minimum.

Figure 3.1-5. Discrete Output Circuit Characteristics



E:	Differential Input (E_{IN})	
VOLTAGE:	0.0 to $\pm$ 5V (Differential)	
	When the "HI" input is positive with respect to the "LOW" input shall indicate positive voltage.	
	When the "HI" input is negative with respect to the "LOW" input the ADC output shall indicate a negative voltage.	
OVERVOLTAGE PROTECTION:	$\pm$ 30V DC or AC Peak to Peak (Differential and common mode)	
INPUT IMPEDANCE:	Differential	1.0 meg-ohm minimum*
	Common mode to DAU logic return	1.0 meg-ohm minimum*
	With DAU power off, input circuit failure voltage input	20 KIL-ohm minimum*
COMMON MODE:	Common mode voltage $\pm$ 10 volts DC or AC Peak to Peak	
COMMON MODE REJECTION RATIO:	60 db dc to 1.0 kc.	
FREQUENCY:	DC to 600 Hz.	
	Noise filtering above 600 Hz.	

* with input voltage of 0.0 $\pm$ 5 volts

Figure 3.1-6. Analog Input Circuit Characteristics

2. Discrete Word (ea channel)	Test word selects a discrete output channel but inhibited to the Tug subsystem.	A discrete output signal is sent back to the computer as a word from the discrete input circuit.
3. Analog Select. (ea channel)	Test word selects an analog input channel. A test voltage is applied to the circuit while not disturbing the analog signal from the Tug subsystem.	The voltage is converted and digitized into a word which is sent to the computer.
4. Conditional Monitoring	Power supply voltages.	Status register input bit.

Computer program control is utilized to the maximum extent possible for performance isolation and identification of self test operational modes and failures. Self test does not stimulate or interfere with any vehicle subsystem operation.

Failure of any input or output channel will not result in the loss of any other input or output channel and will not affect the DAU operation, except signals directly related to the respective failed input or output channel. The DAU circuitry is current limited so that a circuit failure does not overload the power supply or other circuit or cause DAU operational degradation. Current limiting circuits include input and output channels multiplexing circuits, transmit and receive circuits and power supply.

The physical and performance characteristics of the DAU are summarized in Table 3.1-6.

Table 3.1-6. Data Acquisition Unit Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	15 lb.
Size	13 x 6 x 7 in.
Power	30 watts
Voltage/Current	28 VDC ±10%/1.1 Amps
Operating Temperature Range	$-65^{\circ}F$ to $+160^{\circ}F$
Non-Operating Temperature Range	$-90^{\circ}F$ to $+203^{\circ}F$
Installation	Standard avionic equipment mounting
Reliability MTBF	10,000 hrs.
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Analog Input Signals	0.0 to ±5 VDC (Differential)
Discrete Input Signals	0.0 to ±6 VDC (Differential) 0.0 to ±28 VDC (Differential)
Discrete Output Signals	0.0 ±0.5 to 5 ±1 VDC

Measurement Processor Unit

The programmable MPU is presently configured as a single functional unit to provide the means of acquiring and formatting data for outputing by the Communication Subsystem. The functional block diagram showing operation of the measurement processor is as shown in Figure 3.1-7.

The unit accomplishes its task by monitoring the data bus and extracting all data whose address compares with that programmed for telemetry. The specific addresses and data handling instructions are loaded by the DMS via the operational data bus based upon the particular mission phase.

The collected data is stored in a buffer memory at locations established by the buffer program instruction. Readout is accomplished by a clock which may be independent of the read-in clock to permit asynchronous transfer of data between two independent systems. The output rate may also be controlled by the user system.

The physical interface with the data bus is through the IU. The measurement processor buffer/user interface provides the capability for the user to asynchronously extract the selected data from the buffer memory in either of two modes: (1) a serial dump of the contents of the buffer memory, or (2) addressable memory data dump limited to selected hardwire measurement inputs.

This interface will include:

- External Clock - User provided timing for buffer memory dumps.
- Data Out - Measurement processor provided lines for serial data transfer from buffer memory to user.
- Data Address - User provided address code lines for specifying locations by selected address of data desired within buffer memory configuration.
- Control Signal - User provided signals for transferring in data address and reading out buffer memory.

The MPU will have a hardwire input capability identical to the DAU with the exception that the quantity of each type of input will differ from that of the DAU. The control of this function will be provided by the instructions contained in the stored program memory and the operation decoder section of the unit. The physical and performance characteristics of the MPU are summarized in Table 3.1-7.

Status and Control Panel

The status and control panel (S&CP) will be mounted external to the Tug and interface with the Tug DMS through an umbilical connector to provide the functions of: (1) displaying status of vehicle subsystems or modes of operation during flight prior to deployment or during ground readiness tests,

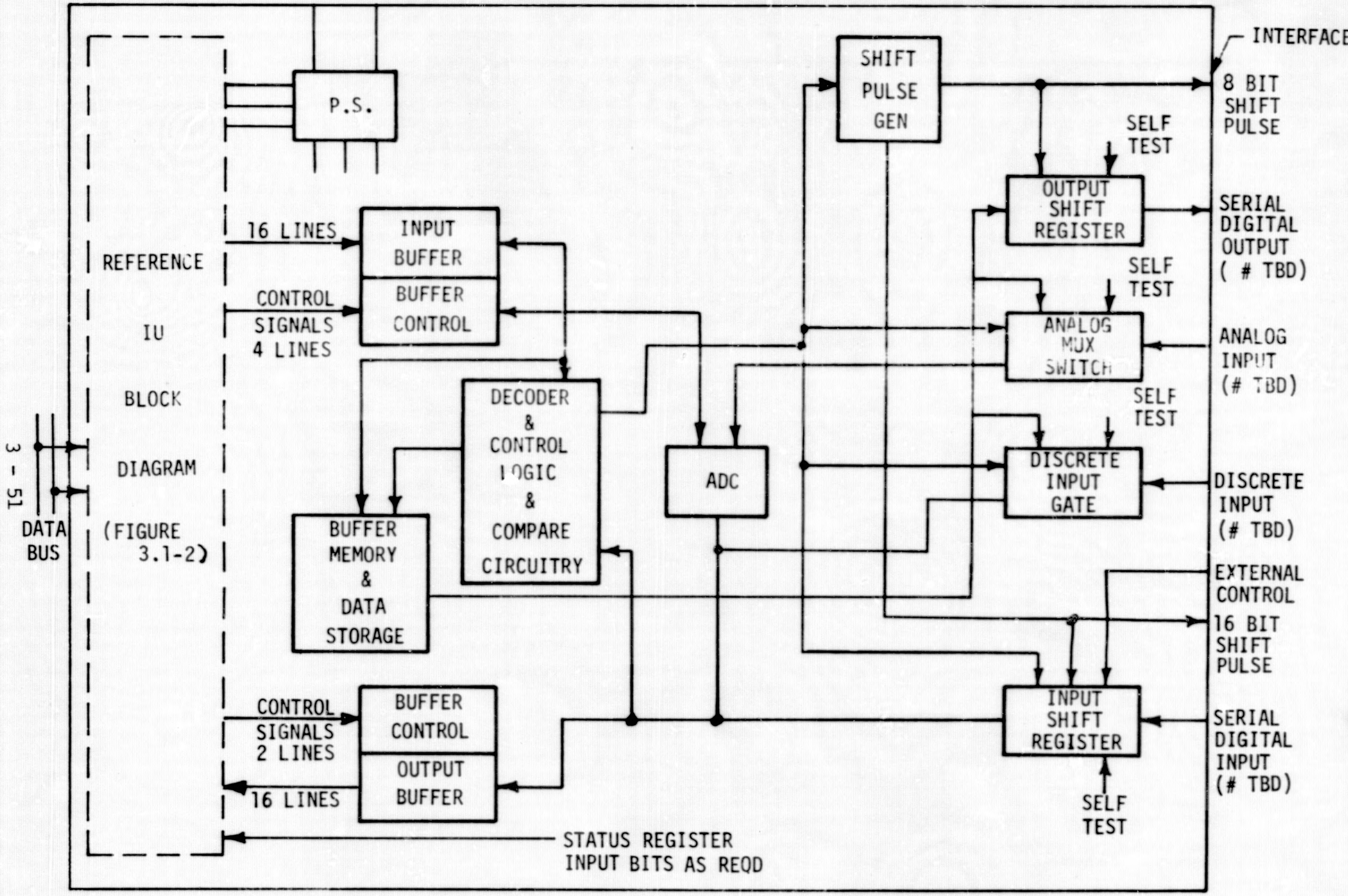


Figure 3.1-7 Measurement Processor Functional Block Diagram

Table 3.1-7. Measurement Processor Unit Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	20 lb
Size	13 x 6 x 9 in.
Power	40 watts
Voltage/Current	28 VDC ±10%/1.5 Amps
Operating Temperature Range	-65°F to +160°F
Non-Operating Temperature Range	-80°F to +203°F
Installation	Standard avionic equipment mounting
Reliability MTBF	10,000 hrs
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Analog Input Signals	0.0 to ±5 VDC (Differential)
Discrete Input Signals	0.0 to ±6 VDC (Differential 0.0 to ±28 VDC (Differential)

(2) control and test during operations prior to deployment of the Tug vehicle, and (3) vehicle state vector update.

To achieve this, the S&CP will illuminate displays as prescribed by serial digital data inputs and transmit data depicting operation instruction established by the operator. The functional internal data processing and control are as shown in Figure 3.1-8.

The S&CP also provides an interface for the monitoring of selected subsystem hardwire measurements and the exercising of direct control of safing functions while in the EOS cargo bay.

When operating through the data bus for control of Tug subsystems, commands initiated from the S&CP will be processed through the DMS control computer prior to execution. To accomplish this, the control computer will interrogate the S&CP interface for data transfer. The control computer will

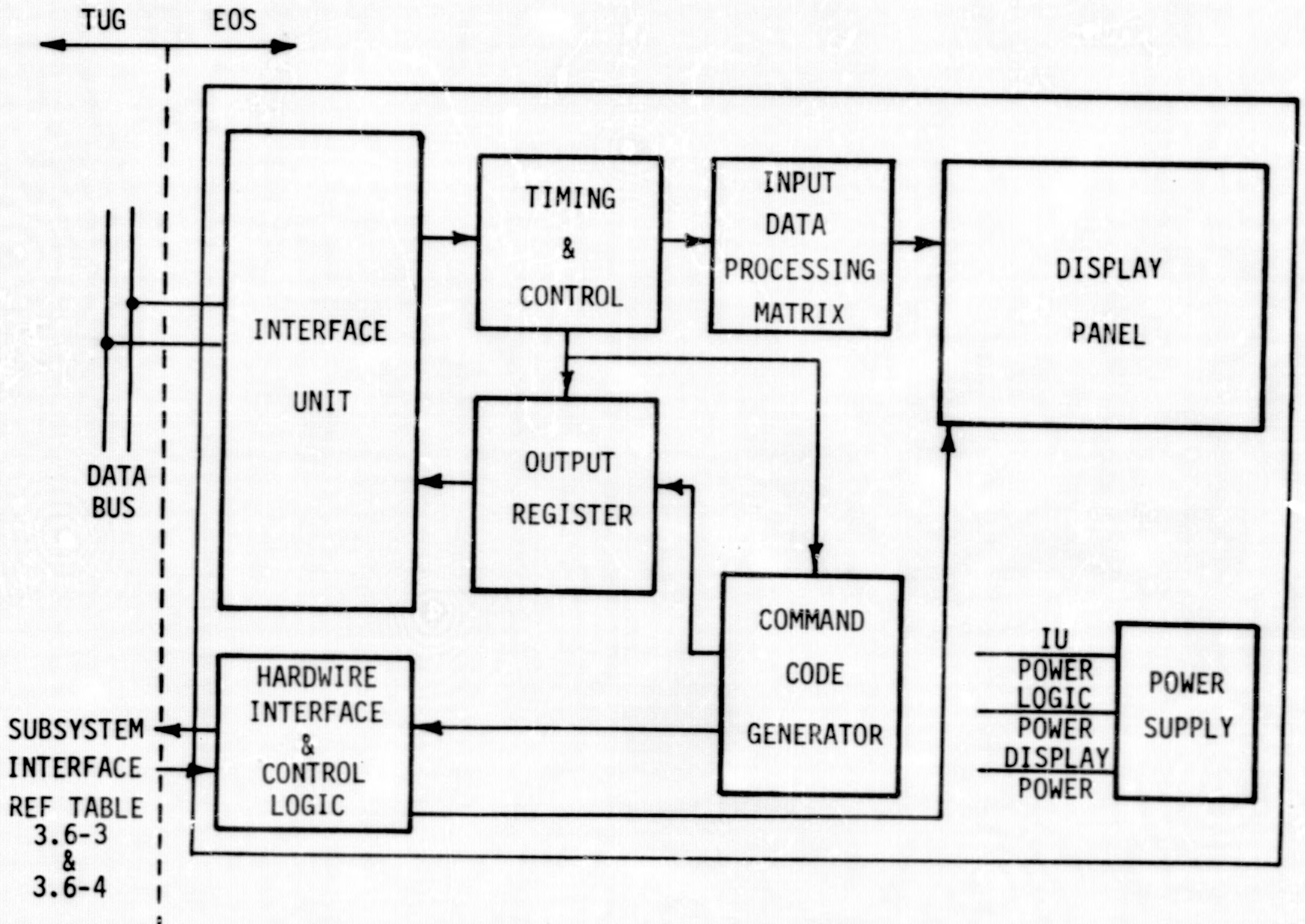


Figure 3.1-8 Status & Control Panel Functional Block Diagram

interpret the instruction and initiate the appropriate sequence to simulate the desired subsystem, acquire the results and relay them to the S&CP for display.

The physical and performance characteristics of the S&CP are summarized in Table 3.1-8.

Table 3.1-8. Status and Control Panel Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	30 lb (Max)
Size	22 x 5.5 x 8.75 in. (Max)
Power	225 watts (Max) all displays energized 40 watts (Max) standby
Voltage/Current	28 VDC ±10%/8A (Max) & 1.4A (standby)
Operating Temperature Range	-65°F to +160°F
Non-Operating Temperature Range	-80°F to +203°F
Installation	Installed on EOS in area accessible to EOS crew
Reliability MTBF	10,000 hr without lamp indicators 4500 hr with lamp indicators
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Input/Output	Compatible with data bus and hardwire control.

Tape Recorder

An on-board tape recorder was added to the baseline DMS configuration to provide the capability to record main engine data during main propulsion burn time. The data is recorded and stored on magnetic tape which is removed and analyzed during the Tug refurbishment/maintenance cycle. Six different main

engine burn times were identified for the baseline Tug mission. The duration of each burn is as follows:

Burn	Duration (Seconds)
First	1290
Second	552
Third	356
Fourth	303
Fifth	8
Sixth	8
Total	2517 Seconds

No specific data rates were identified in the available information pertinent to the Tug main engine design. Therefore, a maximum bit rate was assumed by making a relative comparison between the quantity of measurements identified for the Tug main engine and the quantity identified for the EOS main engine in the NR Phase "B" study. The comparison is a valid one in that the two engines utilize a similar control and instrumentation concept. The maximum bit rate assumed was 5000 bits per second. A slow tape speed of 1.875 inches per second was also assumed to minimize the amount of tape required. Based on these assumptions, the following basic preliminary requirements were established for the selection of the on-board tape recorder:

Tape Record	Digital (NRZ)
Tape Speed	1.875 IPS
Packing Density	2700 BPI
Tape Length	450 ft

Tape recorders are available off-the-shelf to accomplish the engine firing data recording specified. Slight deviation in some parameters such as tape speed may be required, however, other parameters such as length of tape may be varied to compensate for this eventuality. The physical and performance characteristics of a Kenologic Corporation tape recorder Model RSL now available to accomplish the recording are shown in Table 3.1-9.

Data Bus

The data bus cable harness and the IU's are configured to provide the transmission media between all potential users of the data bus and to provide means of coupling data on and off the data bus. The transmission cable is a

Table 3.1-9 Tape Recorder Characteristics Summary

Physical Characteristics	
Item	**Characteristics**
Weight	5.5 lb
Size	6 x 5 x 5 in.
Power	6 watts
Tape	Width: 1 in. Length: 450 ft
Operating Temperature Range	0°F to +125°F
Operating Pressure Range	27 psia to 10^{-7} TORR
Performance Characteristics	
Parameter	**Performance**
Data Input	Digital-Non-Return to Zero (NRZ)
Tape Speed	1.875 in./sec

two conductor twin axial, twisted shielded, jacketed cable. The transmission cable operates as a balanced pair terminated at each end. Transmitter/receiver remote terminals are connected in parallel to the cable as shown in Figure 3.1-9. All terminals are transformer coupled to the data bus. Isolation resistors provide 200 ohms impedance between the transmission cable and the receiver/transmitter coupling transformer to prevent the data bus from becoming inoperative in the event of a terminal failure. The bus has the capacity for up to 31 terminals.

The data on the data bus will be in the form of a message which will contain one to 16 words. The first word of each message is a control word which may be followed by a maximum of fifteen data words.

The data bus control word is as shown in Figure 3.1-10. The control word format is as follows: (1) the control word sync field consists of a nonvalid Manchester code, (2) the three bit control field contains three Manchester "zeros", (3) the five bit address field specifies the terminal that shall receive a given message, (4) the transmit-receive bit specifies that the terminal will either transmit or receive data words, (5) the four bit mode field specifies the mode or type of operation that the addressed

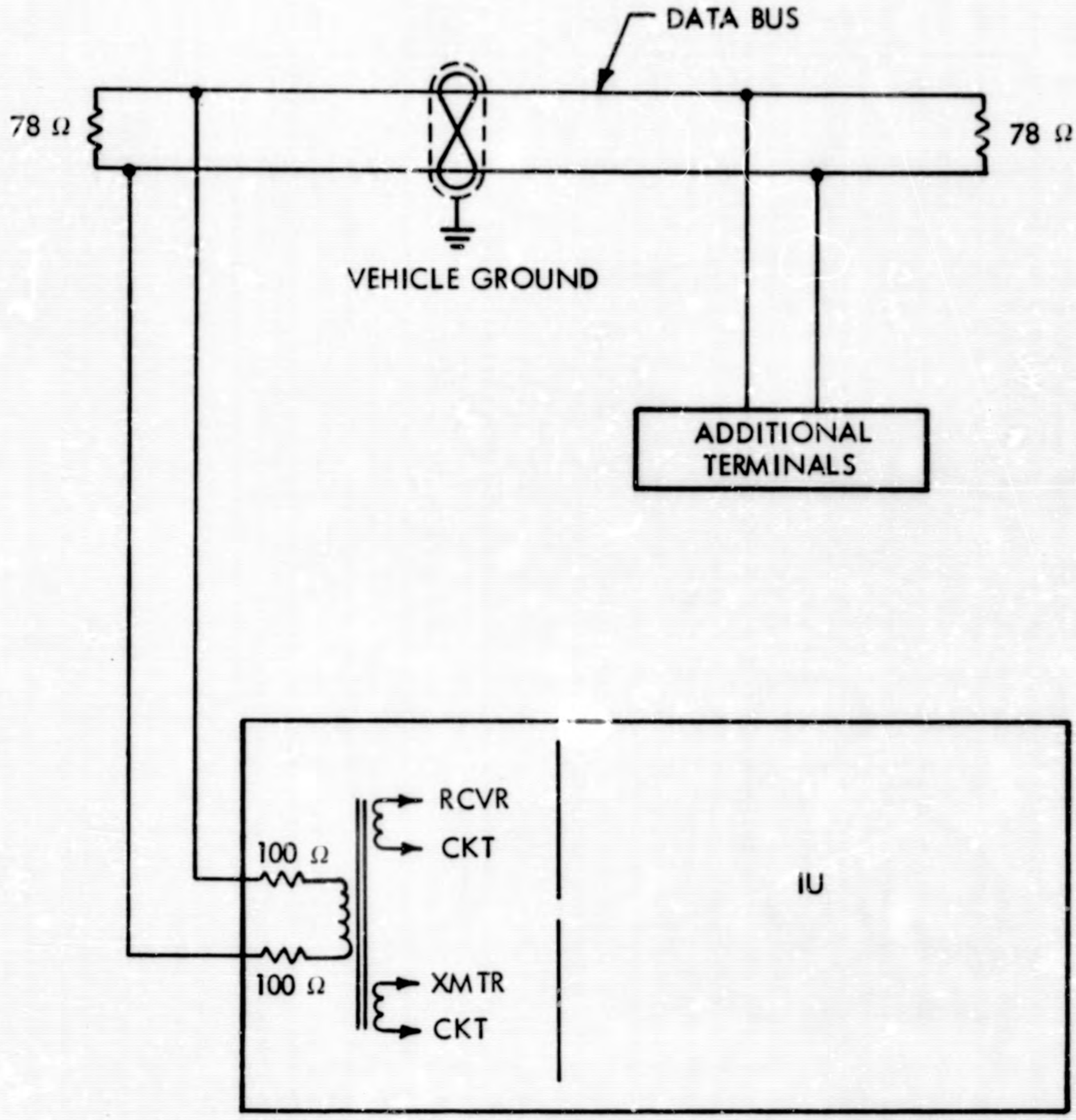


Figure 3.1-9 Data Bus Interface

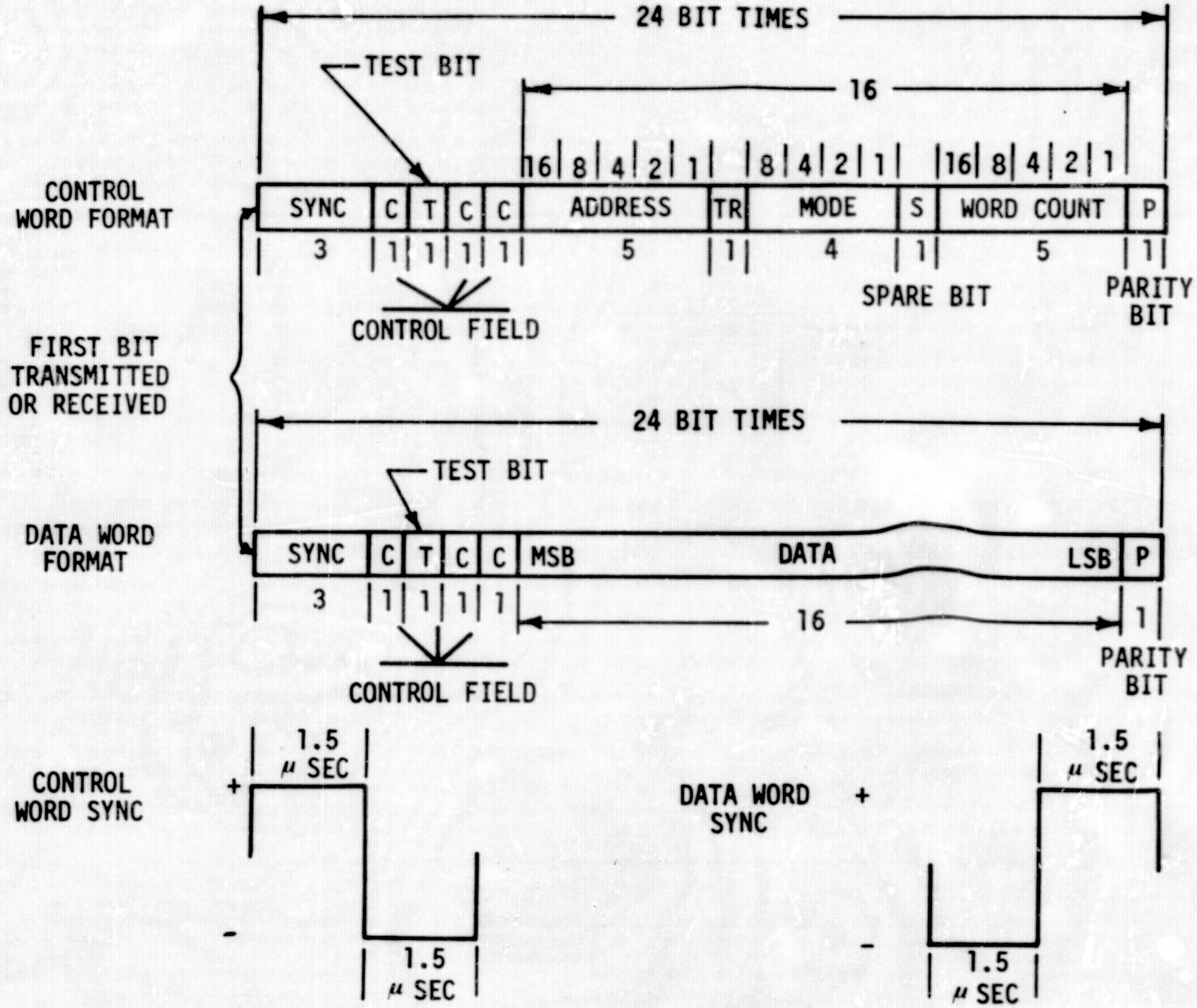


Figure 3.1-10 Data Bus Word Format

terminal is to perform, (6) the word count field incidates the message word count, (7) all words include a test bit. The test bit of control words and data words indicates to the computer one or more of the following terminal conditions: (1) there was a parity or non-valid word in the transmission, (2) the unit has detected a self-test out of tolerance, (3) the last bit of all words will be a parity bit. The parity of all words will be odd "ones".

The data bus word is shown in Figure 3.1-10. The data word control field, test bit, and parity bit are the same as for the control word described above. The data word sync field also consists of a non-valid Manchester code; however, the data word sync may be preceded by a no data time (dead time) of 10 microseconds maximum. The data word data field is 16 bits and may contain any information required for the operation of the system. The data on the data bus is in the form of serial, digital coded signals. The data bus code is Manchester 11 Bi-phase-L at a rate of 1.0 megabit per second.

3.1.5 Alternate Design Approach

The simplex data bus system baselined for the Tug DMS system is the most practical approach within the confines of the Tug prime design drivers of weight and power. However, because of the simplex system approach, mission reliability has not been optimized. With some degree of impact to the two prime drivers a more reliable system may be provided. This alternate approach would require the addition of a second computer and the cross-strapping of critical functions between DAU's.

The vehicle guidance and navigation functions would be contained in one "guidance" computer. The remaining vehicle functions would be contained in the second "control" computer. The guidance computer would be assigned the task of performing all guidance and navigation functions and monitoring the performance of the control computer. The task assigned to the control computer would be the execution of attitude thruster control and thrust vector control (TVC) as directed by the guidance computer, monitoring the guidance computer operation, vehicle sequencing and all data management tasks.

In this approach, the data acquisitions and calculations associated with guidance and navigation would be performed by the guidance computer. When the guidance computer detected a need to initiate a TVC or attitude control thruster command, it would direct the control computer to issue the appropriate control function.

The guidance computer would monitor the command as issued by the control computer to verify that the proper control commands were issued. Prior to issuing a command under the direction of the guidance computer the control computer would perform a reasonableness test to evaluate if the command instruction as received from the guidance computer was within reason for that portion of the mission phase. Either computer detecting a no-go condition would result in the command not to be executed.

The execution of normal vehicle sequencing would be in accordance with preprogrammed instructions contained in the control computer. Any control commands issued by the control computer would be monitored by the guidance

computer to assure that the proper commands were issued for that phase of the mission. Again, either computer detecting an unacceptable condition would result in inhibiting the execution of the command.

To avoid loss of the Tug vehicle in case of failure of one of the computers, both computers would have the minimum programming required to perform the critical functions of the other computer. This mode of operation would be initiated by a self-test failure of one of the computers or under control of the up-link command system.

To avoid loss of a critical function of a subsystem, all critical functions would have their control cross-strapped to the output of two distinct DAU's. This will enable continued use of the subsystem should one DAU fail.

Interfacing with vehicle subsystem in this approach would be the same as that described in the baseline system. However this approach would require the existence of a direct interface between the two computers and possibly a small section of shared memory.

This approach provides protection against unscheduled events occurring because of a failed computer. It also provides for failure of one of the DAU's. However, it does impact DMS weight and power requirements.

3.2 GUIDANCE, NAVIGATION AND CONTROL

The primary Guidance, Navigation and Control Subsystem (GN&C) is comprised of a strapdown 3-axis inertial measurement unit (rate gyros, accelerometers, and associated electronics), a gimballed star tracker, a horizon edge tracker, an autocollimator, and an engine control assembly (auxiliary propulsion system engine drivers and main engine gimbal servo electronics) as shown in the functional block diagram of Figure 3.2-1.

The GN&C subsystem receives a state vector handoff from the EOS prior to electrical demating. From that point the subsystem elements provide navigation and guidance data to the central computer through appropriate DAU's in the DMS. The computer uses this data to determine vehicle position, attitude, and velocity. Based on this state vector determination and programmed mission timeline requirements, the computer generates main engine thrust commands, steering commands, and auxiliary propulsion system (APS) stabilization and control commands. The commands are delivered to the appropriate components through the DAU's and engine control assembly. The GN&C subsystem and DMS work in conjunction to provide autonomous control for performance of the Tug mission. Autonomy is relinquished in the normal mode only for man controlled Tug/payload docking. For abnormal modes, the subsystem design provides for a ground override capability to interrupt atuonomous operation.

Included in the GN&C subsystem is an autocollimator to determine relative structural alignment between the star tracker and horizon tracker mounting bases. The alignment data is fed to the central computer where it is used to

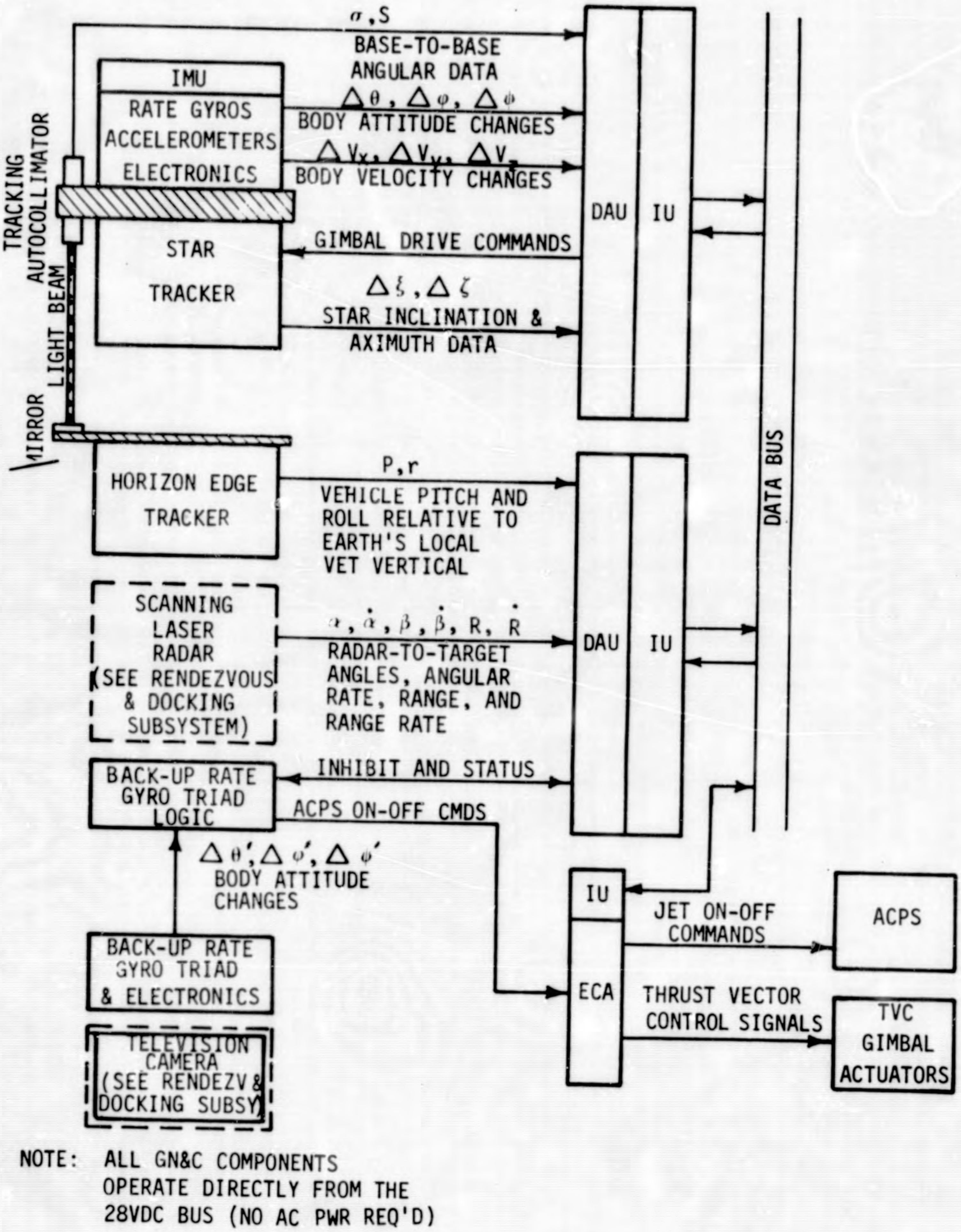


Figure 3.2-1 Guidance, Navigation, and Control Subsystem Functional Block Diagram

reduce the bias error between the star tracker and horizon tracker and hence improve star/horizon navigation accuracy.

To provide backup vehicle rate stabilization in the event of a primary GN&C subsystem or DMS failure, a completely analog separate rate stabilization system is incorporated in the subsystem design. The backup system operates independent of the DMS. It consists of a rate gyro triad and associated logic package wired directly to the APS components. The backup stabilization system is enabled by the loss of a computer inhibit signal and the existence of vehicle angular rates indicating an unsafe condition.

3.2.1 Requirements

GN&C and GN&C related requirements can be categorized as: (1) mission requirements (functional and performance), (2) ground rules and guidelines, (3) subsystem requirements (functional and performance), and (4) installation constraints.

Mission Requirements

In the baseline mission the Tug/Payload combination will be carried to a 100 n.m., 28.5° inclination orbit where it will be deployed and activated. The Tug will then phase in low earth orbit, perform a transfer burn to 100 n.m. x 19,300 n.m. orbit (with a 2° plane change), coast to 19,300 n.m., perform a plane change (26.5° to 0° inclination) and circularization burn, and deploy the up-pay-load. Following deployment of the up-payload, the Tug will transfer to an appropriate phasing orbit and back to a geosynchronous orbit to rendezvous with a down-payload 6000 n.m. either up or down range. The Tug will then dock with the payload, phase in geosynchronous orbit, burn and inject into 270 n.m. transfer orbit (with a 26.5° plane change), perform a plane change (26.5° to 28.5° inclination) and circularization burn, transfer to a 100 n.m. x 100 n.m. rendezvous orbit, and perform a terminal rendezvous to within 300 meters of the EOS. Docking operations will be performed by the EOS.

The mission performance requirements as specified in the study plan are defined below.

Function	Performance Requirement: Low Earth Orbit	Performance Requirement: Geosynchronous Orbit
Autonomous Orbit Injection		
- Position	10 KM	50 KM
- Velocity	5 M/sec	5 M/sec

Alternate missions which include delivery of payloads to geosynchronous orbit and empty return, and empty ascent for retrieval of payloads do not impose any more stringent requirements than the baseline mission.

Ground Rules and Guidelines

1. Ground-based maintenance repair and/or refurbishment.

2. Minimal on-orbit functional test prior to Tug/Shuttle separation.

3. Mission time 7-days; 6-days detached from the Shuttle and 1-day in Shuttle bay.

4. Fail-safe operation will be provided. Fail-safe operation is interpreted as precluding destruction of the payload or any unsafe condition for the Shuttle or the Shuttle crew.

5. The Tug will be provided a navigation update from the Shuttle prior to Tug/Shuttle separation (2 KM and 2 M/sec each axis).

6. The GN&C will be unpowered during Shuttle ascent and descent except for heaters as required.

7. The GN&C will be designed for a 20 mission life with refurbishment after each mission.

8. Autonomous operation with ground override capability is considered a requirement except for docking operations.

9. Attitude during burns will be maintained to within 0.1° of the desired attitude.

Subsystem/Component Requirements

Specific GN&C subsystem and/or component requirements were outlined in the Tug study plan, and these requirements are being adhered to with changes or deviations only where coordinated with the NASA in meetings or telecons. The subsystem and/or component requirements being used as a basis for this point design study are outlined in Table 3.2-1.

Installation Constraints

Specific installation constraints apply to several of the GN&C components as follows:

1. The inertial measurement unit (IMU) should be located such that its principal axes (X, Y, and Z) are aligned to a reasonable tolerance to the corresponding vehicle principal axes. This assumes that the APS engines and main engine gimbal actuators are also aligned to the vehicle principal axes.

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Table 3.2-1. G, N & C Subsystem/Component Requirements

Component	Performance Requirements (3σ)	Function	Requirement Source	Comment
1. Strapdown, 3 Axis Inertial Measurement Unit		-Provides short term inertial attitude and state vector information. -Provides angular rate information for vehicle stabilization.	Guidelines + Telecon Coordination	The guidelines implied that a separate rate gyro pkg and accelerometer pkg were required. The combining of these 2 functions in a single Inertial Measurement Unit was coordinated with the NASA in a telecon on 11-16-71.
-Random Drift	$\pm 0.1°$/hr			
-G-Sensitive Drift	$\pm 0.3°$/hr/g			
-Accel Threshold and Resolution	± 0.1 M/sec			
2. Gimballed Star Tracker		-Provides attitude update to the Inertial Measurement Unit. -Provides star angle information for star/horizon navigation (state vector update).	Guidelines + Telecon Coordination	The guidelines called out 2 gimballed star trackers. The use of a single tracker is based upon the adequacy of sequential (rather than simultaneous) star fixes. Coordinated with the NASA in telecon on 11-16-71.
-Azimuth and Evaluation	± 30 Sec			
-Gimbal Freedom	120° X 120°			
3. Horizon Tracker		-Provides earth local vertical information for star/horizon navigation (state vector update).	Guidelines	The 0.1° instrument accuracy was specified for low earth orbit only - The requirement has been extended to synchronous orbit for navigation accuracy.
-Instrument Accuracy	$\pm 0.1°$			
-Altitude Range	100 - 20,000 NM			
-Tilt Range	$\pm 5°$			

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Table 3.2-1. G, N & C Subsystem/Component Requirements (Cont)

Component	Performance Requirements (3σ)	Function	Requirement Source	Comment
4. Autocollimator -Linear Range -Operating Distance -Accuracy	 ±30 Min 13 ft ±1%	-Provides inter-sensor alignment between the star tracker and horizon scanner.	NR Derived	The guidelines do not specify an autocollimator. Since the navigation accuracy sensitivity to inter sensor alignment adds to instrument error, an autocollimator was added.
5. Engine Control Assy	15 On-OffCMDS 2 Gimbal Servo AMPL.	-Provide the capability to fire APS engines and to gimbal main engine from the data bus.	NR Derived	Required for operation of 14 APS engines and 1 main engine.
6. Backup Stabilization Assembly -Threshold and Resolution -Interfaces -Functional Life	 ±0.01°/Sec No DMS or Primary Power Dependency 30 Min	-Provides rate stabilization for fail-safe operation in the event of a primary G, N & C, DMS, or power system failure. -Provides logic for the above emergency takeover.	Guidelines and NR Derived	Threshold and resolution requirements based upon an estimate of safe vehicle rates when in the vicinity of another orbiting body.

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2. The star tracker should be mounted directly on the same base (navigation base) as the IMU and closely mutually aligned, prior to installation in the spacecraft. This close alignment is required to optimize the accuracy of the attitude reference update by the star tracker.

3. The star tracker/IMU combination should be located such that the star tracker will be able to utilize its full 120° x 120° field of view in a direction which allows a sufficient number of trackable stars to be brought into its field of view during phasing orbits and coast periods while simultaneously tracking the earth's horizon with the horizon tracker. The star tracker/IMU combination should also be located such that autocollimation between it and the horizon tracker can be accomplished to reduce inter-instrument alignment errors.

4. The horizon tracker must be located such that the earth's horizon may be tracked during phasing orbits and coast periods while simultaneously tracking stars. Furthermore, sufficient field of view must be provided for each tracker head to permit accommodating both low earth orbit and synchronous orbit with subtended angles (155.58° at 100 n.m. and 16.96° at 20,000 n.m.) plus sufficient operating clearances.

5. The Backup Stabilization Assembly (BSA) principal axes (X, Y, and Z) should be aligned to the corresponding vehicle principal axes (as defined by APS engine locations). The alignment achieved through primary and secondary structure is sufficient.

3.2.2 Subsystem Trades

Star Tracker/Horizon Tracker Alignment Alternatives

The need to track the earth's horizon and stars simultaneously imposes particular installation constraints on both the horizon tracker and the star tracker (i.e., the horizon tracker must look at the earth while the star tracker looks away from the earth). This dictates that the two sensing systems be located on opposite sides of the vehicle, separated by as much as 15 feet. Furthermore, the horizon tracker heads are required to cover all earth subtended angles, from low earth orbit (approximately 160°) to synchronous orbit (approximately 17°). This imposes a requirement for very large scan angles on the tracking heads leading to a requirement to deploy the horizon tracker heads after the Tug is removed from the EOS bay in order to achieve the relatively large unobstructed field of view (FOV).

It is obvious that any relative misalignment between the star tracker coordinate system and the horizon tracker coordinate system contributes a bias error to the star/horizon angle measurements and hence reduces navigation accuracy. The fact that the sensors are necessarily located far apart, coupled with the need to deploy the horizon tracker heads, indicates that this error may be rather large (15 $\widehat{\text{min}}$ to 30 $\widehat{\text{min}}$) when all structural deformations

caused by handling (ground and orbit), thrusting, docking and undocking, and thermal time histories (ground, orbiter bay, space environment, etc.) are considered.

Because of the concern for the effects of structural misalignment between the navigational sensors analyses were conducted to grossly evaluate injection error sensitivity to this misalignment.

Error analysis data was generated by simulating major portions of the Tug mission timeline; for example, the sequence of orbital phasing, transfer injection burn, midcourse coast, and orbit insertion into geosynchronous orbit. Position and velocity errors at the initiation of a mission timeline segment are propagated through successive coasting/navigation and thrusting/guidance segments in order to determine the resultant injection error.

Error Sources

Due to the short duration of the study, the list of error sources was kept to a minimum, consisting of specified point design values, state-of-the-art values for non-specified parameters, and estimates of sensor mounting bias uncertainties. The error sources are grouped into two categories, observational sensing and inertial measuring. This group is consonant with the two mission operations of coasting/navigation and thrusting/guidance. Values for the major error sources are listed in Table 3.2-2.

Two values of navigational sensor misalignments were evaluated; (1) 30 $\widehat{\text{min}}$ representing misalignment without an autocollimator, and (2) 15 $\widehat{\text{sec}}$ representing misalignment with an autocollimator.

Performance Evaluation

A summary of the results of the performance analysis, showing state vector uncertainties at the initiation and termination of each mission phase, from Tug deployment through synchronous orbit injection, are given in Table 3.2-3. These results indicate that satisfactory injection accuracy can be achieved if an autocollimator (or equivalent) is employed, and that structural misalignment of 30 $\widehat{\text{min}}$ yields unsatisfactory injection accuracies.

These results are not considered conclusive since navigation accuracy is sensitive to several other parameters, requiring a parametric analysis outside the scope of this study to evaluate. Because of the trend of these preliminary results however, an autocollimator has been included in the GN&C baseline.

A more comprehensive discussion of this performance analysis complete with detailed supporting data is contained in Appendix A of this volume. Date contained in References 3.2-1, 3.2-2, and 3.2-3 were used in the performance of this analysis.

Table 3.2-2. Error Sources

Error Designation	3σ Value	Origin
Initial State Errors		
3 Components of Position	2KM per axis	Shuttle Handoff
3 Components of Velocity	2 M/S per axis	Shuttle Handoff
Initial Attitude Misalignment		
3 Components of Attitude	0.1 Deg per axis	Mission Spec
Gyro Errors		
3 Components of Random Drift	0.1 Deg/Hr	Mission Spec
Mass Unbalance Drift	0.3 Deg/Hr/G	Mission Spec
Torquer Scale Factor Uncert	0.01%	State-of-art
Accelerometer Errors		
3 Components of Bias Error	0.3X10-4 G	State-of-art
Scale Factor Uncertainty	0.01%	State-of-art
Nonlinearity	0.1X10-4 G/G^2	State-of-art
Horizon Tracker Errors		
Total Random Errors	6 min	State-of-art
Star Tracker Errors		
Total Random Errors	30 sec	State-of-art
Optical Alignment Uncertainty		
Without Autocollimator	30 min	Estimated
With Autocollimator	15 sec	State-of-art

3.2.3 Subsystem Operation

The sequence of operations that the Tug must perform to accomplish its mission may be divided into a series of phases within which unique functions need to be satisfied. Satisfaction of these functions requires the operation of specific elements of the GN&C subsystem within a particular phase. The following is a discussion of these operational phases.

Initial On-orbit Power Up

Power application, except for necessary environmental control elements, will occur after the Tug is deployed from the EOS cargo bay. While the Tug

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Table 3.2-3. Autocollimator Tradeoff Data for Representative Tug Mission

Coasting Navigation Parameters:
- Measurement Frequency - 2.5 and 1 Minute
- Star Tracker Random Noise - 30 Sec
- Horizon Tracker Random Noise - 6 Min
- Optical Alignment Uncertainty - 30 Min
  - With Autocollimator - 15 Sec

Thrusting Navigation Errors:

See Table 3.2-2

Mission Timeline		Autocollimator				Without Autocollimator			
		Position Error		Velocity Error		Position Error		Velocity Error	
Event	Cumulative Time Hrs	KM	Ft	M/Sec	FPS	KM	Ft	M/Sec	FPS
Orbit Phasing	0	3.464	11376	3.464	11.38	3.464	11376	3.464	11.4
	5	2.805	9204	3.3	10.7	4.206	13800	4.9	16.1
100X19323 Burn	5	2.805	9204	3.3	10.7	4.206	13800	4.9	16.1
	5.4	7.014	23011	8.2	27.0	10.159	33329	10.6	34.8
100X19323 Coast	5.4	7.014	23011	8.2	27.0	10.159	33329	10.6	34.8
	10.4	15.066	49431	1.5	4.9	>100		>15	
Orbit Insertion Burn	10.4	15.066	49431	1.5	4.9	Does Not Satisfy Mission Requirements			
	10.6	15.745	51659	4.7	15.5				
		G&N Capability Satisfies Mission Requirements of 50 KM and 5 M/S							

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North American Rockwell

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remains captive, however, all engine thrusting capability will be inhibited. Self-testing of the various subsystems will be accomplished in order to verify their proper operational status. At the conclusion of these tests, a transfer of position, velcoity, and clock data along with coarse attitude data from the EOS to the Tug will be accomplished. The Tug will be released from the EOS, and once it is an appropriate distance from the EOS, an automatic attitude alignment program will be initialized. The IMU, horizon edge tracker, star tracker, and APS will work together to find and maintain the state vector of the Tug at all times. Star measurements will be taken at 150 second intervals during this period in preparation for the next phase of operation - main engine burn.

Main Engine Burn

During the main engine burn(s), the TVC, APS, and IMU will be actively engaged in maintaining a programmed attitude. The IMU will measure acceleration over the burn period and the guidance computer will use the measured data to calculate the precise engine cutoff point. When all program requirements have been satisfied, engine cutoff will occur and transfer to the ascent coast phase will commence.

Ascent Coast Phase

During this period the horizon edge tracker will continuously monitor the earth's subtense angle (requiring an active APS), the star tracker will search for stars (with 90 degree or near 90 degree separation) and take measurements at one minute intervals, and the guidance computer will update the attitude data received from the IMU. The autocollimator is active at all times to maintain knowledge of alignment between the horizon edge tracker and the star-tracker/IMU mounting bases. This coast phase is expected to require about 5.5 hours after which entry into the circularization burn phase is initiated.

Circularization Phase/Payload Deployment

The end of the coast phase will place the Tug near the ideal apogee point on an elliptic curve. The main engine will be ignited and the necessary velocity change will be sensed by the IMU/guidance computer along with the required plane change. The burn will place the Tug in a circular orbit at geosynchronous altitude and zero degree inclination. The payload will be deployed at this time and the Tug will again commence taking star-tracker/horizon edge tracker measurements in order to update the stored IMU data. The Tug may continue taking these measurements for up to 72 hours and will utilize the gathered data to reduce position, velocity, and attitude errors prior to initiating a return-to-low-earth-orbit burn.

Payload Deployment/Payload Retrieval

If the mission requires that the Tug find and retrieve a return payload from orbit, it may be necessary to enter a Tug/payload convergence mode. During this period, the Tug orbital velocity will be changed such that the Tug

and payload will converge toward a point in space that will allow the rendezvous and docking subsystem to become the primary source of guidance data.

If only retrieval of a payload from geosynchronous orbit is required, the 72 hour convergence mode may or may not be necessary depending on whether or not the low earth orbit (LEO) to geosynchronous earth orbit (GEO) was an intercept transfer orbit or not.

Rendezvous and Docking

This mission phase is covered in the Section 3.3.

Return to EOS

After a stay in GEO for some minimum period of time, position, velocity, and attitude errors will be reduced to values which must lie within permissible bounds. At the precise time, as determined by the computer stored program, the main engines will be ignited and the Tug will enter a 270 X 19,300 NM by 26.5 degree inclination transfer orbit. After engine cutoff, the horizon edge tracker, star tracker, IMU, autocollimator, and APS will remain on active status during this and similar periods. Tug position and velocity will be updated by periodically taking appropriate star and earth measurements. Sometime prior to reaching the orbit perigee point, data gathering will cease and the main engines will be ignited. Another main engine burn will place the Tug into a 270 NM circular orbit with a 2° plane change and phasing will commence for final descent to LEO. During the phasing period, the aforementioned subsystems will remain on active status and will continuously gather data for use in reducing state vector errors. At a predetermined point and time, ignition of the main engine will occur and the Tug will be placed into a 100 X 270 NM elliptic transfer orbit. Again, the navigation sensors will gather earth and star data to improve the inputs received from the inertial system. Upon reaching the transfer orbit perigee point, the final main engine burn will be initiated. The burn will place the Tug in a 100 NM by 28.5° inclination circular orbit. The Tug will improve its knowledge of position and velocity utilizing the previously discussed subsystems during the phasing-with-EOS period. The rendezvous subsystem will be activated and the search for the EOS will commence. Upon detection of the EOS, the rendezvous system data will be used by the guidance computer to begin an approach sequence of operations. The object of the operations is to bring the Tug within 300 meters of the EOS and begin the stationkeeping phase. The rendezvous and docking operations are discussed under that section. All subsystems not required for stationkeeping will be deactivated. The Tug will await EOS activity or further instructions from earth-based or EOS personnel.

3.2.4 Component Characteristics

Research into available candidates for GN&C subsystem component selection yielded information that resulted in the establishment of a baseline configura-ation in which is made extensive use of technology to be developed through the 1976 time frame. The technology to be developed does not, however, represent a high risk effort. This technology is basically in the area of packaging optimization and as such can be achievable with evolutionary or minimum design development.

This section provides detailed information concerning the components selected for the baseline including theory of operation, selection rationale, and component characteristics.

Gimballed Star Tracker

Baseline Selection: Kollsman Model KS-199 utilizing projected 1976 technology. Table 3.2-4 summarizes the physical and performance characteristics of the unit as detailed in Reference 3.2-4.

Theory of Operation

The Kollsman Solid State Spaceborn Star Tracking System selected for a baseline is a fully automatic source of data used for orienting, pointing and/or stabilizing a vehicle in space. Figure 3.2-2 is a functional block diagram of the gimballed star tracker.

The significant advantages of the system include:

1. Solid State Sensor - eliminates the requirement for a high voltage power supply (needed for photomultipliers, image dissectors, vidicons, etc.) and the associated voltage breakdown problems.

2. Solid State Sensor - provides high reliability and relative insensitivity to stray light.

3. Solid State Sensor - provides the capability to tolerate exposure to direct sunlight.

4. The over-all system electronics includes microminiaturized circuits, thereby providing increased reliability as well as minimum weight, size and cost effectiveness.

The Kollsman Solid State Spaceborne Tracking System provides a spacecraft with locally-derived present attitude information and continuously computed correction signals. It can acquire and track a star even when a satellite or spacecraft is rotating at a rate of $1/2^{o}$/second.

The tracking system consists of the tracker and associated electronics package. The system is completely self-contained and can be digitally commanded by an external computer to point the telescope anywhere within a 120 degree square angle. It is capable of tracking stars of +2.0 silicon magnitude or brighter - approximately 100 stars.

Table 3.2-4. Gimballed Star Tracker Characteristics Summary

Physical Characteristics	
Item	**Characteristics**
Weight	25 lb.*
Size	Tracker outline 15 x 9.63 x 8.88 in. Electronics outline 9 x 7.5 x 6.5 inc.
Power	18 watts
Voltage	28 VDC
Operating Temperature Range	$0^{o}F$ to $+150^{o}F$
Pressure	Operates in vacuum
Vibration (all axes)	0.2 g^2/Hz 10-2000 Hz random 15 g sinusoidal 10-2000 Hz sine
Shock (all axes)	75 g for 2.5 ms sawtooth pulse
Acceleration (all axes)	11.3 g duration 4.5 minutes/axes
Installation	Must look upward (Z-axis) and have a $\pm 60^{o}$ FOV for each axix.
Reliability MTBF	16,000 hrs.

* Detailed Weight Breakdown

Telescope and star detection electronics	1.9 lbs.
Star Signal processing electronics	1.1 lbs.
Digital interface and computation	1.2 lbs.
Base and two gimbal structure	16.0 lbs.
Servo drive and readout electronics	2.5 lbs.
Power conditioning	2.0 lbs.
Total	25.0 lbs.

Table 3.2-4 (Cont'd)

Performance Characteristics	
Parameter	Performance
Accuracy	10 arc seconds RMS per axis (1σ)
Recognition	Accuracy and track stars +2.0 magnitude (silicon) or brighter in space environment (100 stars)
Sun/Earth Impingement	Operate as close as 30° to sunline, 13° to earth's surface
Gimbal Travel	$\pm 60^{\circ} \times \pm 60^{\circ}$
Inputs	2 axis digital pointing commands for acquisition. Accepts 50 digital cmds/sec.
Outputs	2 axis pointing angles (digitized). Serial differential at 625 KHz rate.
Articulation	Pitch and roll type gimbals.
Search Pattern	Programmed by computer
Acquisition Time	100 milliseconds
Bandwidth	1 Hz
Optics	
Type	Reflective
Aperture	2.5 inches
Focal Length	3.1 inches
Field of View (FOV)	1° cone
Sensor	Silicon photodiode

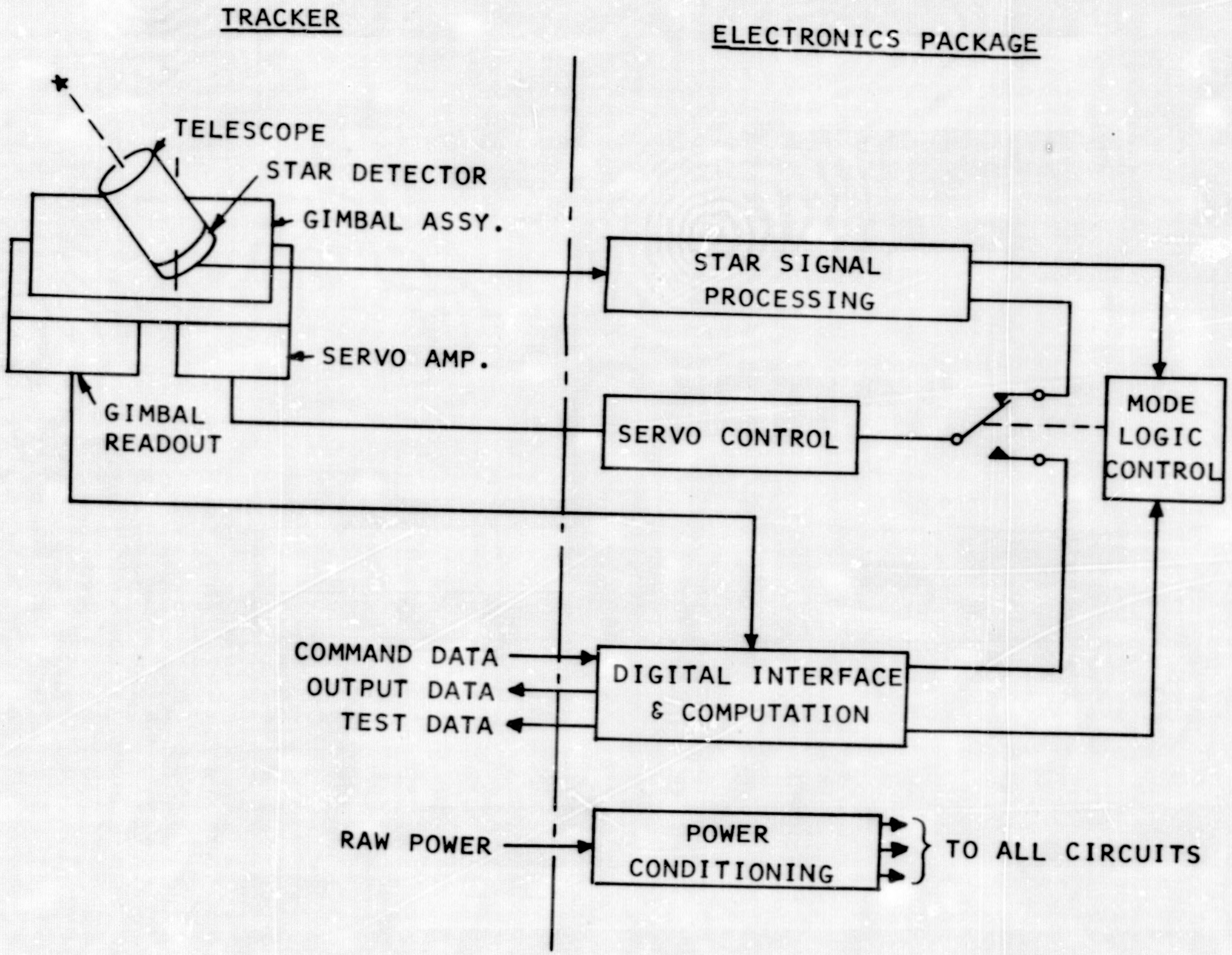


Figure 3.2-2 Gimballed Star Tracker Functional Block Diagram

The tracker is made up of a photoelectric telescope, two positioning gimbals, precision angle pick-offs and a housing assembly. The telescope contains reflective and refractive optical elements, a nutating fiber scanner and a silicon photosensor.

The telescope provides modulation of the star light entering it to produce basic tracking and error signals. The unique design of the telescope permits operating as close as 30 degrees to the sunline and 13 degrees to the earth's surface.

The positioning gimbals permit precisely controlled movement of the telescope. Each gimbal is provided with a direct drive dc torque motor and precision angle pick-off. The torque motors respond to either external digital command signals or telescope-derived error signals.

Component Selection Rationale

Customer specification of a gimballed star tracker with a pointing accuracy of $\pm$ 30 sec. (3σ) limited the equipment candidates. The baseline selection of the Kollsman KS-199 tracker was based on the following considerations:

1. Meets accuracy requirement ($\pm$30 sec. 3σ).
2. Provides large gimbal angles ($\pm60^{o}$ azimuth and elevation) which allows use of single tracker for both in-plane and out-of-plane star fixes.
3. Development status - modification of existing flight qualified design.
4. Silicon detector allows exposure to direct sunlight.

Horizon Tracker

Baseline Selection: Quantic Industries, EDT-312A Model IVA. Table 3.2-5 summarizes the physical and performance characteristics of the unit as detailed in Reference 3.2-5.

Theory of Operation

The Quantic Industries horizon edge tracker, adopted for a baseline, utilizes sensors operating in the 14 - 16 micron carbon dioxide (infra-red) absorption band. Exceptionally long life is attained by using radiation thermocouple detectors, which, unlike thermistor bolometers, eliminate the need for mechanical dither about the horizon line. Careful thermal design, plus a thermistor-controlled temperature servo, maintains the detector section at a near-constant reference temperature as environmental temperatures vary. Figures 3.2-3 and 3.2-4 are functional block diagrams of the horizon edge tracker.

Table 3.2-5. Horizon Tracker Characteristics Summary

Physical Characteristics	
Parameter	**Performance**
Weight	Per sensor 7.75 lb. 4 sensors 31.0 lb. Electronics Unit 14.0 lb. Total Assembly 45.0 lb.
Size	Per sensor 4.75 x 5.25 x 6.0 in. Electronics 6.0 x 9.8 x 8.5 in.
Power	38 watts
Voltage	28 VDC
Operating Temperature Range	$-35^{\circ}F$ to $+165^{\circ}F$
Installation	Can see earth edge with attitude changes of $\pm 160^{\circ}$ or less. Noral range is 86° per tracker head.
Reliability MTBF	1,330,000 hrs.

Performance Characteristics	
Parameter	**Performance**
Digital output steps	0.01 degree/count
Thermal and mechanical drifts	± 0.01 degree immediately after calibration cycle. Length of time between calibration cycles will depend on environment. At constant temperature, calibration will hold for days. If sensor experiences rapid temperature changes, more frequent calibration will be needed. Pitch and roll outputs are not interrupted during calibration.
Horizon variations (not including earth oblateness)	Varies with altitude and locator selected. Typical worst-case error at n mi is ± 0.06 degree.

Table 3.2-5 (Cont'd)

Parameter	Performance
Electrical noise (with a 1-sec system time constant	Less than 0.002 degrees rms
Altitude and linear tilt range	Sensor will operate without adjustment from 80 to 25,000 n mi. Tilt capability increases from ±10 degrees at 80 n mi to ±20 degrees at 300 n mi and then decreases to ±3 degrees at 25,000 n mi.
Acquisition range	In the coarse tracking mode, the earth can be acquired at pitch and roll angles of ±160 degrees at 80 n mi tapering through ±140 degrees at 400 n mi, to ±13.5 degrees at synchronous altitude.
Time constant*	0.5 to 1 sec (this is not a system limitation; the time constant can be made either slower or faster if desired).

*Time constant is defined as time required for outputs to reach 63 percent of their final value in response to a small step angular tilt.

The key features of the horizon sensor subassembly are summarized below:

1. The sensor is essentially static; no dither, chop, continuous scan, or other periodic mechanical movement isused. A mirror mounted on crossed thermostatic bimetals deflects in response to the bimetal expansion, thereby keeping the sensor field of view (FOV) locked on the earth as the spacecraft attitude or altitude changes. Lack of continuous motion or bearings results in an exceptional level of mechanical reliability combined with no restrictions on sensor operating time in space.

2. All electronic circuits are redundant.

3. Sensor instrument accuracy is 0.06 degree (excluding horizon variations) over a full ± 5 degrees of pitch or roll.

4. The sensor operates in the 14- to 16-micron CO_2 absorption band, thus eliminating most cloud effects. A unique FOV splitting method allows easy implementation of various desired horizon locators to minimize errors stemming from Earth radiance effects.

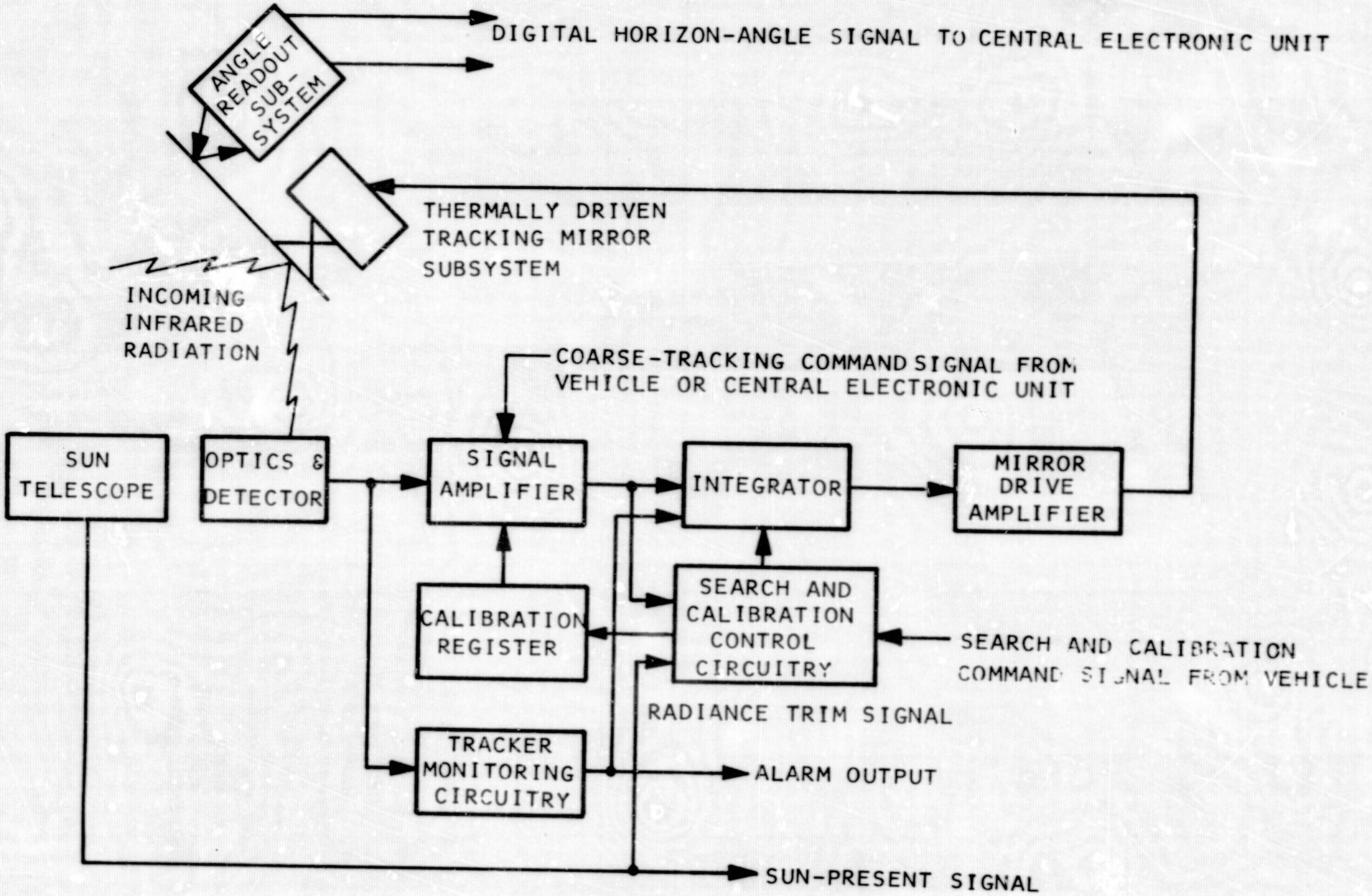


Figure 3.2-3 Single Horizon Tracker Functional Block Diagram

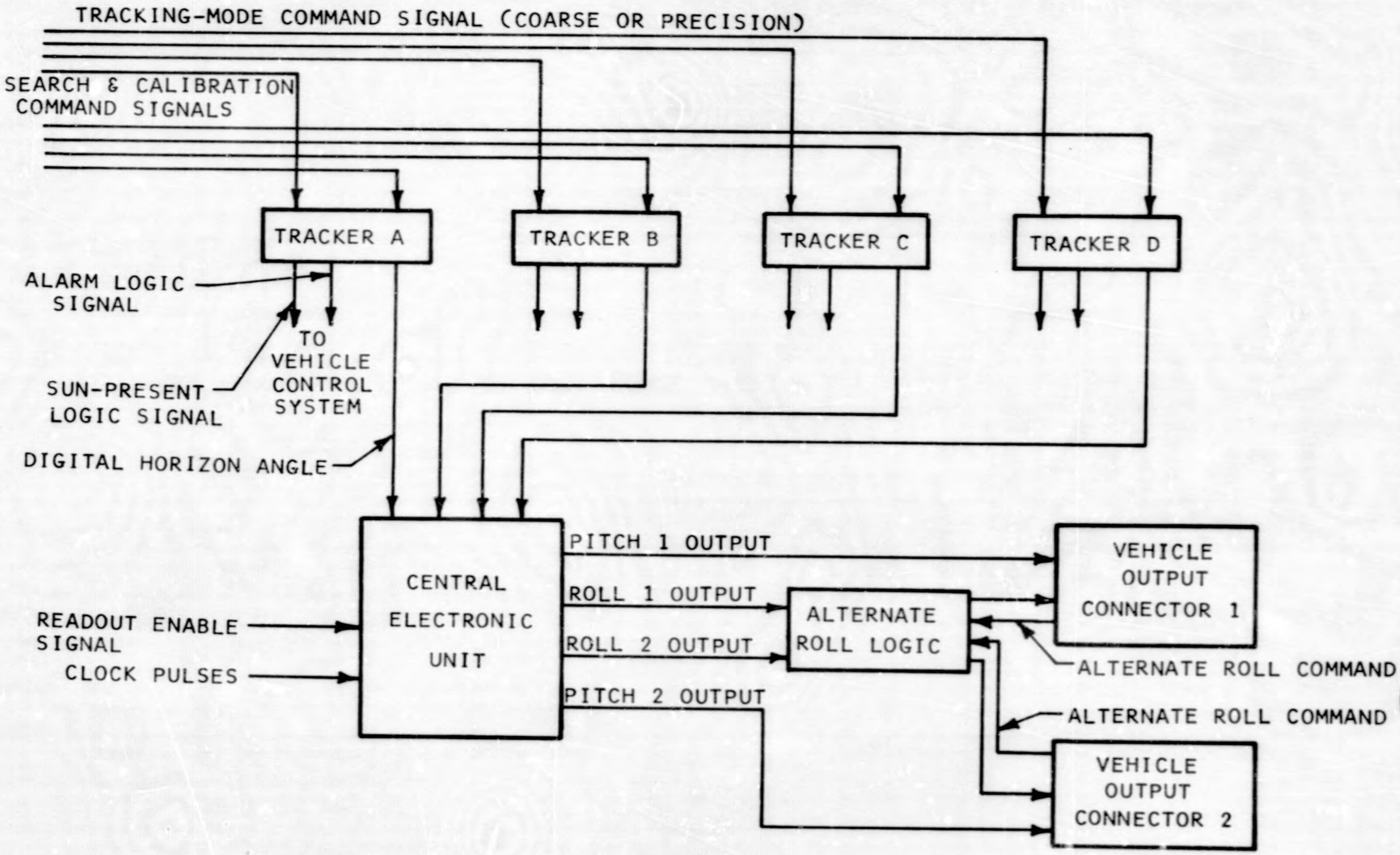


Figure 3.2-4 Horizon Tracker System Functional Block Diagram

5. The sensor is inherently unaffected by variations in horizon radiance levels.

6. A wide-angle acquisition capability is automatically provided, which allows initial Earth acquisition or recovery from accidental extreme attitudes up to ± 160 degrees at lower altitudes.

7. Serial digital outputs are provided. Parallel digital output of the direct-digital angular read-outs of the sensors is available if desired. Analog outputs can also be provided.

The use of four sensors, tracking at 45, 135, 225, and 315 degrees in azimuth with respect to the orbital path, provides redundancy, since any three will permit computation of pitch and roll attitude errors. Continued operation is also provided for those infrequent, short periods when the sun is in the field of view of one sensor.

The electronics unit is redundantly designed so that no single failure will affect performance. A separate connector is supplied to each sensor, and redundant input/output connectors and a separate test/checkout connector is provided.

A rapid self-calibrating operation is provided, controlled by command from the optical-processor, to correct for any spurious (infra-red) thermal changes or electronic drift. During this calibration, all sensors are driven so as to view cold space, and any unbalance is nulled by appropriate operation of a digital bias register. The existing pitch/roll signals are held during this calibration so that transients do not enter the G&C system.

Component Selection Rationale

The requirement to operate from low earth altitudes (100 n mi) to synchronous altitudes with an accuracy of $\pm 0.1^{\circ}$ limited the equipment candidates. The selection of the Quantic EDT 312A, Mod. IVA horizon tracker was based upon the following considerations:

1. Exceeds accuracy requirement of $\pm 0.10^{\circ}$ instrument error ($\pm 0.06^{\circ}$).

2. Provides 100 to 20,000 n mi altitude range.

3. No rapidly moving parts (dithering) are required which improves reliability.

4. Development status - design has been space qualified.

Inertial Measurement Unit

Baseline Selection: Modified Hamilton Standard IMU Martin-Marietta P/N 9600016-010. Table 3.2-6 summarizes the physical and performance characteristics of the unit.

Theory of Operation

A strapdown IMU is characterized by gyros and accelerometers that are directly attached to the vehicle structure. The sensors measure linear and angular motion of the vehicle with respect to inertial space and their outputs are expressed in vehicle or "body" coordinates. The IMU outputs are used to determine present vehicle attitude from some prior inertial reference attitude. This is accomplished entirely by computational means. The inertial attitude thus determined is used to resolve the accelerometer outputs into the inertial coordinate frame where they are doubly integrated (along with a gravitational model if applicable) to yeld vehicle position. Linear equations of motion are integrated once to obtain velocity. Figure 3.2-5 is a functional block diagram of the IMU.

Component Selection Rationale

A modified version of an IMU being built by Hamilton Standard for the Martin-Marietta Corp. was selected for the baseline. The modification consists of changing the unit from 4-gyro, 4-accelerometer system to a 3-gyro, 3-accelerometer system. The Martin-Marietta part number for the unit is 9600016-010.

This unit has the lowest weight and lowest power requirements that could presently be found while satisfying customer requirements. It is being developed for the Viking program and will be operational by 1975.

Maintaining the 4-gyro, 4-accelerometer configuration for the baseline system would provide fail-operational capability at a small additional cost in weight and power.

Engine Control Assembly

Baseline Selection: Honeywell, Inc. Repackaged Driver Circuit. Table 3.2-7 summarizes the physical and performance characteristics of the unit.

Theory of Operation

The Engine Control Assembly (ECA) is comprised of 14 engine valve/ ignition driver circuits and 2 main engine gimbal servo amplifiers. The ECA contains the required interface units and power supplies to allow interfacing directly with the data bus and require only 28 VDC power. One-shot amplifiers are utilized to allow the commanding of minimum APS engine impulses either from the control computer or by the ground operator in the remote manual control mode. The minimum impulse capability is considered to be required for docking operations to provide the necessary level of control. Figure 3.2-6 is a functional block diagram of the ECA.

Table 3.2-6. Inertial Measurement Unit Characteristics Summary

Physical Characteristics	
Item	**Characteristics**
Weight	21 lb.
Size	9 x 11 x 6 in.
Power	125 watts peak, 40 watts average (peak watts include heater power for temperature stability at +160°F)
Voltage	28 VDC
Operating Temperature Range	0 to +80°F
Non-operating Temperature Range	Cycled up to +280°F for 380 hrs.
Installation	Must be mounted on same navigation base as the star tracker.
Reliability MTBF	Expect 60,000 hrs.

Performance Characteristics	
Parameter	**Performance**
Gyros	
Torquing Rates	Low - 2.73 arc sec/pulse High - 21.8 arc sec/pulse
Max. Attitude Rates	Low - 12.5°/sec High - 100°/sec
Accelerometers	
Max. Acceleration	X-axis 20 g Cross-axis 5 g
Scale Factors	X-axis 0.0402 ft/sec/pulse Cross-axis 0.0101 ft/sec/pulse
Outputs	32 KHz rate at 1 micro sec pulse width (each pulse is Δv or $\Delta \theta$)

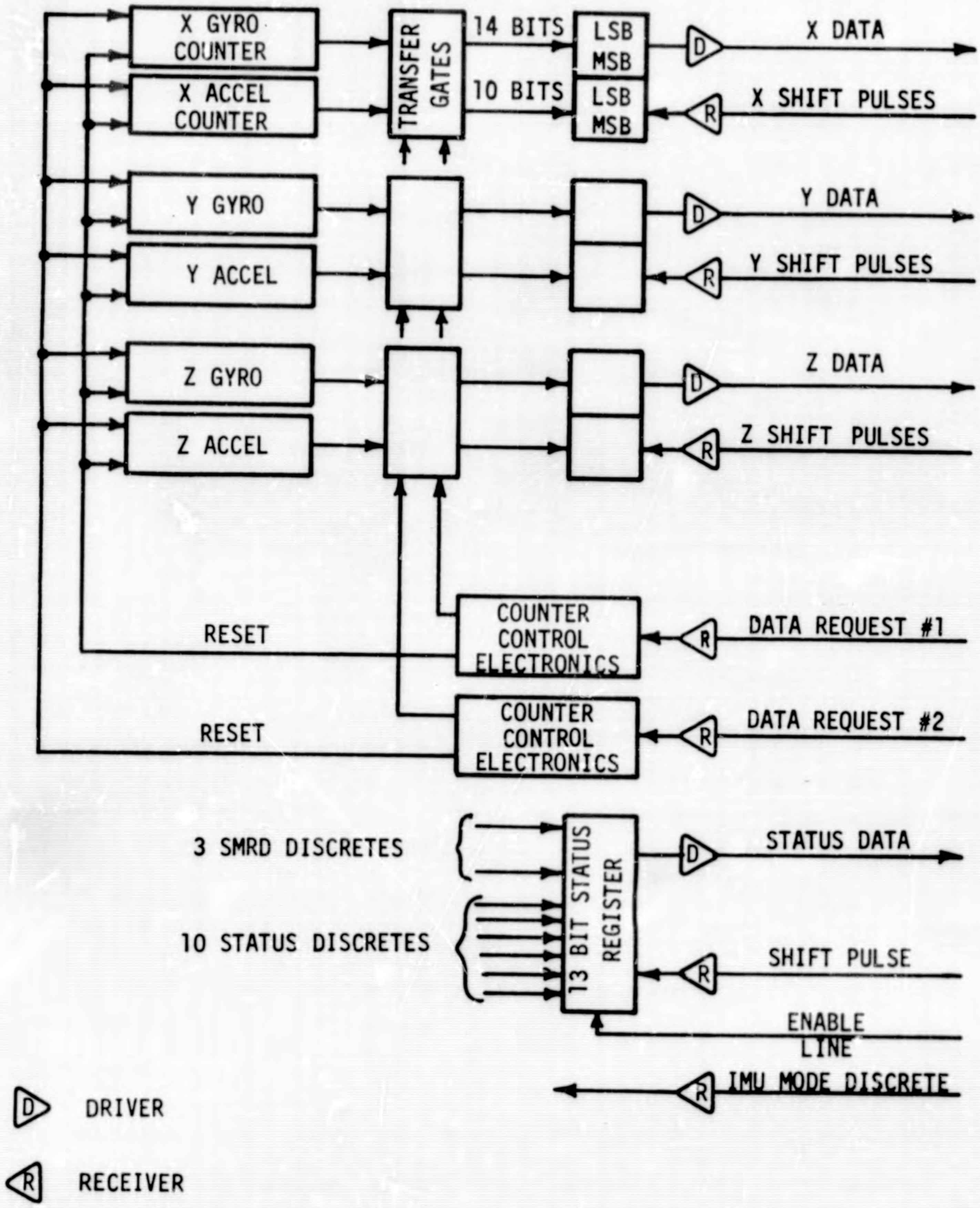


Figure 3.2-5. Inertial Measurement Unit Functional Block Diagram

Table 3.2-7. Engine Control Assembly Characteristics Summary

Physical Characteristics	
Item	Characteristic
Weight	14 lb.
Size	7.6 x 7.6 x 7.6 in.
Power	18.5 watts
Voltage	28 VDC
Operating Temperature Range	+20°F to +200°F
Installation	Prefer aft end of vehicle to be near APS and main engines
Reliability MTBF	10,000 hrs.

Performance Characteristics		
Parameter	Performance	
Inputs	Data Bus	66 cmds, serial digital
	Backup Stabilization	28 discretes
Outputs	Data Bus	64 signals, serial digital

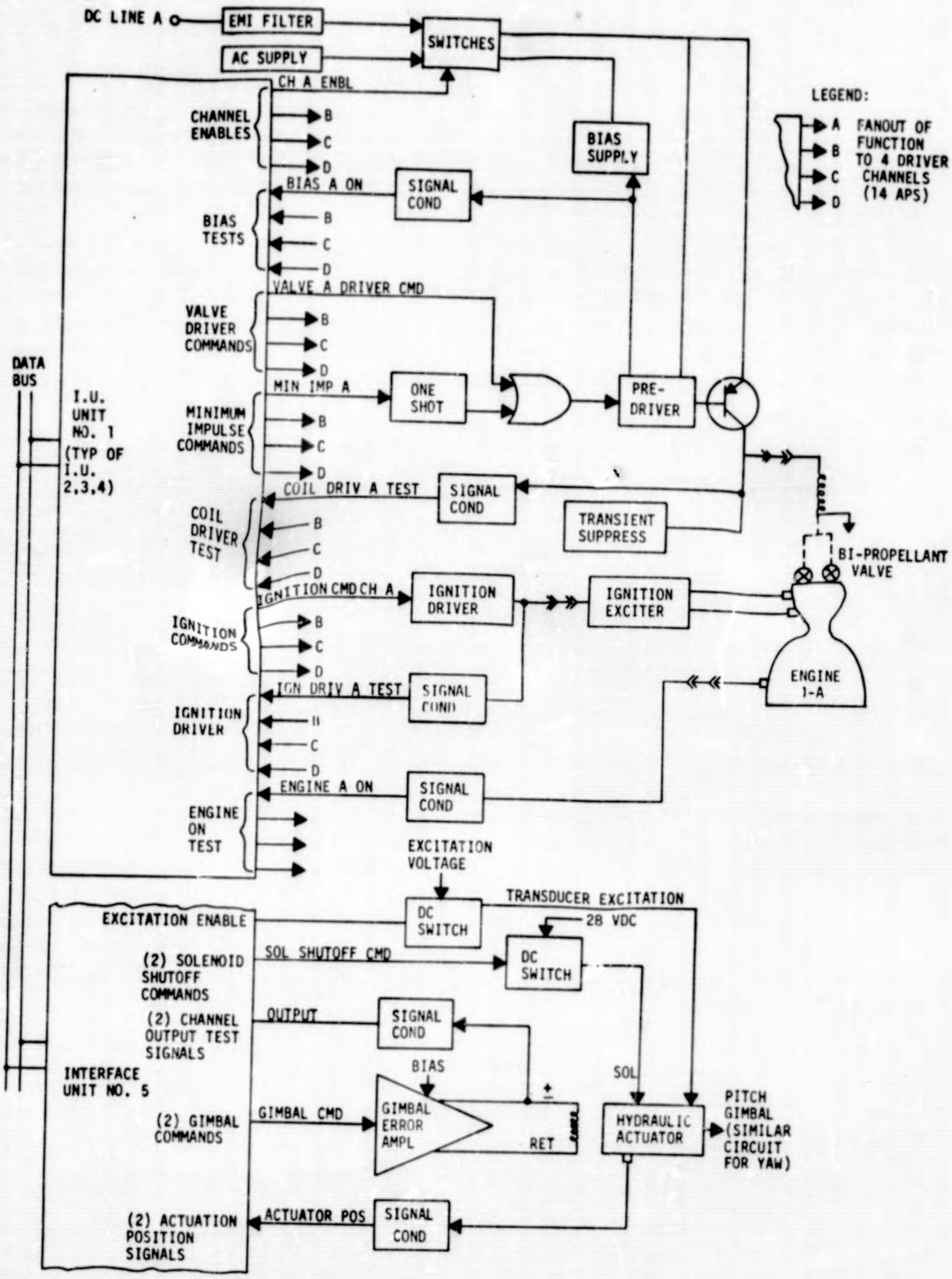


Figure 3.2-6 Engine Control Assembly Functional Block Diagram

Component Selection Rationale

Honeywell, Inc. has driver circuitry that can be repackaged to match Tug requirements and interfaces.

The ECA will necessarily be a custom assembly designed to be compatible with Tug engine arrangements and interfaces. A single, self-contained package containing all interface circuitry required to fire and gimbal engines is considered to be weight effective. The Honeywell design was selected as typical on the basis of known existing circuitry.

Separate packages for various portions of the circuitry were considered to allow location of the appropriate driver circuits close to the engines being controlled. Limitations in the present interface unit design (16 inputs and 16 outputs) and the requirement for multiple internal power supplies when functions are broken up, made a single unit a more desirable approach.

Backup Stabilization Assembly

Baseline Selection: Honeywell Model GG-1500 and Logic Package. TABLE 3.2-8 summarizes the physical and performance characteristics of the unit.

Theory of Operation

The BSA consists of 2 packages, one containing 3 single degree-of-freedom rate gyros with the associated electronics and power supply and the second containing the necessary jet select and fail-safe logic to energize the backup system and deploy the payload on command. These packages would be located in the aft end of the Tug to be near the APS jet drivers and the APS engines.

The fail-safe logic would be designed to bring the back-up system on-line and inhibit the primary system inputs in case of a failure. This would be accomplished by providing an inhibit signal from the computer during normal operations which would be removed in case of failure. The loss of this inhibit signal plus an unsafe vehicle rate measurement would both be required to cause back-up system takeover. Appropriate time delays would be included to prohibit an inadvertant takeover. Uplink override capability would also be provided. A simplified fail safe logic diagram is shown below:

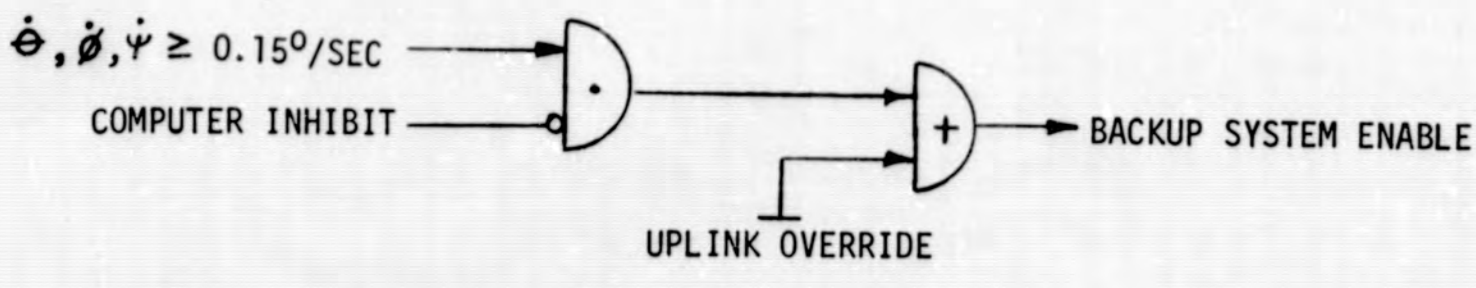
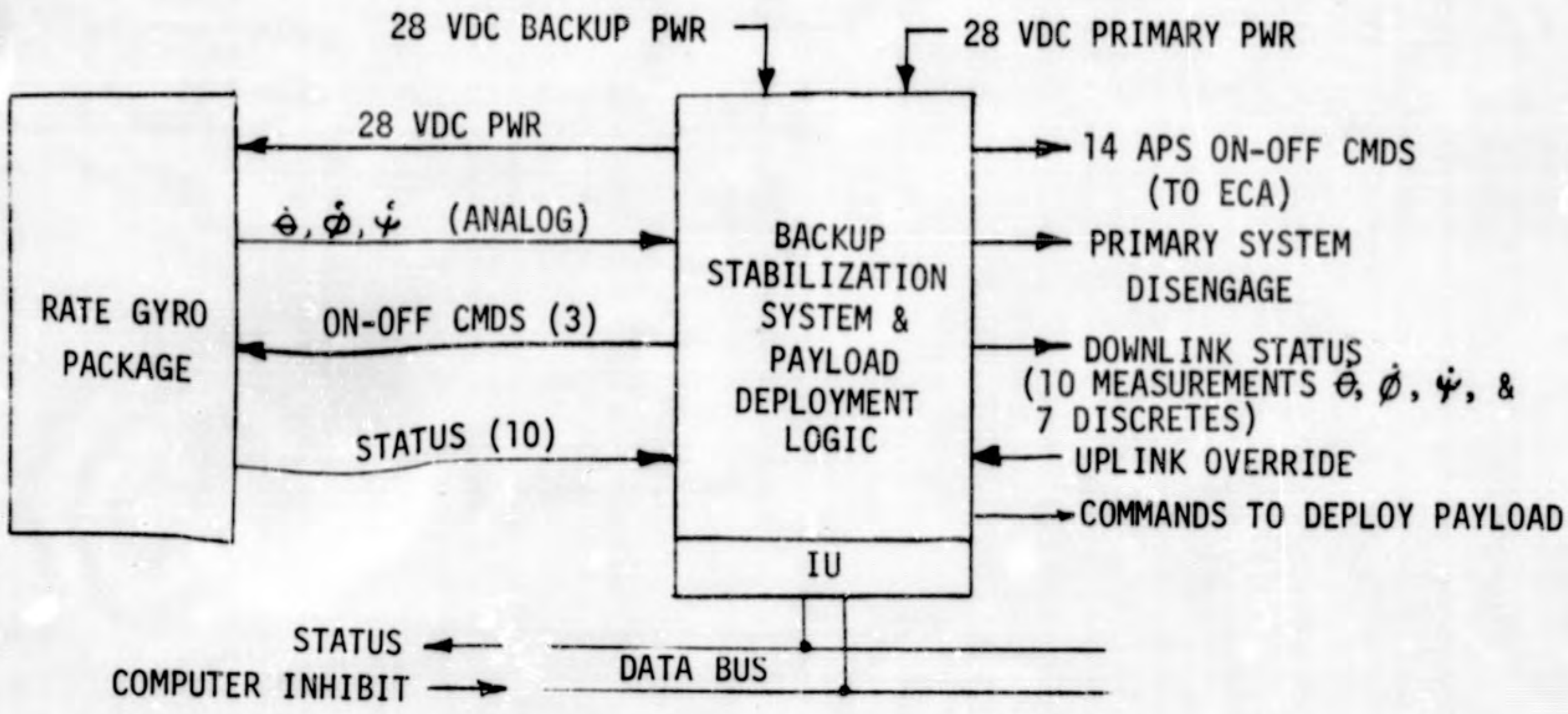


Table 3.2-8. Backup Stabilization Assembly Characteristics Summary

Physical Characteristics	
Item	**Characteristics**
Weight	Gyro Package 3 lb. Logic Package 3 lb. Total Assembly 6 lb.
Size	Cyro Package 4.53 x 3.28 x 3.93 in. Logic Package 3 x 3 x 3 in.
Power	73 watts peak, 17 watts continuous (peak watts includes heater power)
Voltage	28 VDC
Operating Temperature Range	$-65^{\circ}F$ to $+200^{\circ}F$
Installation	Principal axes should be aligned to the corresponding vehicle principle axes
Reliability MTBF	3000 hrs.

Performance Characteristics		
Parameter	**Performance**	
Threshold and Resolution	0.01°/sec.	
Inputs	Data Bus	1 serial digital
	Uplink	3 discretes
Outputs	Data Bus	1 serial digital
	Engine Driver	14 discretes
	Downlink	3 analog, 7 discretes
	Deploy Payload	2 discretes

A block diagram of the Backup Stabilization Assembly is shown below:



Component Selection Rationale

A Honeywell, Inc. Model GG-1500 (contains 3 GG-440 rate gyros and associated electronics) and a jet select and fail safe logic package were selected for the baseline.

The baseline selection is considered typical of equipment available from several vendors and was made to facilitate the point design approach.

Designing fail safe capability into the primary control system by adding redundancy and redundancy management was given consideration but was determined to be much more complex than the baseline and was not considered weight effective.

Autocollimator

Baseline Selection: Modified Barnes Eng. Co. MINEAC II, Table 3.2-9 summarizes the physical and performance characteristics of the unit. More detailed information may be found in References 3.2-6 and 3.2-7.

Theory of Operation

The basic elements of an autocollimator are a light source, optical system, light-sensitive detectors, preamplifiers, and an acquisition detector. Figure 3.2-7 is a functional block diagram of the autocollimator.

Table 3.2-9. Autocollimator Characteristics Summary

Physical Characteristics	
Item	**Characteristics**
Weight	10.9 lb.
Size	4 x 4 x 10 in.
Power	5 watts
Voltage	28 VDC
Operating Temperature Range	-40^{o}F to $+165^{o}$F
Installation	Requires optical path between horizon tracker and star tracker mounting bases.
Reliability MTBF	332,000 hrs.
Performance Characteristics	
Parameter	**Performance**
Linear Range	$\pm$30 arc minutes
Output Scale Factor	200 mv/arc sec.
Resolution (Noise Equivalent Angle)	0.01 arc sec.
Operating Distance	15 ft.
Operating Frequency	1000 Hz
Output Impedance	Less than 100 ohms
Linearity	5 percent
Symmetry	5 percent
Accuracy	1 percent
Cross Coupling	1 percent
Null Quadrature	0.5 arc sec. equivalent
Output	Analog voltage $\pm$ VDC

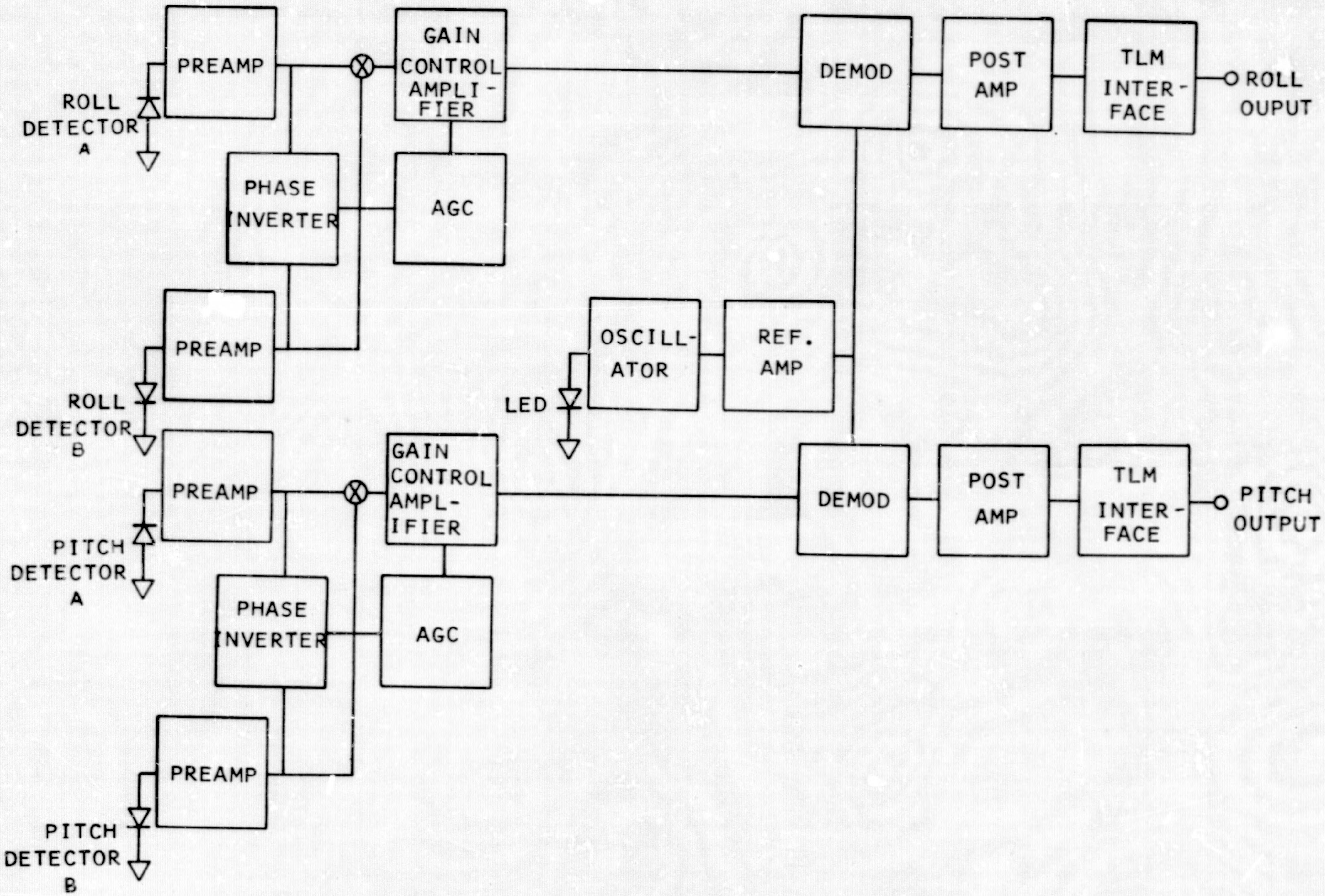


Figure 3.2-7 Autocollimator Functional Block Diagram

The light source projects a modulated, collimated light beam to the reflecting surface on the target. A reticle mark in front of the light source gives the outgoing beam a sharply defined, square cross section. Upon receiving the reflected energy, the autocollimator generates electrical signals which are directly related to the angular deviation between the reflected and the projected light beams. Matched silicon detectors are used in a photovoltaic mode for maximum stability and are not biased.

If the plane of the reflecting surface shifts by as little as 0.01 arc-second, an error signal proportional to the deviation is produced. The signal is linear within ±10 minutes, of arc in the MINEAC II and can be extended in the space designed system. The preamplifiers and detection electronics are designed to provide DC voltage outputs representing alignment shifts.

A modified version of the Barnes Engineering Company Miniature Electronic Autocollimator (MINEAC) was selected for the baseline. Modifications consist of extending the linear range to ±30 minutes of arc, increasing the operating distance to a minimum of 15 ft., altering electrical design to permit operation with 28 VDC input, and reducing package weight.

The boost environment and the extremes of temperature to which the Tug will be subjected are expected to cause significant stresses in the vehicle structure. These stresses are expected to cause misalignments between the IMU/star tracker mounting base and the horizon edge tracker mounting base which would contribute significantly to the errors associated with state vector determination. The autocollimator is designed to measure the changes (in two axes) and utilize the information to correct the angle subtended by the local vertical vector and the star direction vector.

Component Selection Rationale

The MINEAC II space-designed system is more than adequate to measure basic alignment changes. Selection of this particular unit was made to facilitate the point approach. It is reasonable to assume that other vendor's can produce equipment to satisfy the Tug requirements.

3.2.5 Alternate Design Approaches

Several alternate, design approaches were given consideration during this point design study. The most significant of these are discussed in the following paragraphs.

Inertial Measurement Unit Alternatives

Inertial measurement units other than that selected for the baseline were given consideration in the interests of reducing weight. Two promising candidates were the Autonetics MICRON and the Litton Industries NAVAID. The use of either of these units could provide a weight (15 to 18 lbs.) and power (15 to 25 watts) reduction in the GN&C subsystem. The applicability of these units to the Tug is based primarily on their development status.

Autonetics MICRON

The Autonetics MICRON (micro-navigator) was considered as a possible alternative to the strapdown IMU. MICRON is being designed as a lightweight, low cost strapdown guidance system that incorporates LSI electronic technologies. The main constituents of a MICRON are: two small electrostatically-suspended gyro-accelerometers (MESGA's), a miniature 4096-word/24-bit MOS computer, servo electronics, power supply and battery, environmental control and structural housing. The final design will weigh about three pounds, occupy less than 90 cubic inches and consume about 12 watts of power. Input and output data is in the form of 24-bit parallel words.

MICRON is not an operational system but further development is continuing under Air Force funding. Considering the obvious potential of the system, as indicated above, it should be watched closely since continuing progress in its development could produce a truly revolutionary navigator promising random drift rates (0.04°/hr), which would satisfy the Tug's most stringent requirements. Utilization of this system will depend on when it becomes operational and how well it meets design objectives including performance, reliability, and cost.

Litton Inertial NAVAID

Litton Industries has under development an inertial autonavigator employing multiple function inertial instruments to perform the roles of accelerometer and gyroscope in a single device. These instruments are intended for use on a stable platform. The stable platform contains two of these 3 ounce instruments mounted orthogonally to replace the customary pair of two-degree-of freedom gyros and three accelerometers used on platform systems. By reducing the number of instruments and their size, Litton has been able to reduce the platform to a 3.15 inch by 3.8 inch size and 1.9 pound weight (electronics and power supply weight not included). The platform requires 25 watts of power.

The system is being developed primarily for aircraft navigation and missile guidance at the present time but the future may find application in space vehicles if it becomes competitive with strapdown systems. Company engineers say that gyro accuracy could range from 10 degrees per hour to 0.01 degrees per hour. If design goals are met, development of an all-attitude platform system of adequate reliability may find application in space vehicles designed for relatively short mission times.

Star Tracking Alternatives

The present baseline GN&C contains a single gimballed (120° x 120° FOV) star tracker which is consistent with the requirements specified in the study plan. Strapdown star trackers (typified by the ITT, Dual Mode, Aerobee tracker) and star mappers (typlified by the Honeywell mapper used on the Space Precision Attitude Reference System) were given consideration in the interest of reducing inert weight.

The strapdown star trackers employ an electronic scan over a limited field-of-view (8° cone) and eliminate the need for gimballing. In the acquisition mode the entire 8° FOV is scanned. The tracker then enters a

tracking mode in which the brightest object (star) is selected and scanned over a very small FOV (on the order of 16 min). The position of the star is measured in two axes with respect to the boresight of the tracker.

The ITT strapdown star tracker weighs approximately 18 lbs. complete (tracker, control unit, and sun shield/sun sensor/shuttle assembly) which is not considered a significant weight saving over the selected baseline gimballed tracker (25 lbs.). Furthermore, the 8° FOV is a very undesirable limitation for the Tug application requiring the use of two trackers and/or an unacceptable frequency of relatively large vehicle maneuvers in order to take fixes on stars separated by large angles (approx. 90°).

A star mapper of the type used on the Space Precision Attitude Reference System (SPARS) utilizes a lens system to focus a star image on a detector which consists of a symmetrical array of light sensitive lines. If the vehicle is orbiting holding one principal vehicle axis on the local vertical and one principal vehicle axis along the velocity vector (e.g., gyro compass mode), trackable stars will periodically pass through the FOV and will be detected as the star image crosses each line of the detector. The time between line crossings is then used to determine the star position which is used for attitude reference update and star/horizon navigation.

Because of its simplicity, the SPARS type star mapper is a light weight, low power, reliable device, but, since the measurement schedule is dependent upon the random passing of stars through its relatively limited (8°) FOV it isn't considered a desireable approach for the Tug application.

A statistical approach was used to estimate the number of star crossings per orbit rather than attempting to evaluate any particular family of orbits. If the 150 third magnitude and brighter stars were evenly distributed in space, the 8° mapper would sweep 6.98 percent of the celestial sphere and therefore detect 10.5 stars per orbit.

This assumes that the spacecraft would be oriented so that the mapper would sweep a band of the sky which is essentially in the orbital plane. If a yaw bias is applied, the mapper would sweep a band which is out of the orbital plane, and a smaller portion of the sky would be swept per orbit. The table below shows the sky coverage and the average number of star passages, for six values of the out-of-plane angle from 0 to 90 degrees.

Fixed Angle Out Orbital Plane	Portion of Total Sky Covered (Per Orbit)	Average Number of +3 mag. (and Brighter) Star Passages Per Orbit
0°	6.98%	10.5 stars
15°	6.25%	9.4
30°	6.05%	9.1
45°	4.95%	7.4
60°	3.50%	5.2
75°	1.80%	2.7
90°	0.12%	0.18

It should be noted that the sun interference problem would further reduce the number of star fixes available per orbit when using the star mapping approach.

In low earth orbit, where the orbital period is short, the measurement schedule is acceptable (every 8 to 9 minutes) especially if a long time is allowed to reduce the system state errors and a very low drift inertial navigator is used. In synchronous orbit, however, where the orbital period is 24 hours, over two hours would elapse, on the average, between star sightings.

Although no refined performance analysis of the star mapping technique relative to the Tug timeline and attitude profile constraints was performed, it was felt that the resulting measurement schedule is unacceptable and therefore star mapping was discarded as a viable option.

3.3 RENDEZVOUS AND DOCKING

The Rendezvous and Docking Subsystem is composed of components which operate in conjunction with the data management and communications subsystems to accomplish automatic rendezvous and man-controlled docking. The subsystem is designed to be supported by a ground-based pilot and control facility, and by visual and laser aids included on the target vehicle. A functional block diagram of the subsystem is shown in Figure 3.3-1.

A modified version of the "YAG" scanning laser radar being developed by ITT was chosen for the baseline design to perform the function of target acquisition. Self-test, payload rendezvous and docking, and EOS rendezvous with the payload attached to the Tug is accommodated by the addition of a three-position mirror and test reflector to the basic unit. For close range high scan rate capability, the basic unit will be modified to permit the rapid field of view scan necessary for smooth display of data at the pilot console and stable range feedback information.

The television camera used in the subsystem design is a black and white version of the unit developed for the Apollo program to transmit video signals from the lunar surface. The camera is controlled by the DMS and its output is transmitted to the ground station via the Communications Subsystem FM transmitter and antenna system. To provide adequate illumination of the payload mounted visual aids used in docking, a lighting system is included in the design configuration. The subsystem design rigidly mounts the television camera and provides for a fixed field of view. Performance and operational requirements are satisfied by this approach while savings in weight and improvement in reliability are gained.

A Tactical Air Navigation (TACAN) transponder system is included in the subsystem to provide ranging information during EOS/Tug rendezvous and docking.

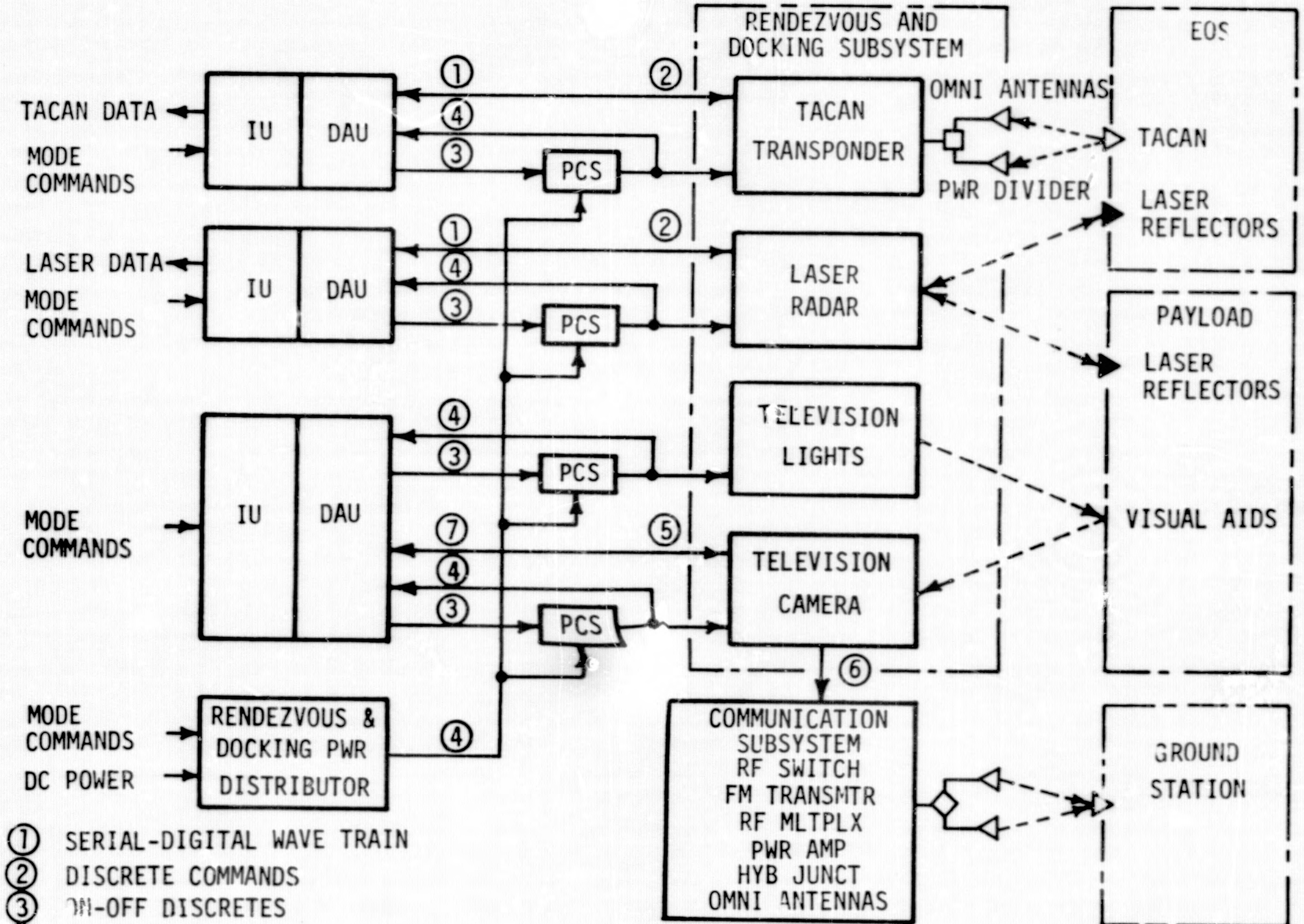
3.3.1 Requirements

Since the basic mission requirement for the Tug is to place and/or retrieve payloads in geosynchronous orbits, and then return to the EOS, it is of paramount importance to perform rendezvous and docking operations with minimum complexity. This emphasis should therefore be reflected in the rendezvous and docking requirements. Requirements and constraints are discussed in the following paragraphs, and are summarized in Table 3.3-1.

Tug/Payload Rendezvous and Docking

Operational Sequence

The Tug is the active vehicle in rendezvous and docking with passive payloads. Rendezvous begins at a range of not more than 100 km, based on the predicted laser radar acquisition distance. The Tug autonomously controls closure to 300 meters, the range at which docking maneuvers begin. Included in docking is a stationkeeping period of undefined length, maintained at the



① SERIAL-DIGITAL WAVE TRAIN
② DISCRETE COMMANDS
③ ON-OFF DISCRETES
④ 28 VDC POWER
⑤ CHECKOUT STIMULI
⑥ ANALOG OUTPUT
⑦ ANALOG RESPONSES TO CHECK STIMULI: PWR SUPPLY, x & y DEFLECTION, PREAMPLIFIER SIGNAL CONDITIONING AND END-TO-END RESPONSE TEST USING LENS-MOUNTED LIGHT-EMITTING DIODE

Figure 3.3-1 Rendezvous and Docking Subsystem Functional Block Diagram

Table 3.3-1. Summary of Rendezvous & Docking Requirements and Constraints

Subsystem Design Requirements	Source
1. Tug active rendezvous and docking in synchronous orbit	NASA Guidelines
2. Maximum rendezvous range is 100 km	NASA Guidelines
3. The rendezvous sensor is laser radar	NASA Guidelines
4. Maximum docking range is 300 meters	NASA Guidelines
5. Remote pilot with television for payload docking	NASA Guidelines
6. Nominal rendezvous orbit injection delta-V's are 47 ft/sec outbound, 3000 lb payload mission 5893 ft/sec outbound, 4160 lb payload mission 296 ft/sec inbound, all missions	NR design analysis
7. Docking accuracy requirements (see Table 3.3-2)	NASA Guidelines
8. Tug is active during Tug recovery rendezvous	NASA Guideline
9. EOS is active during Tug recovery docking	NASA Guidelines
10. The payload is designed for passive cooperative docking	NASA Guidelines
11. The laser rendezvous accuracy is 0.1 meter range and 0.1 degree at 100 km	NASA Guidelines
12. Television quality is 250 lines, 15 frames/sec	NASA Guidelines
13. Autonomous injection accuracies are: (3 sigma) <10 km, 5 meters/sec in low earth orbit <50 km, 5 meters/sec in synchronous orbit	NASA Guidelines

Table 3.3-1. (Cont'd)

Subsystem Design Requirements	Source
14. The laser must be located on the forward end of the Tug with an unobstruced 30 x 30-degree field of view parallel to the vehicle x-axis to accommodate close rendezvous and docking with the payload. The laser must also have a 30 x 30-degree unobstructed field of view approximately orthogonal to the x-axis to accommodate close rendezvous with the EOS with a payload attached. It is also advantageous to mount the laser either on the star tracker/IMU navigation base or the horizon tracker base in order to minimize misalignment between the laser and IMU.	NR design analysis
15. The television camera must be located on the forward end of the Tug with an unobstructed 52.6 degree field of view parallel to the Tug x-axis to facilitate man-in-the-loop docking with the payload. It should also be located as close to the Tug centerline as possible and opposite to the laser to give maximum geometrical advantage to the combined visual and laser data. Normal structural alignment of the television camera should be sufficient.	NR design analysis
16. The payload docking lights must illuminate the visual docking aids such that sufficient and unambiguous cues are given to the remote pilot.	NR design analysis

300 meter range. Docking closure and contact with the payload is accomplished by a pilot on the ground using onboard television and laser radar downlink data. After hard docking is verified, the Tug/payload autonomously returns to the EOS.

Design and Performance Criteria

Circularization in the payload orbit involves two delta-V regimes: (1) injection from a 100 n mi by synchronous transfer orbit at synchronous altitude using the main propulsion system is nominally 5893 ft/sec: (2) injection from a phasing orbit to return to synchronous altitude after deploying a payload may nominally require only 47 ft/sec from the APS. In any case, injection accuracy should place the Tug to within 50 KM and 5 m/sec (3 sigma) of the payload.

Table 3.3-2. Docking Accuracy Requirements

Parameter	Structural	G & C
Centerline Miss Distance, ft	0 to 1.0	0 to 0.75
Miss Angle, deg	0 to 5.0	0 to 1.0
Longitudinal Velocity, ft/sec	0.1 to 1.0	0 to 1.0
Lateral Velocity, ft/sec	0 to 0.30	0 to 0.30
Angular Velocity (Combined maximum of pitch, yaw and roll motion), deg/sec	0 to 0.50	0 to 0.50

To initiate rendezvous, and periodically during the rendezvous phase, the Tug measures relative target range with the laser radar to at least ±0.1 meters, and line - of - sight angle to at least 0.1 deg. To attain the angular accuracy, the predicted laser characteristics must be considered. These include a line-of-sight angular rate of ≤0.025 deg/sec, which includes both the target motion across the laser field of view and the Tug drift rate across its own attitude deadband.

All maneuvers performed during the stationkeeping and docking phases are commanded by the remote pilot using television and laser data. Television requirements are 250 lines and 15 frames/sec. Docking accuracy requirements are specified in Table 3.3-2.

In order to provide wide mission flexibility, the Tug should be capable of acquiring an arbitrarily oriented target. Passive laser reflectors will therefore be mounted so that adequate reflected beam power is obtained from any angle. Additional reflectors, to be used during docking approach, are located near the target docking port to permit sequential laser measurements necessary to determine three-axis angular alignment errors.

Tug/EOS Rendezvous and Docking

Operational Sequence

The Tug is the active vehicle when rendezvousing with the EOS, and carries out autonomous operations similar to those of Tug/payload rendezvous. At 300 meters the Tug assumes a passive role for docking, but is subject to remote commands from the EOS. The EOS performs the active docking maneuvers.

Design and Performance Criteria

EOS rendezvous orbit circularization by the Tug is nominally 296 ft/sec. Injection accuracy is within 10 KM and 5 m/sec (3 sigma). EOS rendezvous operations are similar to those of payload rendezvous. Upon the completion of rendezvous, the Tug stands ready to receive commands from the EOS to begin

systems power-down and docking operations. Docking accuracy requirements are the same as those for payload docking.

3.3.2 Subsystem Trades

Numerous trades were conducted at the subsystem level to determine approaches which influenced the selection of components. Only two of these trades are of sufficient importance to deserve separate discussion. Both gimbal and zoom television capability were requirements at the start of the study. The topics are discussed in this section.

Gimballed Television Camera Capability

Laser radar is used to rendezvous the Tug to a station keeping position within 300 meters of the target vehicle. Television coverage is considered required only when moving the Tug from this 300 meter station keeping position to the docked position. Since both the laser radar and television are normally boresighted along the Tug X-axis (docking axis) and since the laser radar field-of-view is 30° maximum, a television camera with a 52.6° field-of-view will always aquire the target automatically if it is being tracked by the laser radar.

Also, the Tug baseline Communications Subsystem provides omni-directional antenna coverage, thus eliminating any possible conflict between pointing the television camera (synonomous with pointing the vehicle) and antenna pointing requirements.

The baseline non-gimballed television camera restricts the vehicle attitude profile to a 52.6° cone if the target is always to be kept in the field-of-view. This restriction could be somewhat undesirable, (but not unacceptable) when circling the target to align with the docking port. A gimballed camera, it should be noted, would also be restricted to covering something less than a hemisphere due to installation constrainsts. Furthermore, large attitude excursions would necessarily cause the laser radar to lose track which is considered highly undesirable if not unacceptable during this phase of docking.

NR docking simulations performed in support of the Apollo program and the RMU/ATS-V rendezvous and docking studies indicate that a gimballed camera approach could possibly create serious human factors problems in presenting the visual data to the ground pilot (i.e. the pilot loses the "feel" of target location with respect to his normal controls). This is an intuitive extrapolation of past experience, however, and can only be finally determined through sophisticated man-in-the-loop simulations. Such simulations are recommended for follow-on Tug study phases.

It is therefore felt that a non-gimballed camera approach will accomodate the docking phase requirements, and that the additional search capability provided by gimballing would only provide minimal improvement. For these reasons, the additional complexity, failure modes, and weight were considered the overriding factors and a fixed camera was baselined.

Television Zoom Capability

Consideration was given to both zoom lens and to a movable wide angle lens which could be rotated in or out of position in front of a narrower field of view lens to give two distinct coverage capabilities. It was determined through analysis that a single, fixed, wide-angle lens could provide adequate coverage from the 300 meter stationkeeping distance to docking contact. This includes adequate resolution to recognize the target and its attitude at maximum range.

It was felt, therefore, that longer range capability would only provide capability to inspect the target from a distance which was not considered a requirement.

Based upon the adequacy of a fixed lens system, the additional complexity, failure modes, weight and electrical power associated with a zoom capability was not considered warranted.

A more detailed discussion of the analysis leading to this tension is contained in the television camera component selection rationale.

3.3.3 Operations Analysis

Rendezvous Operational Modes

At least two alternate rendezvous operational modes are feasible for autonomous navigation to television acquisition range. These modes, and the constraints imposed by Tug and payload position uncertainties, have been analyzed for the three Tug missions. The logical relationship between modes is presented in Figure 3.3-2.

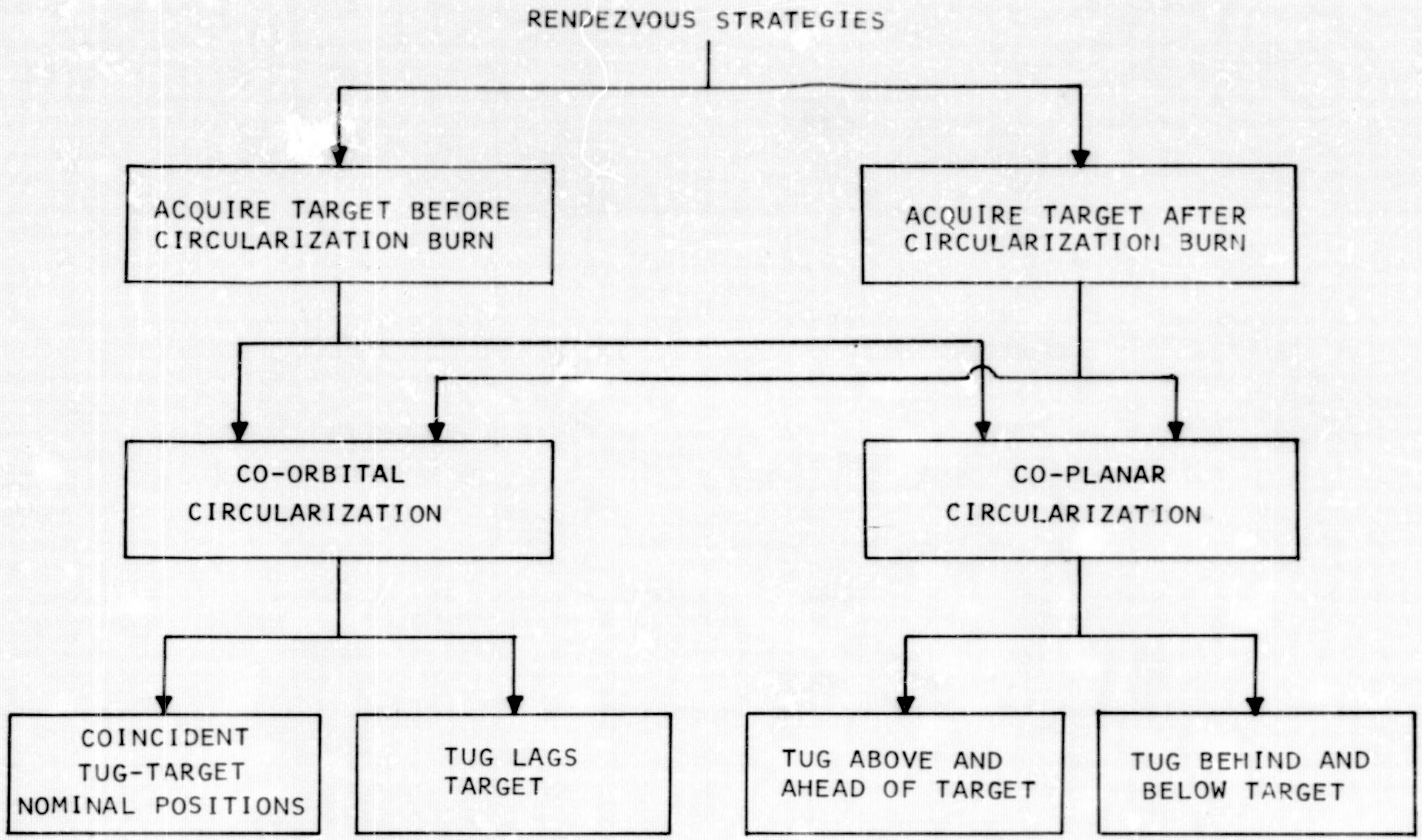


Figure 3.3-2 Rendezvous Strategies

Target Location After Tug Circularization

One candidate method is to locate the target after the Tug has circularized in the rendezvous orbit. Relative position uncertainties are large in this case, and include propagated errors from the transfer orbit. If the Tug circularizes at the target nominal position, the target could be located anywhere with respect to the Tug, within the position uncertainty sphere. Acquisition of the target by either television or laser radar in this case requires a complicated spherical search maneuver. The search maneuver is simplified, and may not involve vehicle rotation, if the Tug circularizes at a distance from the target which is greater than the total radius of position uncertainty.

A refinement of this strategy is to circularize above and ahead or below and behind the nominal target position so that closure rates are assured. Unfortunately, unless the Tug velocity uncertainty is very much less than the 5 meters/sec which has been specified, the difference in orbital altitudes needed to insure closure rates is much larger than the 100 km laser range.

Circularization at the target altitude, but lagging the target by the position uncertainty radius, therefore seems to be the best approach with this mode. A trial laser acquisition attempt is performed immediately after the circularization propulsive period is completed. The geometry of the acquisition problem is shown in Figure 3.3-3. If the target is located, closure maneuvers can then begin. If the target is not located, autonomous navigation update operations are initiated to refine onboard position and velocity knowledge. While the update operations are being performed, the Tug is moving relative to the target at a maximum rate of only 7 meters/sec, which appears to allow sufficient time for the navigation update operations to be completed before the target moves out of laser range. The navigation refinements will lead to more accurate knowledge of target relative position, and a subsequent new acquisition attempt is virtually assured of success. In some cases small APS delta-V or attitude maneuvers will be necessary to move the Tug within laser range or the target within field of view.

Target Location Prior to Tug Circularization

The second candidate method of target location involves laser acquisition before Tug circularization. Data on target relative position can be used to refine the circularization propulsion commands. This information is particularly useful since the target position, obtained from long periods of ground tracking, is known more accurately than that of the Tug. The Tug will circularize behind the target, as in the previously discussed case, but will be nearer due to the smaller position uncertainty radius. Laser acquisition of the target will probably be lost during the propulsion operation, however the Tug should reacquire after circularization much more easily than in the previous case.

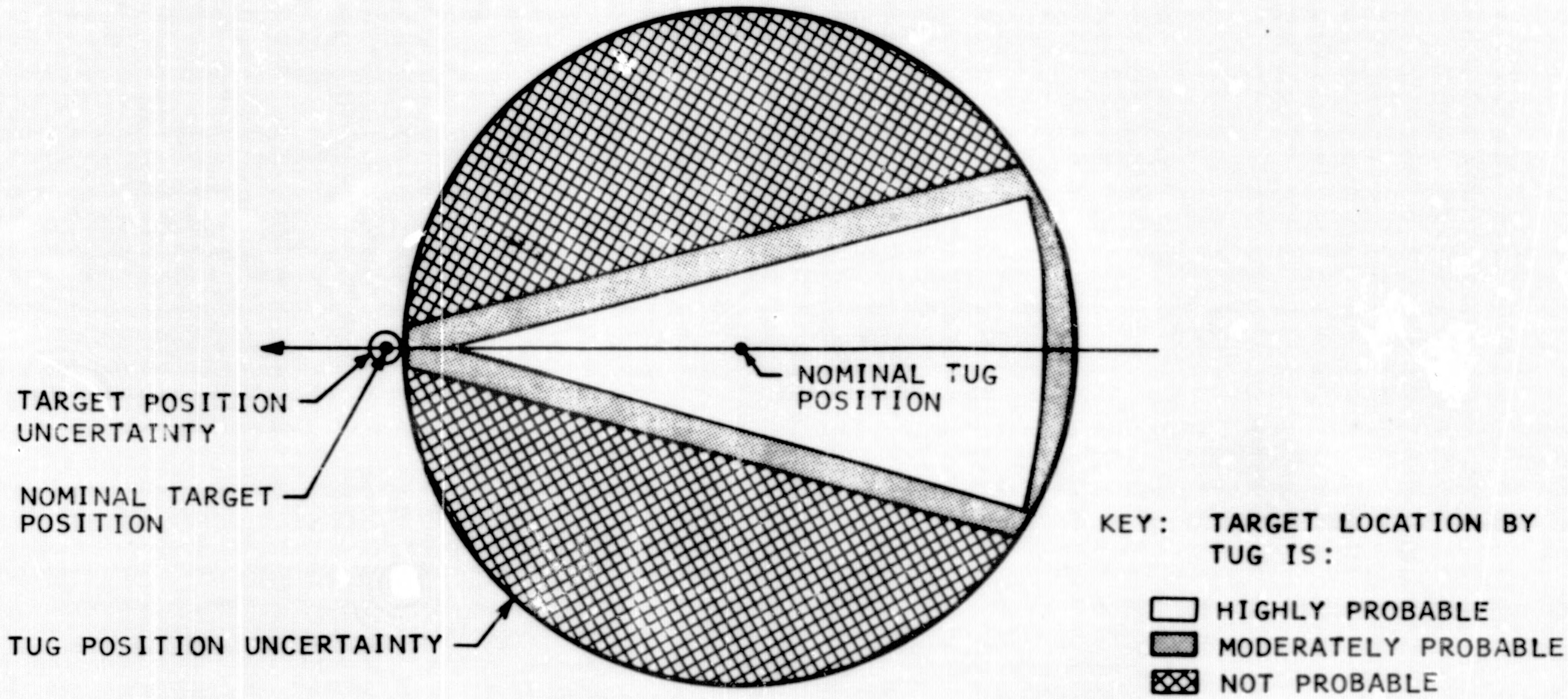


Figure 3.3-3 Relationship Between Position Uncertainty and Target Location Probability

Selected Rendezvous Operational Modes

From the previous discussion it is apparent that laser acquisition prior to the circularization propulsion operation is desirable. It remains to apply the method to individual mission parameters to prove feasibility. The sequence of events aboard the Tug is to calculate the nominal line of sight to the target, rotate the Tug so that the laser views that portion of space, acquire the target, calculate the propulsion guidance command for circularization, rotate the Tug to the initial propulsion orientation, circularize, then reacquire the target. The Tug should detect the target straight ahead, at a range of 300 meters plus the position uncertainty radius.

Estimates of vehicle parameters for Tug/payload and Tug/EOS rendezvous operations were used in a digital computer program to determine the feasibility of the approach. The vehicle parameters are summarized in Table 3.3-3. It should be noted that part of the data in the table, particularly delta-V values, represent preliminary data and are not consistent with the final values given in Table 2.3-1, Vol. II of this report. As will be shown, the preliminary data yields conservative answers in the analysis. A set of linearized equations taken from Reference 3.3-1 and other sources provided the nucleus of the computer program. An analysis of the results is presented in Figure 3.3-4. Each small circle on the graph represents the thrust period needed to generate the indicated circularization delta-V. Variations in delta-V move the circles up or down on the dashed lines. The solid lines which move upward and to the right indicate range versus time prior to circularization. Although only

Table 3.3-3. Rendezvous Simulation Input Data

	Payload Mission		
	3 KLb	4.16 Klb	8.06 Klb
Nominal Weight, Lb			
Payload Rendezvous	22500	25000	-
EOS Rendezvous	8750	10198	5700
Nominal Delta-V, Ft/Sec			
Payload Rendezvous	47	5893	-
EOS Rendezvous	296	296	296
Thrust Level, Lbs			
Payload Rendezvous	140	10000	-
EOS Rendezvous	10000	10000	10000
Specific Impulse, Sec			
Payload Rendezvous	380	470	-
EOS Rendezvous	470	470	470

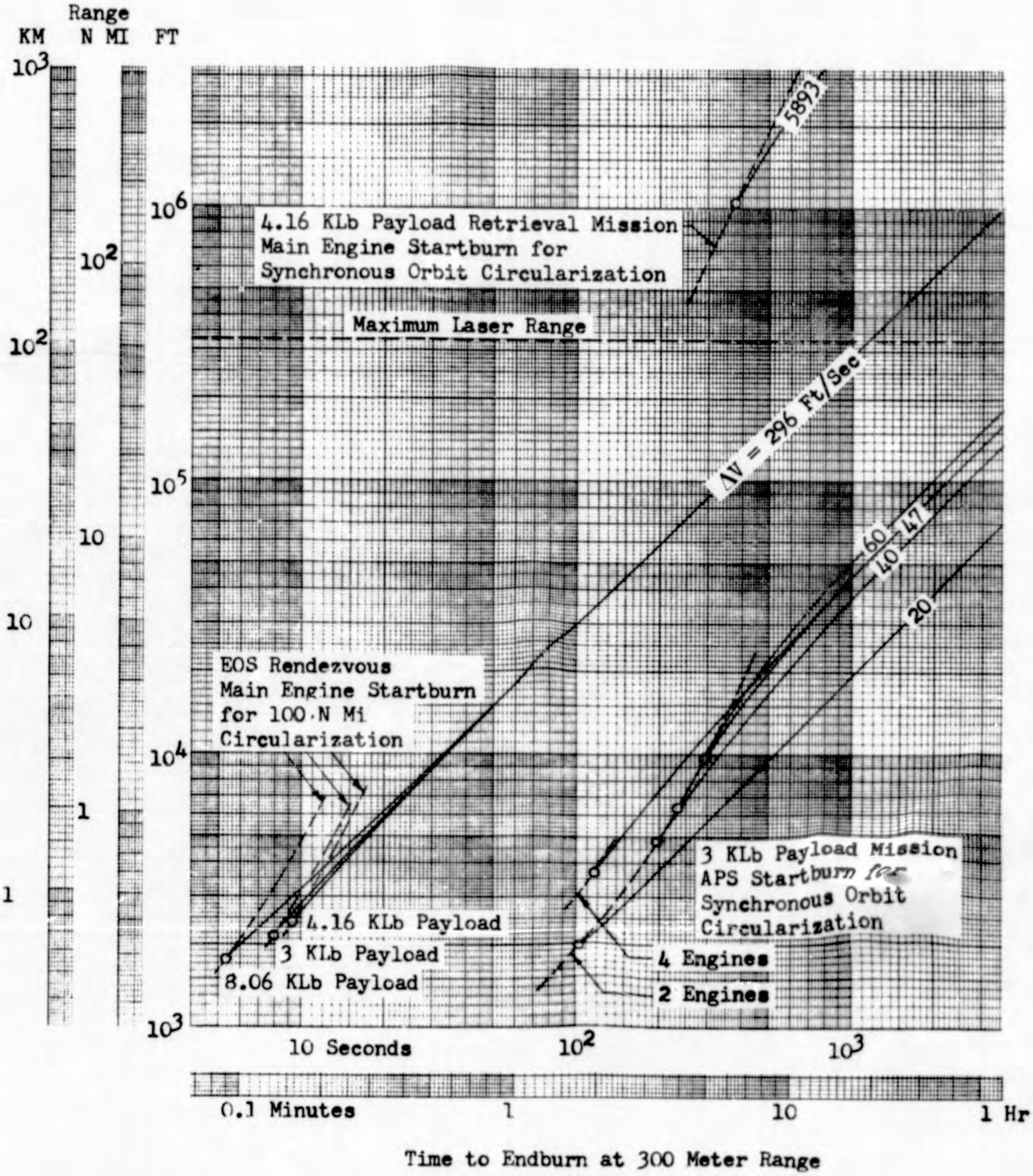


Figure 3.3-4 Analysis of Pre-Circularization Burn Time and Range

nominal conditions are shown on the graph, it appears that adequate time for target location is available in all cases except that of synchronous orbit insertion of the 4.16 Klb payload retrieval case. Final delta-V data (Table 2.3-1, Vol. II) are slightly lower than the values used in the graph, which served to increase the available acquisition time.

The recommended methods of rendezvous are summarized in Table 3.3-4.

Payload Docking Operational Modes

Docking operations begin when the Tug closes to 300 meters from the target in synchronous orbit. Before the docking approach is initiated a number of operations are performed. These are conducted in a stationkeeping (constant range) mode. The target will be acquired by television, if this event has not already taken place during rendezvous. Remote manual control and communication links will be checked out. The target will be given a preliminary visual inspection.

Initial docking port alignment is conducted by circling at a constant range, which is not necessarily the stationkeeping range. Details of this maneuver are discussed next.

Initial Docking Port Alignment

The Tug may first sight the target from any side, or either end; and before final docking, must circle the target to align docking ports. This maneuver could be avoided if the target responds to a commanded attitude appropriate for alignment. However, even if this feature is part of the payload design, circling capability in the Tug is desirable for at least two reasons: the payload may have a failure condition which precludes attitude command response; and, the visual inspection by television of the payload which is provided by circling is a means of detecting and assessing an exterior failure. Thus, the ability to encircle the target prior to final docking is a needed and useful capability.

The circling manuever is not easily accomplished by a remote pilot using television and laser data, because it involves simultaneous control and coordination of all flight variables, many through large ranges. As will be shown, propellant usage in such a maneuver can quickly become appreciable. The alternative is to automate the circling maneuver at the command of the remote pilot. He establishes the direction of rotation, circling range, and has control to start and stop the maneuver. The automatic mode would be used when the docking ports are misaligned by more than approximately 25 degrees.

The simplest circling maneuver is conducted at constant range, with the Tug centerline aimed at the target geometrical center. Lateral APS engines provide the initial and final angular acceleration, while forward-thrusting axial APS engines produce normal acceleration to maintain a constant range. Five types of control functions are performed by the APS while the Tug is circling: the first two are to provide the angular and normal acceleration

Table 3.3-4. Summary of Recommended Rendezvous Methods

Payload Mission	Payload Rendezvous	EOS Rendezvous
3 KLb	Preburn Acquisition	Preburn Acquisition
4.16 KLb	Postburn Acquisition	Preburn Acquisition
8.06 KLb	-	Preburn Acquisition

Definitions: Preburn Acquisition -- The Tug locates the target by laser radar before circularization. Laser data is used to reduce position and velocity uncertainties relative to the target, and modifies the propulsion guidance command to place the Tug nearer to the target. Circularization is nominally in the target orbit, and lags by 300 meters plus the position uncertainty radius. Reacquisition of the payload occurs after circularization with a high probability of success.

Postburn Acquisition -- The Tug circularizes without locating the target. Circularization is nominally in the target orbit, and lags by the position uncertainty radius. Closure to 300 meters proceeds if the target is located by the initial laser scan. Autonomous navigation proceeds if the target is not located, resulting in a new predicted target position. Small delta-V maneuvers may be necessary to close range within 100 km.

already mentioned. Lateral thrusting produces a pitch or yaw disturbance torque which must be controlled by a counter torque. Rotational control of the Tug line-of-sight for television and laser tracking is maintained. Finally, the APS provides three-axis attitude and attitude rate stabilization during the entire operation. Approximate equations were derived for computing the propellant usage by each function. These, and the appropriate data and nomenclature, are summarized in Table 3.3-5.

Consideration of the problem reveals that propellant usage varies proportionately with both the circling angular rate and the range. As the circling rate is increased, a maximum is reached, where continuous axial thrust just maintains a constant range; higher rates cause divergent range. This relationship is shown parametrically in Figure 3.3-5. There does not appear to be a reason for using high circling rates, since a 0.1 deg/sec rate consumes only 30 minutes for a complete half-circle of revolution. Large quantities of propellant can be saved, not only by circling slowly, but also by circling at close range. A value for the safe range depends on the specific target and mission; however, 100 feet was taken as a nominal value for this study.

Table 3.3-5. Circling Maneuver Propellant Computation

Symbol	Eqn or Value	Description and Units
W_{P_L}	$\frac{2\ R\ W\ \dot{\beta}}{57.3\ g\ I_{sp}}$	Lateral velocity control propellant -- Starting and stopping circling motion, lb
W_{P_D}	$\frac{2\ n\ F \ell\ R\ W\ \dot{\beta}}{57.3\ n_{\ell} F_{\ell} r\ g\ I_{sp}}$	Lateral-to-yaw disturbance propellant -- Consumed during starting and stopping circling motion, lb
W_{P_N}	$\frac{W\ R \dot{\beta} \beta}{(57.3)^2\ g\ I_{sp}}$	Range control propellant -- Provides normal force, lb (Neglects startup and shutdown periods)
W_{P_A}	$\frac{2\ I_p \dot{\beta}}{57.3\ r\ I_{sp}}$	Angular rotation control propellant -- Provides rotating line of sight, lb
W_{P_S}	$\frac{57.3\ \Delta t_{min}{}^2 \beta r}{4\ I_{sp} \theta_{db} \dot{\beta}} \quad \frac{(n\ F_r)^2}{I_r} \quad \frac{2(n\ F)^2}{I_p}$	Attitude stabilization propellant used during circling period, lb
F	70	Single axial APS engine thrust, lb
F_{ℓ}	20, 70	Single lateral APS engine thrust, lb
F_r	20	Single roll APS engine thrust, lb
g	32.174	Gravitation acceleration, ft/sec^2
I_r	5180	Roll moment of inertia, slug-ft^2
I_p	42000	Pitch moment of inertia, slug-ft^2
I_{sp}	380	APS specific impulse, sec
ℓ	4	Length from APS station to total cg, ft
n	2	Number of engines

Table 3.3-5. Circling Maneuver Propellant Computation (Cont'd)

Symbol	Eqn or Value	Description and Units
n_ℓ	2, 1	Number of lateral engines
R	≤990	Range, ft
r	7	APS moment arm, ft
Δt_{min}	0.025	APS minimum pulse duration, sec
W	21000	Vehicle weight, lb
θdb	1	Attitude deadband peak amplitude, deg
β	Variable	Circling angle, deg
$\dot{\beta}$	Variable	Circling rate, deg/sec
$\dot{\beta}_{max}$	$57.3(\frac{n\ f\ g}{R\ W})^{1/2}$	Maximum circling rate, deg/sec

Propellant usage does not vary strongly with total circling angle at 100 feet, as is shown by the graph of Figure 3.3-6. At 0.1 deg/sec, increasing the circling angle from 25 to 180 degrees costs approximately 1.1 lb.

Figure 3.3-7 gives a breakdown of propellant usage by function. The major amount is spent controlling range and the torque disturbance caused while beginning and terminating the circling maneuver.

To implement the semiautomatic circling maneuver, it can be programmed as a separate on-board guidance subroutine which develops the appropriate autopilot command. Range control feedback information is derived in this case from the laser radar data. The angular distance is controlled by the pilot, rather than with feedback information. Since rendezvous is terminated at 300 meters, the pilot will control closure to the circling range (approximately 100 feet). Final docking alignment will begin at the circling range, after completion of the maneuver.

Final Docking Approach

Closure and error reduction during final docking is accomplished by commands from the remote pilot. The block diagram of the system configuration during active docking is shown in Figure 3.3-8. The mechanization can be described as a rate command system with attitude hold, which is similar in attitude control to that of the Apollo manned controls. Deflection of the

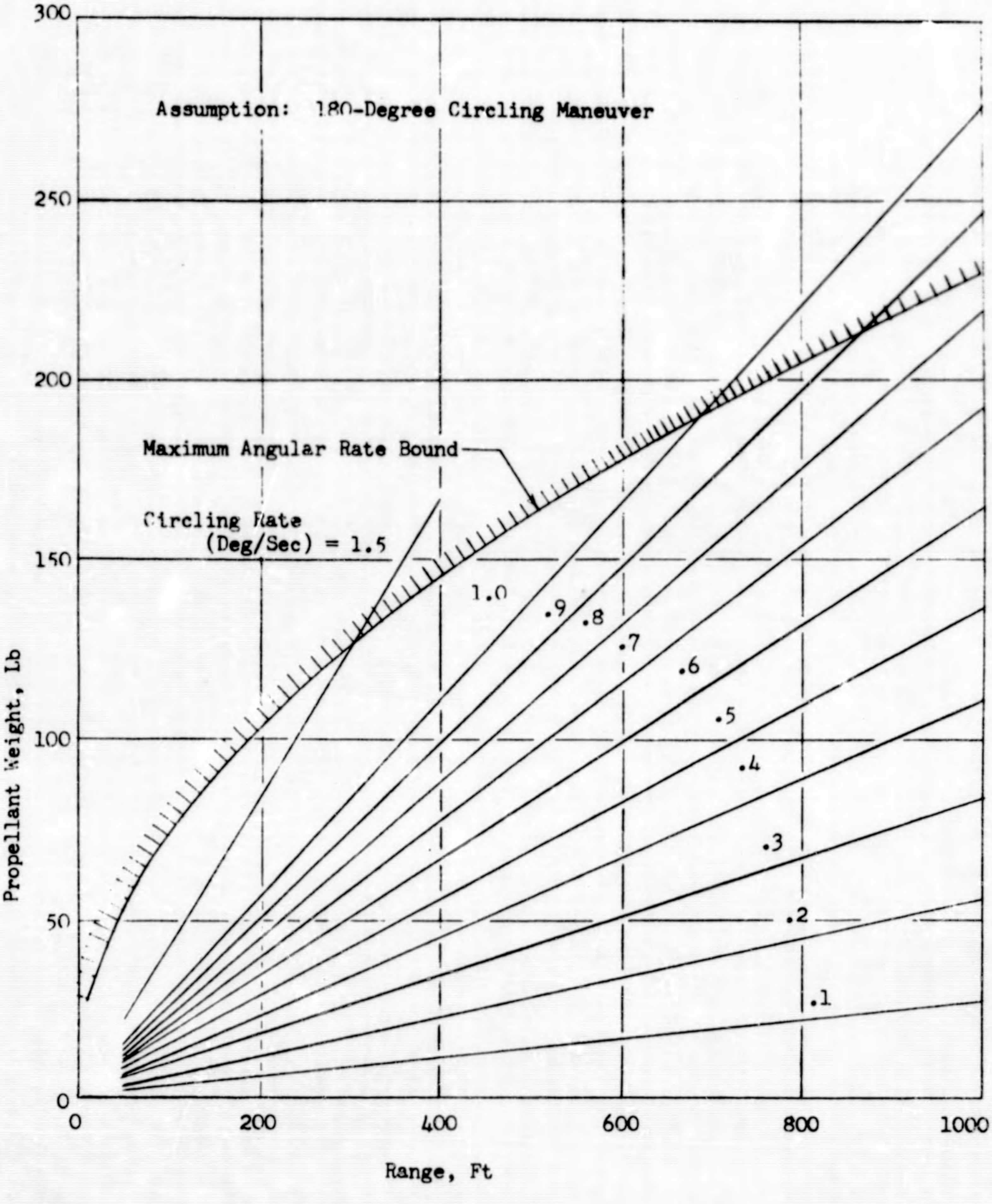


Figure 3.3-5 Circling Propellant Vs Range

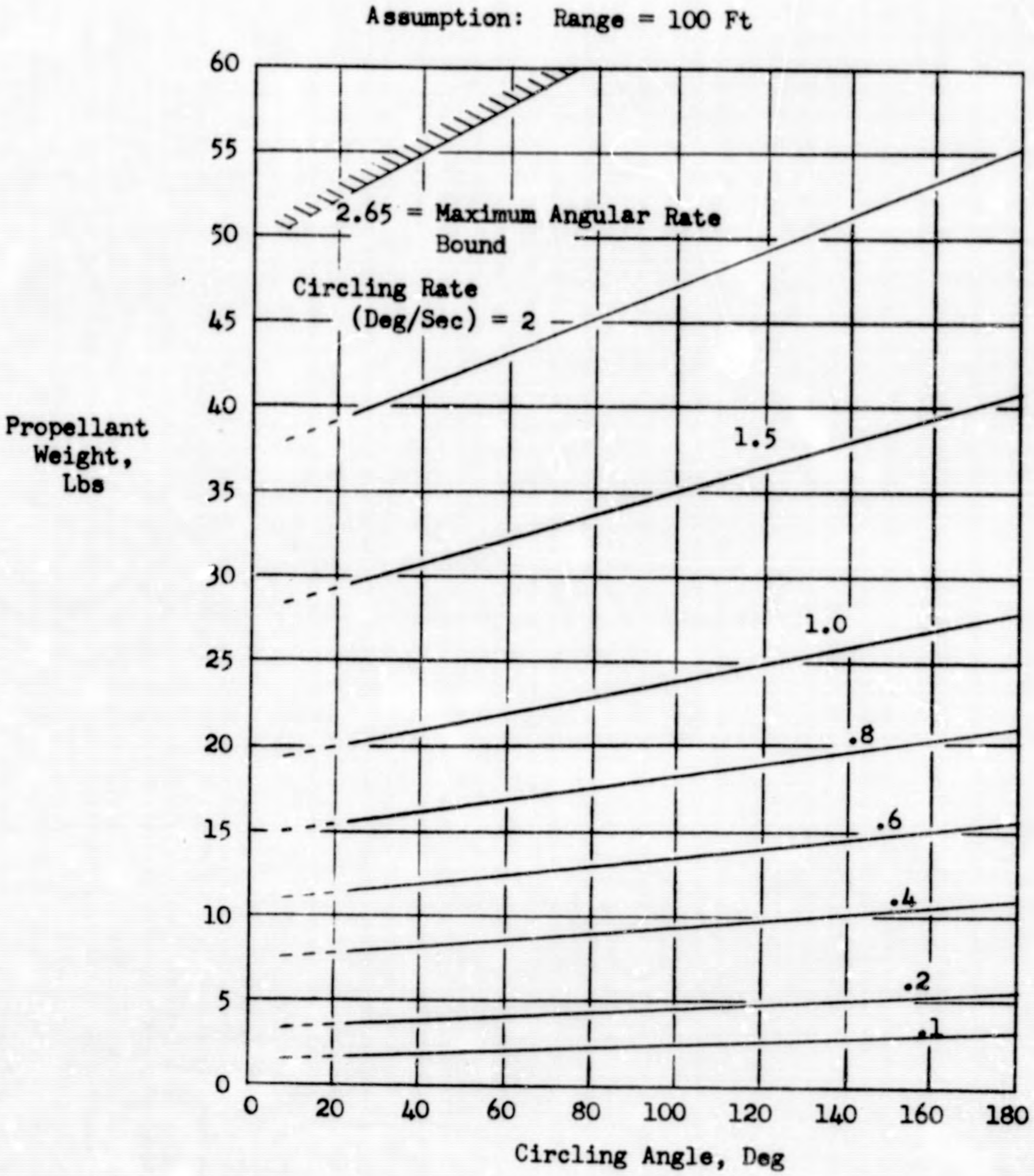


Figure 3.3-6 Propellant Usage Vs Circling Angle

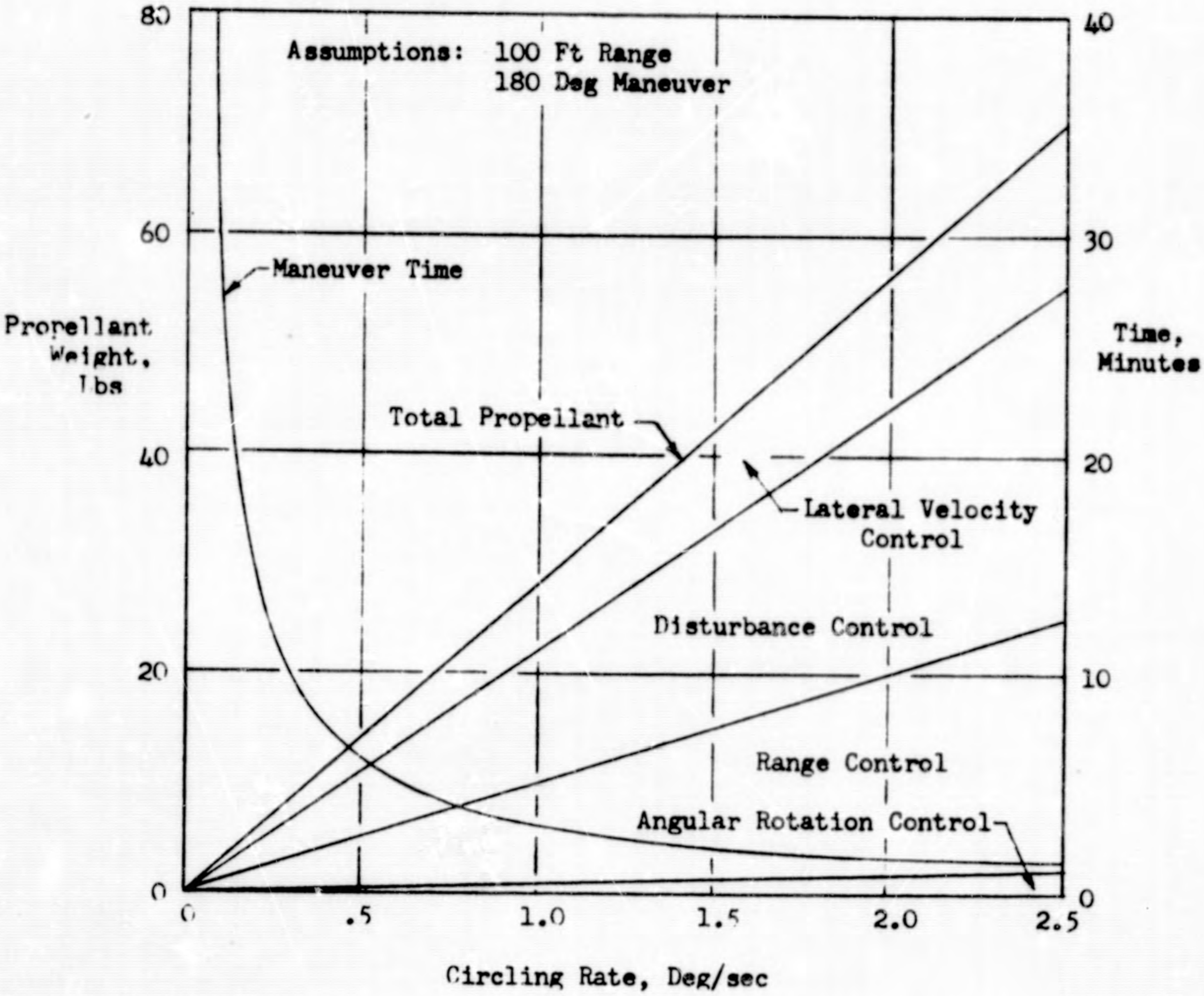


Figure 3.3-7 Propellant and Time Vs Circling Rate

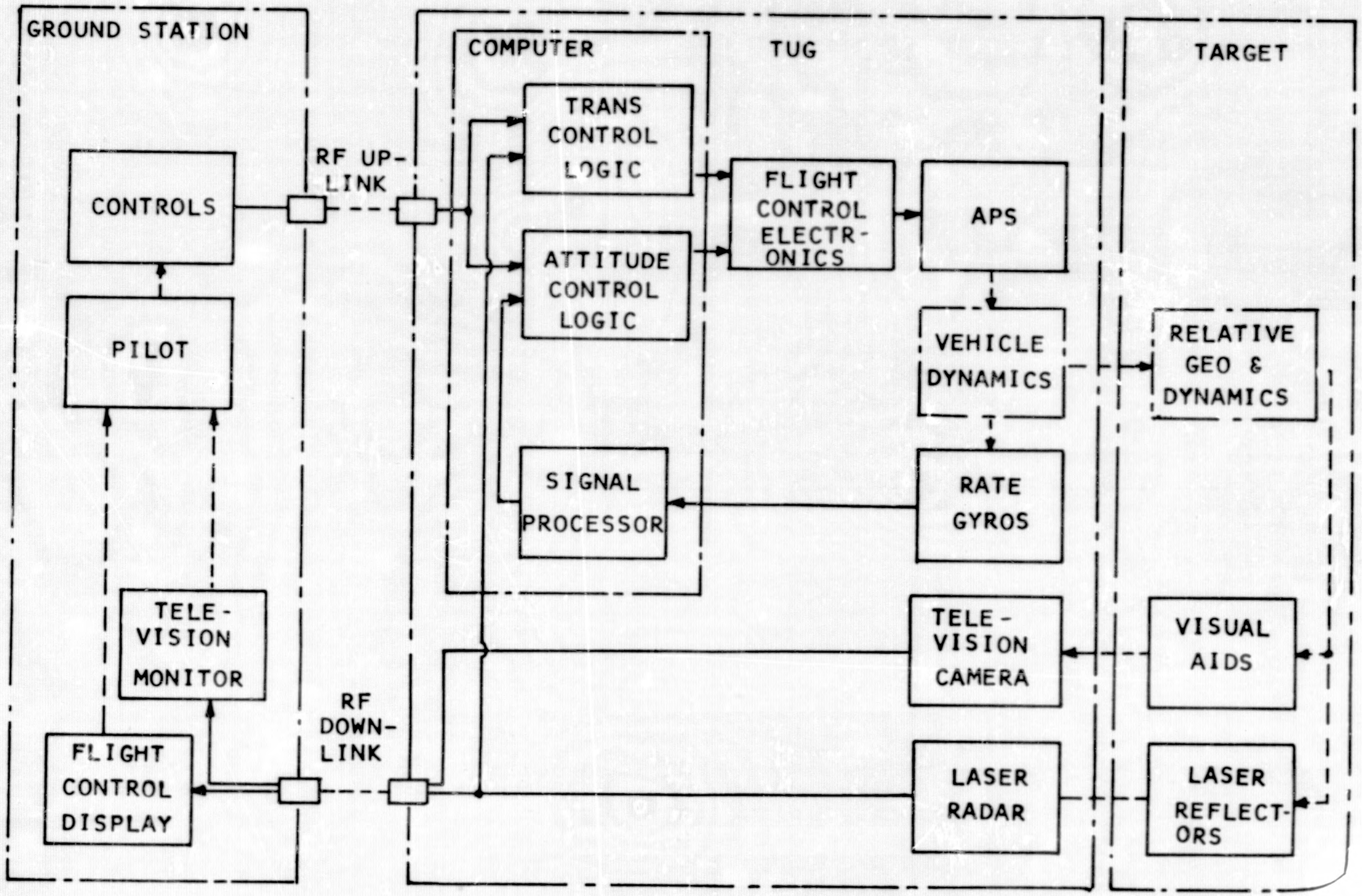


Figure 3.3-8 Docking System Functional Block Diagram

attitude hand controller in any of the three axes produces proportional angular rate commands to the vehicle stabilization loop. When the controller deflection is zero, not only is zero rate commanded, but an additional attitude loop is closed in the stabilization system to reduce drift. Both attitude and angular rate feedback are generated by the gyros.

The translation control is a similar system, controlling both position and velocity; however, two other features are added. The translation is measured with respect to an axis system fixed to the target docking port by means of the laser radar feedback. Positional hold capability is provided in all three axes as long as the three docking port reflectors are in view of the laser. The axial position hold is optional and can be removed by a pilot-originated discrete command. Either constant range or constant closure velocity can be retained by this means.

The system, as described, is conservative in that use of the laser information for both feedback and displays during final closure may be superfluous. It is recommended, however, pending proof by simulation that the pilot is capable of statistically successful docking operations without laser display data or feedback control.

Conversely, additional control could be included if needed, by using an angular feedback derived from laser data which would stabilize attitude and attitude rate with respect to the target docking port axis system.

A completely automatic docking system could be designed using the equipment already described, with the addition of on-board software. The computer programs would handle both the circling and closure phases of docking, leaving only the visual inspection function to the remote pilot. Automatic docking software was not required for the Point Design Tug, and was not studied. A preliminary study of the software has been reported in Reference 3.3-2, however.

In the recommended system the pilot guides the closure operation until contact and latching is completed, according to a preplanned nominal range and range rate schedule. His cues are derived from the displays at his disposal. Principal among these displays is the television monitor; others include:

Laser and laser-derived downlink data

Tug and Payload (target) operating status downlink data

Ground-generated timelines and schedules

His command capability includes the following functions:

Integrated attitude and attitude rate commands

Integrated translation and velocity commands

Discrete commands for mode switching

Selected Payload Docking Operational Modes

Docking operations begin at 300 meters after completion of the stationkeeping period. If the Tug is aligned within approximately 25 degrees of the target docking port, the pilot manually maneuvers for closer alignment, then begins the final approach. A circling maneuver is necessary if the Tug is positioned further than 25 degrees from the target docking port. In this case the pilot closes to approximately 100 feet, then initiates the circling maneuver for closer alignment. When aligned, the pilot uses display data and a range/range rate schedule as cues while commanding the final approach.

Passive Docking Operational Modes

After the completion of the active rendezvous phase with the EOS the Tug assumes a passive role in preparation for docking. The process of passivation involves generally the cessation of main propulsion, the safing of main tanks, and the sequential de-energizing of all non-critical electronics. Many of the details of passivation need not be defined until further Tug and EOS development, since they do not influence timely issues. The status of attitude stabilization during the docking phase, however, is relatively important as it affects the requirements of both vehicles, and should be decided at this time.

The obvious alternate choices are to inert or to maintain the APS during docking. If the APS is not operating during docking no hazard to the EOS from spurious engine operation can occur. From the other point of view, without attitude stabilization, any appreciable disturbance torque can prevent docking and may result in Tug loss. There are many more aspects of the question, and these are summarized in Table 3.3-6.

Perhaps the conservative approach is to assume active stabilization, so that a high probability of successful docking is assured. The hazards of active stabilization in close proximity to the EOS can be reduced by design practices; but disturbances cannot be damped without the APS.

Tug-active stabilization during passive docking with the EOS is therefore tentatively recommended, in full appreciation that further study may lead to the opposite approach.

Table 3.3-6. Comparison of Active and Inert APS Advantages During Tug/EOS Docking

Factors in Favor of Active APS	Factors in Favor of Inert APS
• Abortive attempts to dock can result in rebound angular velocities. These are damped by the APS to permit further docking attempts. • Disturbances in the form of angular velocities resulting from unforseen outgassing, rotating equipment, or external environment such as gravity gradient, atmospheric drag, or solar pressure can be damped. • Delays in the docking schedule can be compensated by re-orienting or stabilizing over a prolonged period of time. The EOS can command a re-orientation of the Tug during the docking phase to avoid adverse sun angles, etc.	• Although fail-safe is specific design requirement, two failures can cause an engine to fire continuously and create a potential hazard. • Future docking simulation studies may show that pilots can reliably dock to a Tug with motion resulting from a predictable disturbance. • Protection for APS plume impingement on the EOS need not be provided. • The APS can be re-energized after inerting, if angular velocities become too large.

3.3.4 Subsystem Operation

The subsystem will be in a power down mode prior to initiation of rendezvous and docking activities. The data management subsystem will activate and checkout the appropriate subsystem components required for either of the particular operational modes described in the following paragraphs. The data management subsystem will power down the components after completion of an operational mode.

Payload Rendezvous and Docking

In the 3000-lb payload retrieval mission the target is acquired before rendezvous orbit circularization. Target acquisition takes place after circularization in the 4160-lb payload retrieval mission.

Acquisition of the target is accomplished using on-board knowledge of target position, computing the direction of the target with respect to the Tug, and searching with the laser radar when within range. Preburn acquisition (3000-lb payload mission) data are used to reduce position and velocity uncertainties relative to the target, and to modify the propulsion guidance command which places the Tug nearer to the target at end burn. Circularization is nominally in the target orbit, and lags by 300 meters plus the position uncertainty radius. Reacquisition of the target occurs after circularization with a high probability of success. Postburn acquisition (4160-lb payload mission) also occurs nominally from the target orbit, but the Tug lags by a larger radius of uncertainty. Closure to 300 meters proceeds if the target is located by the initial laser scan. Autonomous navigation refinement operations proceed if the target is not located, resulting in a new predicted target direction.

Television acquisition of the target can occur prior to, or during, the stationkeeping period which succeeds the rendezvous phase. The laser is switched from the acquisition mode to the scan mode during the stationkeeping period, and is kept in the scan mode throughout the docking operation. The pilot may first view the target from an arbitrary angle. If the Tug is aligned to the docking port within approximately 25 degrees, the pilot proceeds with final closure. With a greater angle, the pilot closes to approximately 100 feet, following a radial path, then switches to a semi-automatic circling mode to align the docking port. Final closure is conducted manually by the pilot using cues from the television and laser downlink data. The television lights are turned on at the descretion of the pilot at close range.

EOS Rendezvous and Docking

With all Tug missions the target is acquired prior to circularization in the EOS orbit. At the nominal 100 km range the Tug conducts an autonomous attitude maneuver to orient the laser side-viewing window to the predicted EOS direction. After target acquisition the Tug reorients for the circularization burn, then reorients again to re-acquire the target. Final maneuvers bring the Tug to within 300 meters of the EOS, where passive docking attitude control begins. The laser is turned off at this time.

TACAN acquisition of the Tug can occur prior to, or during, the stationkeeping period of passive docking. Signal transponding capability will continue until docking contact.

3.3.5 Component Characteristics

The Rendezvous and Docking Subsystem consists of seven functional elements, of which three are off-board: (1) payload and EOS mounted laser reflectors, (2) television docking aids, and (3) the ground control station pilot display and control console. These three elements are discussed in general terms in the following paragraphs. The remaining four elements: (1) TV camera, (2) TV camera lights, (3) laser radar, and (4) EOS ranging transponder, are discussed in more detail.

Payload and EOS Mounted Laser Reflectors

All return payloads will be equipped with passive laser reflectors which, in concert, provide a strong directional return signal during the rendezvous and circling maneuver flight phases. Furthermore, three reflectors are to be placed on the payload such that they are clearly visible to the Tug laser at 0 to 100 ft range.

Placement of the reflectors about the payload for rendezvous acquisition depends on the particular configuration of the payload. A simple cylinder would require five clusters, not counting the docking port reflectors. Each cluster would provide hemispherical reflection. Four of the clusters are mounted around the cylinder periphery and the fifth is on the end opposite the docking port. Reflector placement becomes more critical with an irregularly shaped payload. Detection of more than one reflector during rendezvous would be made inconsequential.

The three docking port reflectors will be placed in a triangular formation centered at a point on the payload opposite the corresponding Tug laser position. Individual reflector identification is made possible by using Tug on-board knowledge of the distance from one reflector to each of the other two. These distances should be made unique to provide identification for roll indexing.

EOS-Mounted Laser Reflectors

Laser reflector mounting provisions on the EOS need not be as complicated an arrangement as is required for the payload. Since the EOS is dedicated to Tug recovery, it will orient so that a single reflector cluster gives a hemispherical field of view centered at the nominal Tug line of sight. The reflector cluster can be mounted any place on the EOS that does not experience an extreme heating environment during atmospheric launch or entry. A more specific description of the mounting provisions cannot be given until detailed studies of reflector material properties in the expected environment have been conducted. However, the mounting provisions are not expected to be particularly constraining. The inside of a cargo bay door, for example, may provide an ideal mounting position.

Payload Visual Docking Aids

The actual configuration and precise specifications of visual docking aids to be imposed on the payload should not be defined without strong support from man-in-the-loop simulation data. Although many simulations have been conducted in the past for Apollo, LM, and numerous advanced vehicles, these provide excellent general background, but are not specifically applicable. Therefore, no criteria are identified in this section; rather, several interesting concepts are discussed.

A simple cross, painted on the forward surface of the payload opposite the television camera, gives the pilot a rough indication of range and three-axis attitude. Pilots are typically able to judge translational and angular rates from positional data. Thus, all of the information necessary can be gained from the simple cross. Both the accuracy and the unambiguous nature of data interpretation by the pilot are in doubt, however.

If a small standoff cross is mounted at the center of the painted cross; the pilot, by aligning the two crosses, can gain much accuracy. He has three objects to align in this case: the painted cross, the standoff cross, and the television picture (which should have reticle markings). Unfortunately, he still experiences ambiguous cues, in that translational and rotational errors cannot be separated. Furthermore, because the visual aids are mounted far from the vehicle centerline, small roll errors produce large centerline miss distance errors.

If the display combines laser data with the three-object visual aid arrangement the pilot can probably perform an adequate docking operation. The laser is mounted on the diameter opposite the television camera and small visual roll errors are easily measured as large lateral miss distances by the laser.

Many versions of more exotic visual aids may be conceived to magnify alignment errors so that they are easily interpreted by the pilot. One of these involves a small light beam from the laser side of the Tug, which reflects from a shaped mirror on the payload. The mirror is angled at 45 degrees and the beam is directed across the payload face to the area of the painted cross, where a second 45-degree mirror is mounted. When the pilot has maneuvered the Tug so that the light image is centered in the cross, the errors are nulled. Simulation studies are of course needed to verify the feasibility of the method.

For the purposes of this study, the three-object visual aid concept, including laser data, is recommended. The approach is used in the pilot displays and controls discussion.

Remote Pilot Displays and Controls

As is the case with payload laser and visual aid requirements, the specific arrangement of displays and controls must be defined as the outcome of simulation studies. A comprehensive description of the preliminary

(pre-phase B Tug development) simulation study plan is included in this report as a Supporting Research and Technology development item.

Data to be included in the pilot display equipment are:

1. Television picture

2. Laser data including three-axis range, range rate, attitude and attitude rate

3. Tug and payload operational status

4. Ground generated timelines and schedules.

The controls are currently envisioned to be:

1. Three-axis attitude rate command, with attitude hold null position, right hand controller

2. Three-axis translational velocity command, with translational hold null position, left hand controller

3. Discrete command mode switching.

All of the display items can be superimposed on the television picture for the convenience and efficiency of the pilot.

The laser data should be composed of smoothly varying flight variables; hence, the scan rate of the laser during the docking operation is required to be approximately the same as the television scan rate. Ten frames per second is specified as the slowest permissible laser scan rate. To meet this requirement the laser tracking logic will initiate a sequential "mini scan" about each of the reflectors. No major technical problem in meeting the requirement is anticipated.

Television Camera

Baseline Selection: RCA Mod. QTV-9; Lens, Schneider Optic Kreuznach, Cinegon 1:1.4/16 CCTV 02, Type CM 120. Table 3.3-7 summarizes the physical and performance characteristics of the unit.

Theory of Operation

The TV camera uses conventional monochrome television techniques to provide information. The system operates at commercial (NTSC) scan rates.

The unit employs a Silicon Intensifier Target (SIT) tube for image sensing which has high sensitivity and relative immunity to image burn. The camera is a complete system and provides a composite video output signal with only a single power input (+28 VDC). A dc/dc converter is used to transform the 28 VDC input to all voltages required for circuit operation. Critical voltages are stabilized by internal voltage regulators. A sync generator

Table 3.3-7. TV Camera Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	**Characteristics**
Weight	Camera 7.5 lb Lens 0.5 lb Total Assembly 8.0 lb
Size	16 x 7 x 7 in.
Power	12 watts
Voltage	28 VDC
Operating Temperature Range	-14°F to +122°F
Installation	Must look forward and have a 52.6°FOV for each axis. Should be mounted opposite to scanning laser radar.
Reliability MTBF	500 hrs
PERFORMANCE CHARACTERISTICS	
Parameter	**Performance**
Silicon Intensifier Target (SIT) Tube	
Spectral Response	S-20
Resolution	Min +0% at 200 TVL
Signal Current	Typical 400 nA
Dark Current	Max 15nA at 30°C
Sensitivity and Dynamic Range	1 to 10,000 foot-lambert
Gamma	1.0
Scene Dynamic Range	32/1 min
Shading	Max 20%
TV Video Signal	
Sync Tip	0 ±0.1 V
Blanking	0.280 V ±50 MV
Video Clip Level	1.05 V max
Intra-Scene Dynamic Range	32:1 (Ten 2 gray steps) min
Signal-to-Noise (Random)	35 dB (p-p/rms) for a 500:1 light range, measured in a 2.-MHz bandwidth
Signal-to-Noise (Coherent)	40 dB (p-p/rms) within 2.0-MHz bandwidth

Table 3.3-7. TV Camera Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS (Cont)	
Parameter	Performance
Resolution (Amplitude Response)	Relative to response @ 25 TV lines; 90% min @ 100 TV lines (1.25 MHz) 80% min @ 200 TV lines (2.5 MHz)
TV Scan Rates & Tolerances	
Number of lines per frame	525
Number of fields per second	59.94+
Number of lines per second	15,734.26+
Number of frames per second	29.97+
Tolerance on number of lines/sec	±100 parts per million
Interlace Ratio	2:1
Vertical Blanking	18.4 lines (min), 21.0 lines (max) (0.07 V to 0.08 V)
Horizontal Blanking	0.165 H (min), 0.180 H (max)
TV Performance Summary	
Sensor	SIT camera tube
Sensitivity	Greater than 32 dB signal-to-noise at 3 foot-lamberts
Resolution	80 percent response at 200 TV lines
ALC Dynamic Range	1000 to 1 (minimum)
Non-linearity	3 percent (maximum)
Gray Scale	10 - 2 steps
Video Output Level	1.0 volts p-p into 75 ohms Full EIA sync
ALC	Peak or average detection modes.

furnishes timing pulses for video and deflection circuits, and for synchronizing the operation of the dc/dc converter.

The video circuits amplify and process the sensor output to provide the required camera output format. The processing includes automatic light control (ALC) and automatic gain control (AGC), aperture and gamma correction, sync and blanking addition, and black and white clipping. All electrical interfaces are routed through a single, 26-pin, zero-g connector located at the rear of the TV camera. Figure 3.3-9 is a functional block diagram of the TV camera.

Component Selection Rationale

A black and white version of the RCA Lunar Surface Color Camera (RCA Mod QTV-9) produced for NASA under Contract NAS9-10781 and a fixed lens assembly with 52.6° FOV (Schneider Optic Kreuznach, Cinegon 1:1.4/16 CCTV 02, Type CM 120) were selected for the baseline. A black and white version minimizes communications subsystem bandwidth requirements thereby reducing power requirements.

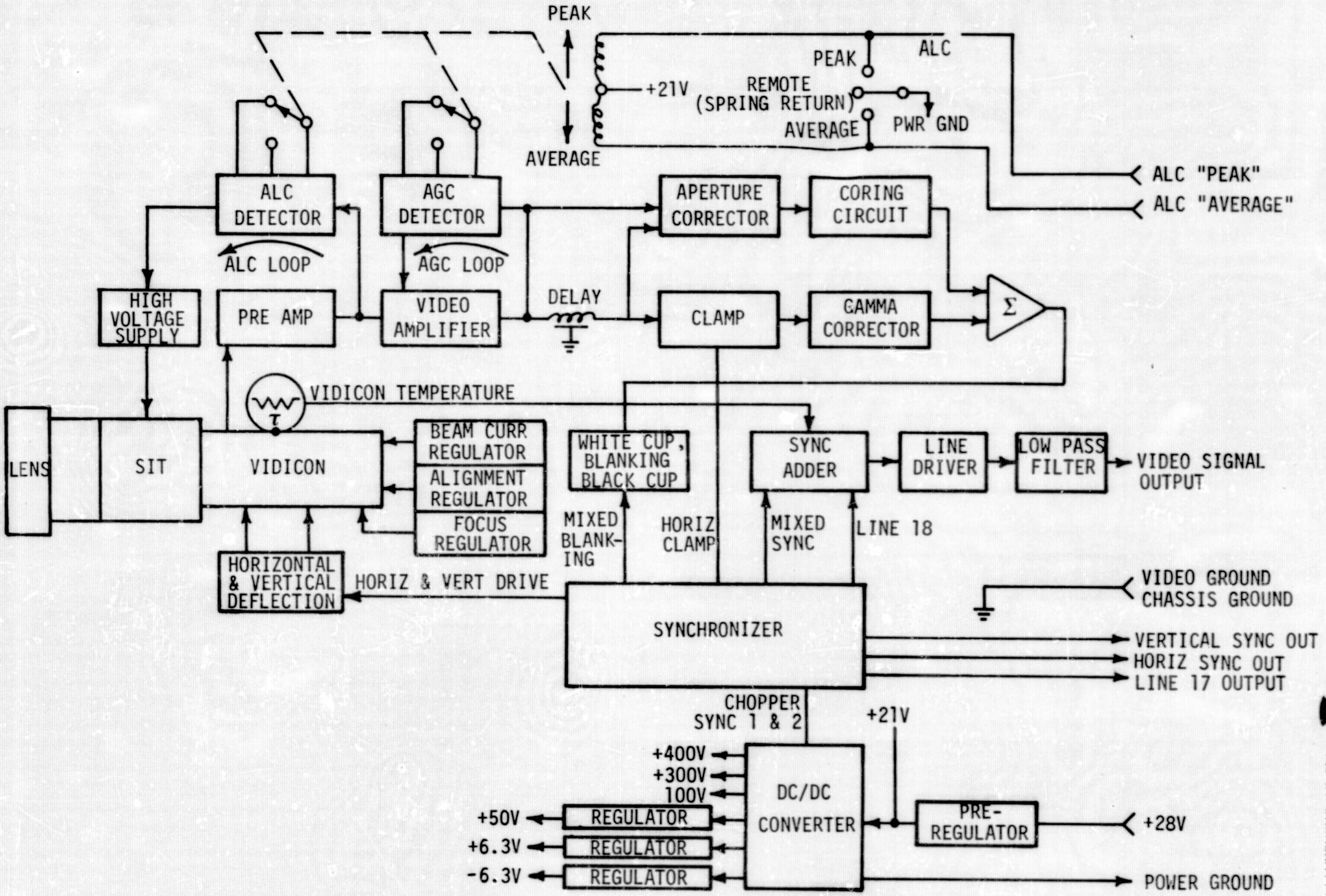


Figure 3.3-9 TV Functional Block Diagram

Since the Scanning Laser Radar System will be used to approach a given target to within 300 meters, there is no need to identify it with a television system to any greater distance. With a 52.6° field of view, the camera can scan a field whose x and y dimensions are nearly equal to the range, or about 300M x 300M. At this range, an illuminated 5 meter target will be scanned by approximately 8.85 TV lines at a 525 TV scan lines per frame rate. This should be sufficient for target recognition (8.0 ± 1.6 TV lines) in accordance with data contained in Reference 3.3-3. At the same time, at very close payload ranges, a target on the payload can be used for at least roll indexing. If the camera lens is recessed one foot from the Tug mold line, a 2 ft x 2 ft target would be clearly visible to provide visual docking data when only one foot away from contact. Considering the experience gained from the Apollo program where the FOV available was approximately 20° and 4 feet off center line when docking, the selected camera assembly should be more than adequate for the task.

The Westinghouse MINSIT and WTC-20 compact cameras have been considered as alternates. At the present time, the weight differences are minimal and the space qualification status of the RCA camera overshadows other minor differences.

Television Lights

Baseline Selection: Modified Grimes Manufacturing Co. Model 30-0390-9. Table 3.3-8 summarizes the physical and performance characteristics of the unit. Detailed data for the unit may be found in Reference 3.3-5.

Theory of Operation

The television and illumination equipment provide the primary source of information upon which the docking operation will be based. Three basic functional requirements must be satisfied: (a) it must provide the capability of acquiring the target from at least 300 meters range, (b) at close range it must illuminate the target sufficiently so that even with sun interference, visual cues needed for precontact alignment will be evident, and (c) it must illuminate the target such that at very close range, the light intensity will not exceed the dynamic range of the television camera.

The field of view FOV of the television camera lens has been selected to provide detection of at least a partially sun-illuminated target at ranges of at least 300 meters. This allows for the widest FOV at predocking ranges and reduces alignment problems prior to docking.

Recent simulation tests described in Reference 3.3-4 have shown that a light intensity of 200 foot-candles is needed for appropriate illumination of the target at distances of 5 feet if there is sun interference. Additional analysis indicated that the illumination could be reduced to values as low as 100 foot-candles or 100 lumens per square foot.

Two conditions need to be satisfied in illuminating the target area required for mutual vehicle alignment: (a) the light beamwidth must be narrow

Table 3.3-8. Television Lights Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	1.5 lb.
Size	2 x 2.5 x 7 in.
Power	52 Watts
Voltage	28 Vdc
Operating Temperature Range	-200^{o} F to $+250^{o}$ F
Installation	Mounted adjacent to TV camera
Reliability MTBF Space Operations Ground Operations	 60 Hrs 5 Hrs
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Light intensity ratio reflected from the target and incident upon the camera lens	10,000
Minimum light intensity reflected from the target when the Tug is at 4.8 ft range	100 Lumens/Ft^2
Spot diameter on target when Tug is at 4.8 ft range	4.75 Ft

enough to provide the necessary lumens per square foot on the target at a selected range; and (b) the beam must be wide enough to illuminate the whole target used for precontact alignment.

In order to determine the maximum pre-docking alignment range, several conditions are specified or assumed:

1. Sunlight reflected from the target area activates the automatic light control (ALC) section of the television camera and effectively reduces the gain of the SIT tube image section. The reflected light incident on the camera lens sets the upper operating limit of the ALC dynamic range.

2. The lower limit of detectable light is 10,000 times less than the upper operating limit of the ALC dynamic range.

3. The light beam has the same FOV as the television camera lens, or about 52.6 degrees.

4. There is 100 watts of power available for the docking light assembly.

5. The light efficiency is 22 lumens per watt with 80 percent of the total input power delivered to the light.

Under these conditions the lights will provide 1760 lumens of luminous flux. Since the minimum illumination of the target desired is 100 lumens per square foot, the light beam spot diameter is 4.75 feet. With a 52.6 degree beam width, the range for the given illumination is 4.8 feet.

Assuming that there will be sufficient light incident on the target from both the sun and the light assembly for coarse alignment as the Tug approaches the payload, this range is sufficient for obtaining final visual cues required for fine alignment prior to making final contact.

Component Selection Rationale

A modified version of the 30-0390-9 light assembly (produced by the Grimes Manufacturing Company) which has been space qualified, is satisfactory for this purpose. The baseline selection is considered typical of equipment available from several vendors and was made to facilitate the point design approach.

Scanning Laser Radar

Baseline Selection: Modified ITT YAG Laser. Table 3.3-9 summarizes the physical and performance characteristics of the unit. Detail data describing the unit are contained in Reference 3.3-6.

Table 3.3-9. Scanning Laser Radar Characteristics Summary

PHYSICAL CHARACTERISTICS	
Item	Characteristics
Weight	Transmitter/Receiver 30 lb. Electronics 15 lb. Total Assembly 45 lb.
Size	Transmitter/Receiver 22 x 9 x 6 in. Electronics 15 x 9 x 9 in.
Power	155 watts peak, 55 watts standby
Voltage/Current	28 VDC/5.5 amp
Operating Temperature Range	$-100^{o}F$ to $+230^{o}F$
Non-operating Temperature Range	$-65^{o}F$ to $+165^{o}F$
Installation Requirements	Must look forward and have $30^{o} \times 30^{o}$ FOV parallel to the X-axis. Must also look parallel to the Z-axis.
Reliability MTBF	3000 hrs.

Table 3.3-9. Scanning Laser Radar Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS	
Parameter	**Performance**
Maximum Range	
Range	100 KM
Range Accuracy	± 0.1 M or ± 0.02% of measured range (whichever is greater)
Angular Accuracy	±0.1 deg.
Range Rate	2 KM/sec.
Minimum Range	
Range	1 ft.
Range Accuracy	± 0.1 ft.
Angular Accuracy	± 0.1 deg.
Range Rate	0.1 ft./sec.
Angular Rate	0.05 deg/sec.
Field of View	
Maximum	30° x 30°
Alignment	± 0.02 deg.
Scan Rate	
Field Coverage at Maximum Range	140 sec.
Field Coverage at 300	0.1 sec.
Passive Target Detection	
Number Detected Simultaneously	3
Detection Reliability (Single Frame)	0.99
Output Data	
Signal Type	Digital
Word Size	24 bits
Acquisition Mode Word Content	Line-of-sight (LOS) range, range rate, angular rate
Track Mode Word Content	LOS range, range rate, angle, angular rate, relative angles, angular rates

Theory of Operation

A functional block diagram of the laser radar is shown in Figure 3.3-10. The laser radar performs the following functions during the Tug-active rendezvous and docking mission phases:

1. Target (down payload) Acquisition - Detects the location of the passively-cooperative target after the Tug has successfully reached rendezvous range (≤100 KM).

2. Rendezvous Approach - Periodically measures the target range and range rate in Tug coordinates while the Tug is closing to initial docking range.

3. Maneuver for Docking - Periodically collects data necessary for the determination of the target docking port centerline, while the Tug is maneuvering for alignment.

4. Final Docking Approach - Periodically measures relative flight variables, including data sufficient to determine range, angle between respective centerlines, relative roll indexing, and all rates.

During the first and second steps the laser beam scans a square (30 x 30 deg.) section of space containing the target. Target detection occurs when a sufficiently strong signal is returned from a target-mounted reflector to the laser receiver. Repeated measurements of target position and velocity by the laser are used to guide the Tug to a position 300 meters from the target. While stationkeeping at this range, the detection of multiple target reflectors provides data to calculate the Tug position relative to the target centerline. These calculations are supplementary data for the remote pilot display, and also contribute to autopilot feedback information.

The laser system continues to provide target reflector information while the Tug is guided to the docking port. This data provides stabilizing feedback signals and is downlinked for pilot display use.

Component Selection Rationale

The ITT, YAG Laser (developmental - NASA Contract NAS8-23973) has been selected for the baseline. It is the only equipment of this type presently being developed under government contract which is being designed to meet customer-specified requirements.

Since a scanning laser radar was specified by the customer in the study guidelines, no alternatives were seriously evaluated. Alternatives which should be evaluated in future weight reduction studies, however, are identified in Section 3.3.6.

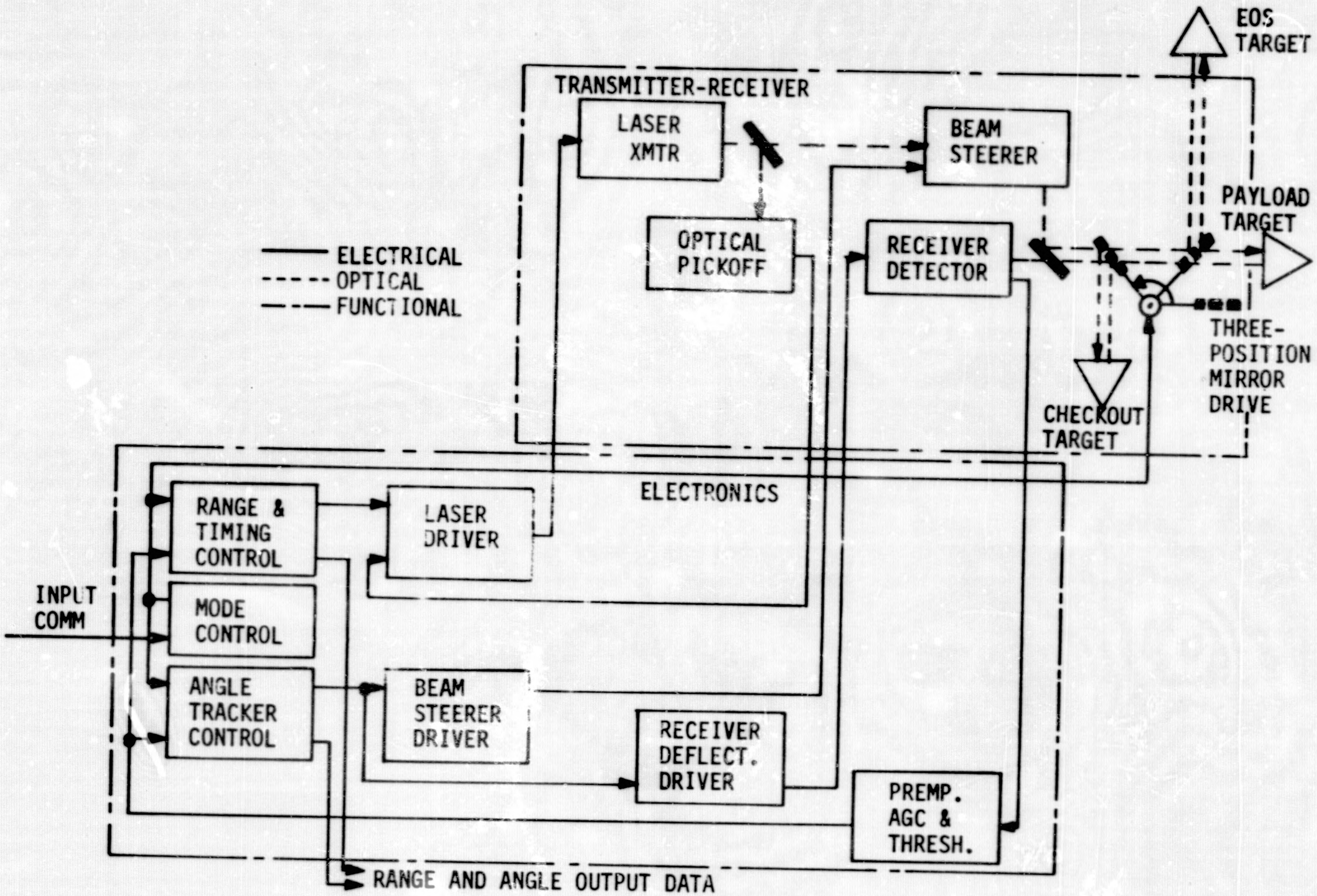


Figure 3.3-10 Scanning Laser Radar Functional Block Diagram

EOS Ranging Transponder System

Baseline Selection:

L-Band Transponder - ITT, Avionics Division, TACAN-DME, Mini-Beacon, P/N BN1-107
L-Band Antenna - Transco Products, Inc. P/N 22135 (L-Band)
Power Divider - Transco Products, Inc. P/N 40011 (L-Band)

Table 3.3-10 summarizes the physical and performance characteristics of the components in the system.

Theory of Operation

The L-band rendezvous transponder and antenna system will be utilized during EOS/Tug rendezvous. Similar to MSFN/Tug ranging, EOS/Tug rendezvous will be accomplished utilizing pseudo-random-noise (PRN) ranging techniques. In determining the EOS/Tug range and range-rate, the EOS transmits a phase modulated RF carrier with a (PRN) binary ranging code. The Tug's L-band transponder receives the interrogating signal (via the 2 L-band antennas and power divider) from the EOS, demodulates the ranging code, multiplies the signal by a known ratio and coherently remodulates and retransmits the signal which is phase related with respect to the receive signal for PRN ranging.

A simplified block diagram of the EOS ranging transponder system is shown below:

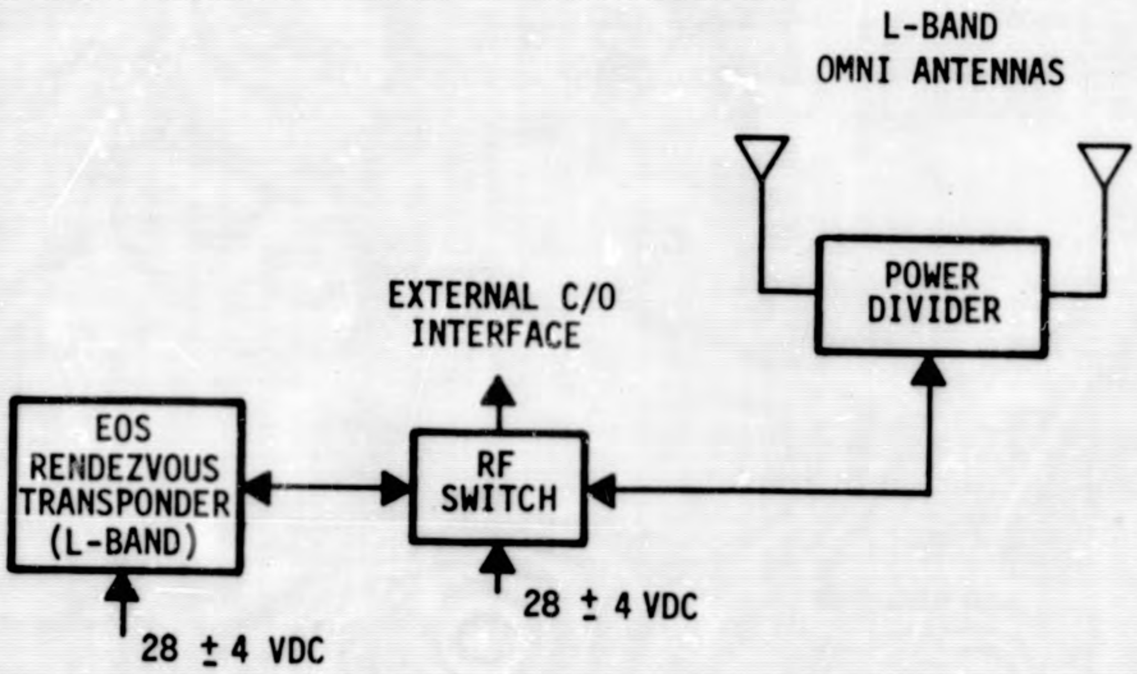


Table 3.3-10. EOS Ranging Transponder System
Component Characteristics Summary

Physical Characteristics of L-Band Transponder	
Item	**Characteristic**
Weight	11 lb.
Size	4.9 x 12.5 x 7.6 in.
Heat Dissipation	40 watts operating, 5 watts standby
Operating Voltage	28 ± 4 VDC
Operating Current	1.4 amperes @ 28 VDC
Operating Thermal Limits	$+32^{o}F$ to $+118^{o}F$
Installation	Standard avionic equipment mounting
Reliability MTBF	40,000 hours
Performance Characteristics of L-Band Transponder	
Parameter	**Performance**
Transponder Distance Accuracy	±0.25 nm (does not include airborne equipment accuracy)
Transmitter	
Frequency Range	Any 10 MHz segment in the: 962 to 1024 MHz with low-band multiplier or 1151 to 1213 MHz with high band multiplier.
Power Output	100 watts peak minimum
Frequency Stability	0.007%
Squitter Rate	1100 pulse pairs minimum
Receiver	
Sensitivity	-90 dbm for 60% replies

Table 3.3-10. EOS Ranging Transponder System
Component Characteristics Summary (Cont)

Physical Characteristics of Antenna, L-Band	
Item	**Characteristic**
Weight	12 ounces
Size	8.4 x 8.5 x 1.5 in.
Operating Thermal Limits	-80°F to +400°F
Installation	Flush mounted to external surface
Type	Annular slot
Reliability MTBF	300,000 hours

Performance Characteristics of Antenna, L-Band	
Parameter	**Performance**
Impedance	50 ohms nominal
Polarization	Vertical
Power Rating	5 KW peak
VSWR	Not greater than 1.6:1
Frequency Range	990 to 1200 MHz

Table 3.3-10. EOS Ranging Transponder System
Component Characteristics Summary (Cont)

Physical Characteristics of Power Divider (L-Band)	
Item	**Characteristics**
Weight	3 ounces
Size	2.5 x 1.75 x 0.5 in.
Operating Thermal Limits	$-65^{o}F$ to $+200^{o}F$
Installation	Standard avionic equipment mounting
Reliability MTBF	500,000 hours

Performance Characteristics of Power Divider (L-Band)	
Parameter	**Performance**
Impedance	50 ohms
VSWR	Less than 1.3:1
Frequency	1100-1400 MHz
Division	2 way even

3.3.6 Alternate Design Approaches

Four alternate design approaches, that were briefly examined during the study, warrant additional mention here. Each of them appear to be entirely feasible and should be kept as viable candidates, should the mission emphasis or design requirements change in the future.

Relative Docking Attitude Control

The recommended baseline docking system can be described as having independent payload and Tug attitude stabilization systems. While the payload is automatically held within a fixed inertial attitude deadband during docking, the pilot commands the attitude rate of the Tug, which also uses an inertial reference. The recommended baseline translation system has laser-derived three-axis range feedback which stabilizes the translation system in a manner similar to the attitude system, but with a payload reference. This difference is exhibited in the fact that the translation system cannot accumulate errors. The attitude errors build up as a result of system biases and external disturbances to which the two spacecraft react differently.

The alternate design is to incorporate an additional attitude feedback loop in the Tug attitude stabilization system. This loop would use laser data and would have the effect of generating error-correcting commands to the inner inertial loop. The pilot still would be capable of commanding the entire multiloop system, but would not be burdened with the task of constantly correcting small attitude errors.

The approach should receive further study to determine if the apparent increase in docking success probability that would result, is not more than offset by a decrease in reliability from the added system complexity. Also, it is not at all certain that the added capability is needed by the remote pilot; the answer must come from man-in-the-loop simulation studies.

Passive Docking With Inert APS

This topic has been fully discussed in the section on EOS docking operations, but is included as an alternate approach to insure that it receives future attention. The recommended baseline system uses active Tug APS to stabilize attitude within docking bounds. Since the Tug is docking to a manned vehicle, the EOS, the potential hazard of active propulsion in near proximity should be carefully examined. Further study may indicate that the APS can be deactivated without reducing the probability of successful docking, with increasing crew safety.

Automatic Docking

Studies of the Orbit-to-Orbit Shuttle (OOS) during 1971 (Reference 3.3-2), included a feasibility analysis of an automatic rendezvous and docking system. When the equipment requirements of OOS and Tug are compared, it is realized

that they are nearly the same. While the Tug includes equipment for remote pilot docking, the OOS includes additional software and computer capability. All other equipment is common. The benefits of automatic docking for Tug are based on the premise that the system has greater APS propellant efficiency. Automatic docking, if there were no need for visual inspection of the payload prior to docking, would relieve the Tug of all television and related communication equipment. The cost of a remote pilot facility could be saved.

In the automatic system, computer capability is needed to accomplish the following tasks now performed by the pilot:

Target approach angle identification

Circling maneuver initiation

Circling maneuver termination

Final closure control

Docking contact operations

The implementation of these tasks are included as one of the objectives of a Supporting Research and Technology Topic.

Use of Ground Tracking

An approach recognized but not considered due to the basic study guidelines is the use of MSFN ground tracking and command of both the Tug and rendezvous target to affect rendezvous of the two. This approach is feasible and would allow elimination of the laser radar.

3.4 COMMUNICATIONS

The baseline Communications Subsystem, as shown in Figure 3.4-1, is composed of three basic subsystems:

1) Antenna
2) Radio Frequency/Intermediate Frequency CRF/IF
3) Baseband

Each subsystem is described below along with a brief discussion of component functions.

Antenna Subsystem

The antenna subsystem provides omnidirectional coverage, thereby permitting transmission and reception at any time without regard for vehicle attitude or antenna pointing. Two flushmounted annular slots installed 180 degrees apart on the vehicle perimeter are coupled through a hybrid junction to the receiver input and the power amplifier output.

The hybrid junction is a four-port coupling device and serves three purposes: 1) it divides output power (from the power amplifier) between the two antennas, 2) it combines signals from the two antennas into a single input to the receiver, and 3) it provides isolation between the power amplifier output and the receiver input.

An isolation band-pass filter is inserted between the hybrid and the receiver to augment the total receiver/power amplifier isolation.

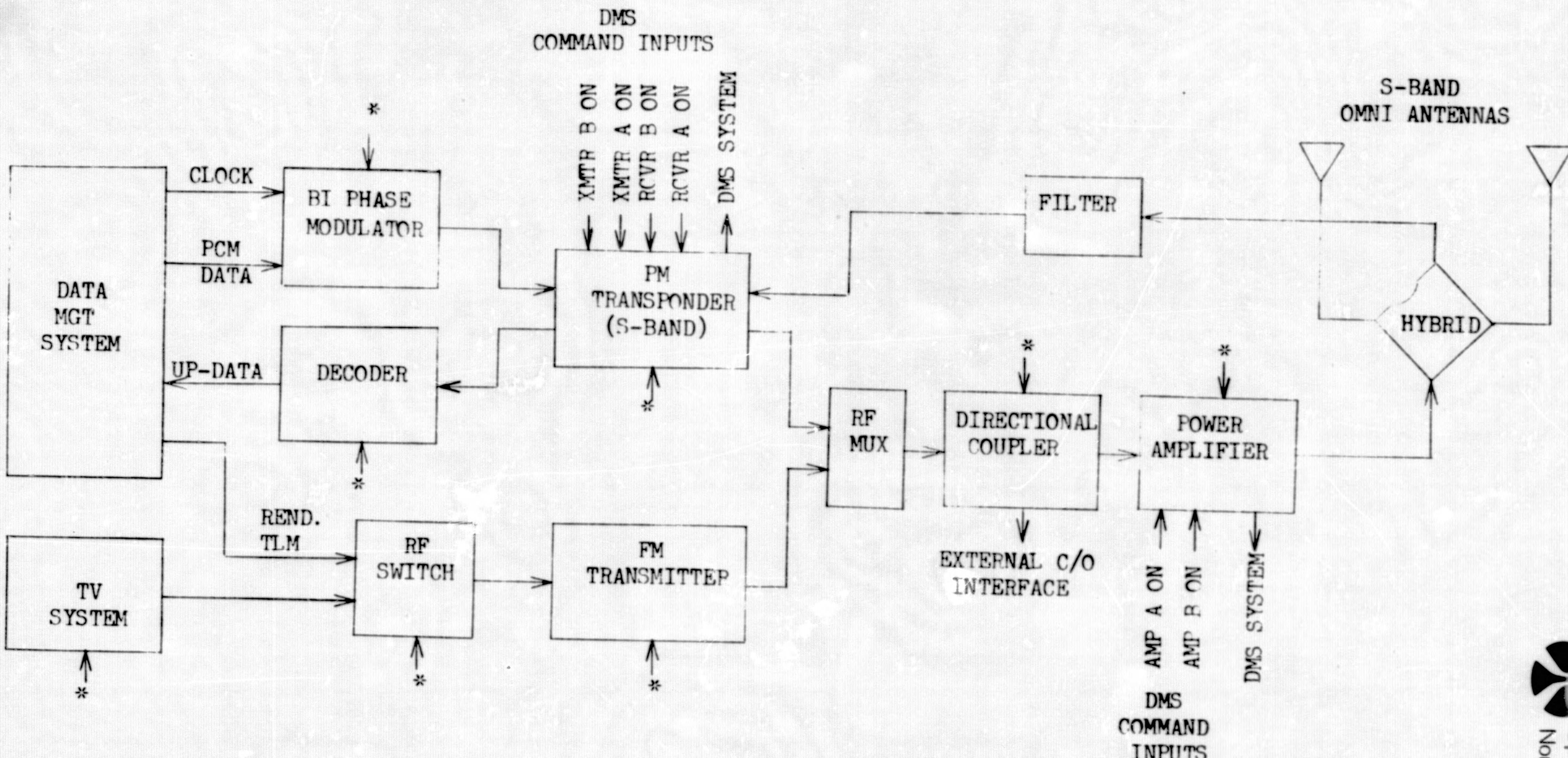
RF/IF Subsystem

The RF/IF subsystem consists of a dual power amplifier assembly, and RF multiplexer, a dual phase modulated (PM) transponder assembly, a frequency modulated (FM) transmitter, and an RF Switch.

The Tug transponder receives the Manned Space Flight Network (MSFN) uplink carrier (2.1 GHz) signal, demodulates the ranging code and coherently remodulates it on a 2.28 GHz downlink carrier. In addition, it demodulates the uplink data 30 KHz subcarrier and routes it to the command decoder.

The transponder transmitter simultaneously transmits the ranging code along with the PCM 1.024 MHz subcarrier. Both signals are phase modulated directly on the downlink PM carrier.

Increased system reliability is provided by the inherent redundancy in the PM transponder and the power amplifier assemblies. Each assembly contains two completely independent units which will be programmed through the DMS. Monitoring circuits in the DMS sense the redundant system failures and automatically provide switching commands to establish a fail-operational condition in either assembly.



* 24 ± 4 VDC

Figure 3.4-1 Communications Subsystem Functional Block Diagram

The FM transmitter provides the wideband transmission capability required for television. The video signal is frequency modulated directly on the FM carrier. The FM transmitter is also utilized to transmit limited telemetry to the Shuttle Orbiter vehicle to indicate Tug Status prior to docking. The coaxial switch provides the means to route either video or telemetry signals to the transmitter.

The RF multiplexer is a passive device that couples both transmitters to the power amplifier and also provides isolation between output stages when simultaneous transmission is required. The power amplifier is a wideband traveling wave tube (TWT) amplifier with a power supply and output filter for suppression of out-of-band signals. Output power is shared by the two downlink carriers.

Baseband Subsystem

The baseband subsystem consists of the bi-phase modulator and the command decoder. The modulator takes the pulse code modulated (PCM) serial bit stream (telemetry) from the DMS and bi-phase modulates it on to an internally generated subcarrier. The modulated signal is then routed to the transponder for phase modulaton onto the carrier.

The decoder accepts a demodulated bit stream from the receiver, detects the command word bits and subsequently decodes and distributes command words to the DMS. In addition, the decoder provides a limited capability for commands independent of the DMS.

3.4.1 Requirements

Review and analysis of the Tug study requirements and mission objectives resulted in the establishment of the Communications Subsystem guidelines and requirements as denoted in Table 3.4-1. The functional requirements established to satisfy the overall Tug Communications Subsystem are as follows:

- Telemetry
- Command and Up-link Data
- Tracking and Ranging (MSFN)
- Television (Payload Rendezvous & Docking)
- Ranging (EOS Rendezvous and Docking)

The Communications Subsystem external interfaces are shown in Figure 3.4-2.

Assumptions

The following assumptions were made in conjunction with this study.

- The prime communications link with the Tug will be in the S-band frequency range of 2.1 to 2.3 GHz.

Table 3.4-1. Subsystem Design Requirements

Design Requirements	Source
1. Communication system to be designed as an integral system with none of the subsystems designed to be removable as a kit or single unit.	NASA Guidelines
2. Communication system compatible with MSFN, Deep Space Network (DSN) and EOS.	NASA Guidelines
3. Tug vehicle to be un-manned	NASA Guidelines
4. Tug missions: Low earth orbit, 100 n.mi., 28.5°, geosynchronous orbit, 19.3 K n.mi.	NASA Guidelines
5. No communications interface between Tug and payload	NASA Guidelines
6. Tug communication system to be designed for a 20 mission life with refurbishment after each mission as required.	NASA Guidelines
7. Temperature environment limits; -100°F to +200°F maximum	NASA Guidelines
8. Communication system in unpowered condition during EOS ascent and descent operation except safety requirement	NASA Guidelines
9. Sustaining power to Tug during ascent and descent is EOS furnished. (300 W average, 500 W peak)	NASA Guidelines
10. The Tug will be provided a navigation update from the Shuttle prior to EOS/Tug separation.	NASA Guidelines
11. Provisions for monitoring of Tug critical functions for EOS crew safety to be monitored by the EOS crew at all times the payload is attached to the EOS.	NASA Guidelines
12. Tug to be designed for autonomous operation except in areas where remote control will significantly reduce weight without compromising operational effectiveness.	NASA Guidelines
13. Rendezvous radar laser system required with compatibility of detecting and tracking a satellite from 100 KM within 0.1 meters and 0.1 degrees.	NASA Guidelines

Table 3.4-1. Subsystem Design Requirements (Cont)

Design Requirements	Source
14. Detail checkout of Tug to be performed prior to installation on Shuttle. Minimum C/O at pad and in orbit. These functional checkouts to be self tests by Tug and will present only a go/no-go status to the EOS crew via hardline or RF link.	NASA Guidelines
15. Ground maintenance repair and refurbishment.	NASA Guidelines
16. Tug passive, cooperative element in EOS/Tug docking	NR Analysis
17. Tug active element in Tug/payload rendezvous and docking	NR Analysis

- The MSFN and DSN 85-foot antenna will be utilized when television transmission is required from the Tug during geosynchronous orbit mission phase.

3.4.2 Subsystem Trades

In reviewing alternate design concepts for the Tug Communications System, two significant trade analyses were performed. The first considered the potential advantages of a digital signaling system (rather than analog) for the television link. The second trade investigated the use of a high gain antenna system for the transmission of wideband data from geosynchronous orbit. The primary considerations of each trade analysis are summarized in the paragraphs below.

Digital vs. Analog Signaling

Through discussions with suppliers regarding television link equipment, it was concluded that comparable system weight, reliability and cost can be realized with either analog or digital equipments. Digital systems are more complex, and for moderate output signal-to-noise ratios (10 to 35 db) digital systems are generally less efficient in the utilization of bandwidth and power.

The intended application of Tug television data implies imagery comparable in quality to that of the Apollo Command and Service Module (CSM) system. The least acceptable carrier-to-noise ratio of the CSM television signal is 8.0 dB. Using a modulation index, m, of 2.0, the wideband improvement for a FM signal, with a carrier-to-noise ratio C/N = 8.0 dB, is 15.5 dB. The resultant output, signal-to-noise ratio (S/N) is 23.5 dB. A six bit digital system utilizing phase shift keying (PSK) with coherent detection requires 10 to 50 percent more power (depending on the detector characteristics) to realize the same output S/N.

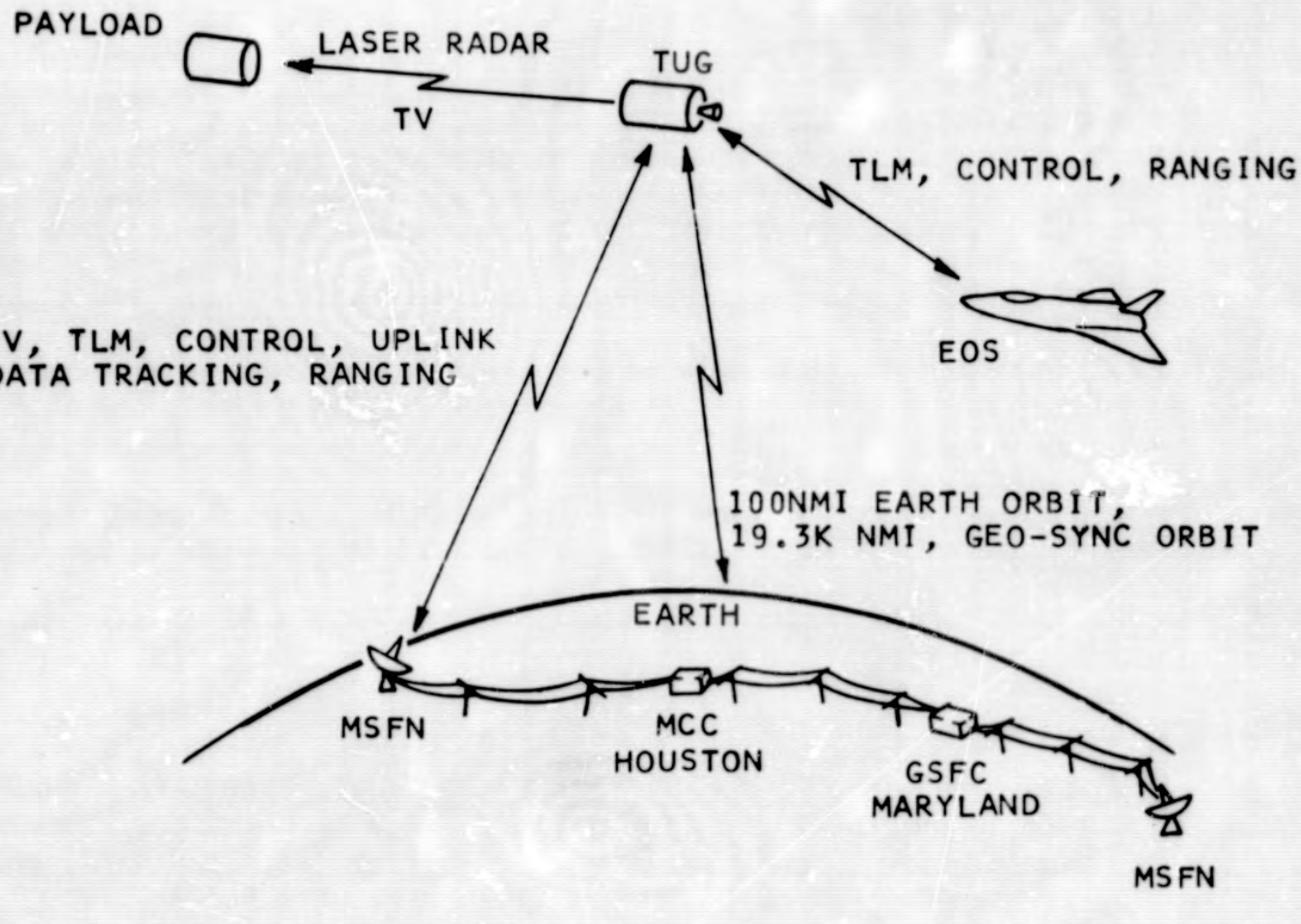


Figure 3.4-2 Communications Subsystem Interfaces

An additional factor is the required ground handling capability. The MSFN system is presently equipped for recording, processing and distributing analog (FM) television signals. Implementation of digital capabilities for similar information bandwidths requires a substantial expenditure.

On the basis of these considerations, the analog system was selected for use on the Tug. Derivation of the information bandwidth and sample S/N calculations are shown in Section 3.4.3 under Performance Analysis.

High Gain vs. Low Gain Antenna System

The Tug antenna system must satisfy two separate and opposing requirements. During all mission phases it must permit reliable reception by the command receivers from every aspect angle. Therefore, an omnidirectional (low-gain) system is mandatory. During periods when the televison link is operating, it is necessary to have an effective radiated power (ERP) of sufficient level to compensate for free space loss and system losses. ERP is the product of transmitting antenna gain and transmitter power, and for the Tug television link at geosyncrhonous ranges the minimum allowable ERP is approximately 2.0 dBW. This can be achieved with either a directional antenna and a low power transmitter or an omnidirectional antenna and a high power transmitter.

It was decided to use the omnidirectional system for the following reasons:

1) A minimum weight of 40 pounds is required for a directional system. The difference in low power and high power transmitters is only 2 to 5 pounds.

2) The hardware and software required for pointing a directional antenna is undesirable from the standpoint of reliability as well as complexity.

3) The cost of a directional system is orders of magnitude greater than an omnidirectional system.

3.4.3 Performance Analysis

Utilizing the functional requirements and operating constraints imposed by interfacing systems, performance parameters were derived for each propagation link. The Tug Communications Subsystem must be compatible with the MSFN, therefore, operating frequencies, modulation techniques, ranging methods, and radiated power levels were established on the basis of projected network configurations and characteristics for the 1979-1985 time period (Table 3.4-2).

In order to maximize the use of MSFN equipment, the Apollo Unified S-band (USB) concept was selected for the Tug Communications Subsystem. Both uplink functions, 1) tracking and ranging and 2) command and updata, are accomplished with a single uplink carrier. Similarly the downlink functions of 1) ranging and 2) telemetry are transmitted on a single downlink carrier. The relatively wideband characteristics of the television link coupled with the requirement to transmit television and telemetry simultaneously necessitates a separate carrier for the television link. Signal spectrums for each link are shown in Figures 3.4-3A through 3.4-3C.

Table 3.4-2. MSFN Interface Characteristics Summary (1979-1985)

Link Function	Frequency (GHz)	Modulation/ Coding	Data Rate/ Bandwidth	Antenna Gain (dB)*	System Noise Temp (K)	Transmit Power (Watts)
Transmit	2.1			**44/53	--	20×10^3
Commands		PCM/PM	2×10^3 bps			
Tracking		PRN	10^6 bps			
Receive	2.28					
Telemetry		PCM/PM	10^6 bps	44/53	55	
Tracking		PRN	10^6 bps			
Receive	2.27					
Television		FM/FM	3 MHz	44/53	55	

*Polarization is right hand circular.

**Gain values are shown for both the 30 foot and 85 foot dishes.

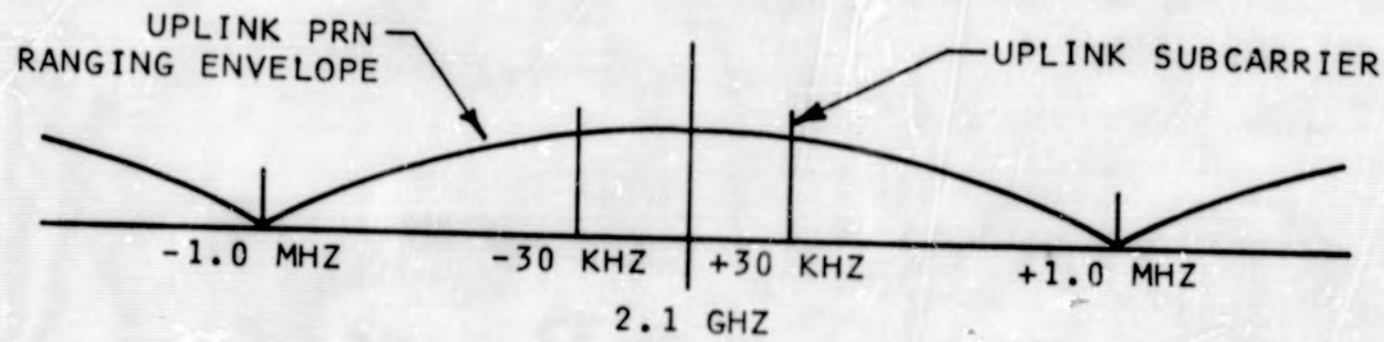


Figure 3.4-3A Uplink PM Signal Spectrum

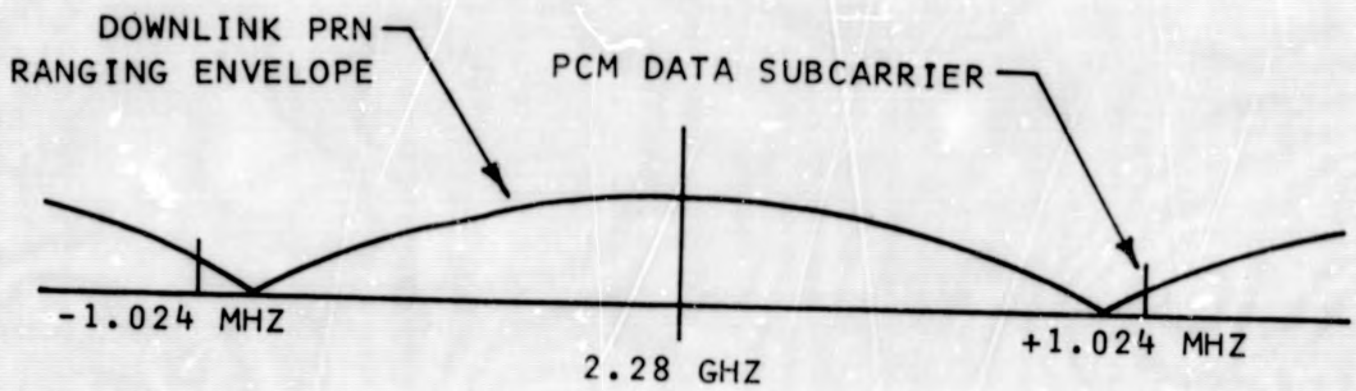


Figure 3.4-3B Downlink PM Signal Spectrum

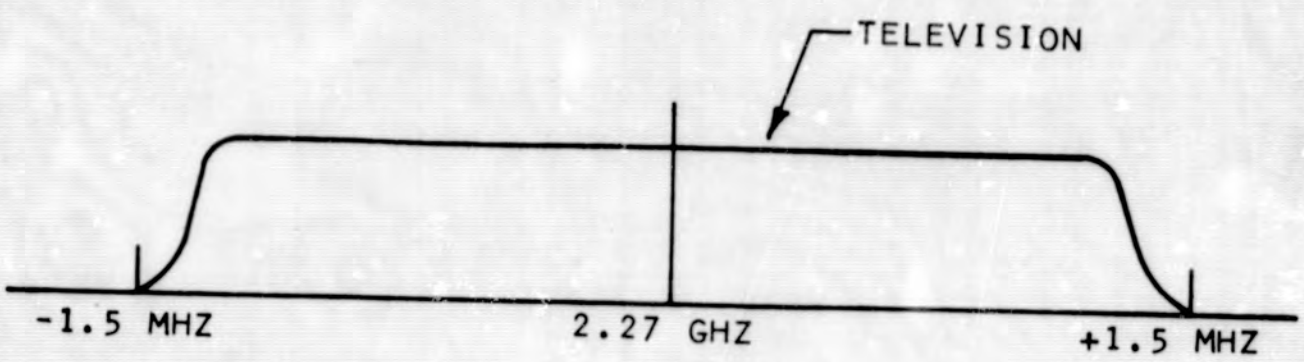


Figure 3.4-3C Downlink FM Signal Spectrum

Figure 3.4-3 RF Signal Spectrums

Link performance considerations are discussed in the following paragraphs along with appropriate rationale for system parameters.

Television Bandwidth Requirements

The television format characteristics specified by MSFC are 250 lines/frame at a frame rate of 15 frames/sec.

Assuming a complete contrast transition, white to black to white (sinusoidally) in a span of picture element equal to the spacing between adjacent scan lines and a frame rate of 15/sec., then the time per line in a 250 line frame with a useful line percentage of 0.95 is,

$$\text{Time/Line} = \frac{1}{(15)(250)(0.95)}$$

$$= 2.8 \times 10^{-4} \text{ seconds}$$

Assuming a 1:1 aspect ratio and a blanking pulse of 16% of the horizontal scan time, the time required to scan a line segment equal to the spacing between adjacent lines is approximately

$$t = \frac{(2.8 \times 10^{-4})(0.84)}{250} = 0.95 \times 10^{-6} \text{ seconds}$$

This time represents a half cycle of the desired variation of picture intensity. The corresponding video frequency is

$$f_m = \frac{1}{(2)(0.95 \times 10^{-6})} = 500 \text{ KHz}$$

C/N_o Required for Biphase Phase Shift Keying (PSK)

Digital link performance is normally measured in terms of the C/N_o required to achieve some maximum bit error rate (BER). Experience has shown that a BER of 10^{-3} is adequate for telemetry signals while digital command links are usually implemented with a BER of 10^{-5}. Modulation theory can be used to determine the required bit energy-to-noise density, E_B/N_o, for a given BER and modulation technique; e.g., an E_B/N_o of 6.8 dB is required to realize a BER of 10^{-3} with a biphase PSK signal. Given C/N_o and the bit rate, R_B, E_B/N_o is simply $\frac{C/R_B}{N_o}$.

C/N Required for FM Signals

The quality of an analog link using wideband frequency modulation (such as the Tug television link) is normally measured in terms of the predetection carrier-to-noise ratio and the subsequent wideband improvement when the improvement threshold is exceeded, i.e., C/N 1. Tug imagery quality requirements are comparable to those of the Apollo CSM, therefore the Apollo C/N requirement of 8 db was used for the Tug.

The carrier level, C, may be calculated as previously shown in the range equation. N is simple the receiver noise power within the predetection bandwidth, B,

$$N = kT_sB$$

where B = $2(m + 1)f_m$

m = modulation index (m = 2 for Tug)

f_m = highest modulating frequency (f_m = 500 KHz)

When C/N is large, a significant improvement is realized in the output signal-to-noise ratio, e.g.,

$$S/N = 3m^2(m + 1)C/N$$

m = modulation index

<u>Range Equation</u>

In the one-way range equation the carrier-to-noise density ratio is given as:

$$C/N_o = \frac{P_tG_tG_r\lambda^2}{(4\pi R)^2\ L_tL_rN_o}$$

C = Transmitted carrier level, watts

P_t = Transmitter power output, watts

G_t = Transmitter antenna gain, numerical ratio

G_r = Receiving antenna gain, numerical ratio

L_t, L_r = Transmitting and receiving system losses, numerical ratio

= Carrier wavelength, meters

R = Propagation range, meters

$\frac{\lambda^2}{(4\pi R)^2}$ = Free space loss

N_o = Receiver noise density, watts/Hz

N_o = kT_s

k = Boltzmann's constant, 1.38×10^{-23} joule/(deg Kelvin)

T_s = System noise temperature, degrees Kelvin

T_s = $T_{antenna} + 290\ (L_r - 1) + T_{receiver}$

Utilizing the range equation, propagation margins were calculated for each Tug communication link as summarized in Tables 3.4-3 through 3.4-5.

Table 3.4-3. Telemetry Link Analysis Summary

Performance Summary	
Data Type	Telemetry (Tug)
Carrier Frequency	2.27 GHz
Data Rate	52 Kbps
Data Quality ($BER/E_B/N$)	10^{-3}/6.8 dB
Range	41,752 (22,500 nmi.)
Modulation @ Baseband: PCM @ RF: PSK	
Link Summary	
Transmit Power	10 dBw
Transmit Antenna Gain	-8 dB
Polarization Loss	-1 dB
Free Space Loss	-192 dB
Receive Antenna Gain	52 dB
Receive System Loss	-0.5 dB
Carrier Level @ Receiver Input	-139.5 dB
Noise Power (N_oR_B @ T_s = 55K)	-164 dB
Received E_B/N	24.5 dB
Detector Degradation	-2 dB
Output C/N	22.5 dB
Margin	15.7 dB

Table 3.4-4. Command Link Analysis Summary

Performance Summary	
Data Type	Command (Tug)
Carrier Frequency	2.1 GHz
Data Rate	10^3 bps
Data Quality ($BER/E_B/N$)	10^{-5}/10.2 dB
Range	41,752 Km (22,500 nmi.)
Modulation @ Baseband: PCM @ RF:PM	
Link Summary	
Transmit Power	56.0 dBw
Transmit Antenna Gain	53.0 dB
Transmit Pointing Loss	-0.5 dB
Polarization Loss	-1.0 dB
Free Space Loss	-192 dB
Receive Antenna Gain	-8.0 dB
Receive System Loss	-8.0 dB
Carrier Level @ Receiver Input	-100.5 dBw
Noise Power (N_oR_B @ T_s = 5500 K)	-143.4 dBw
Received E_B/N	42.9 dB
Detector Degradation	-2.0 dB
Output C/N	40.9 dB
Margin	30.7 dB

Table 3.4-5. Television Link Analysis Summary

Performance Summary	
Data Type	Television
Information Bandwidth	500 KHz
RF Bandwidth (Mod Index = 2)	3.0 MHz
Carrier Frequency	2280 MHz
Data Quality (C/N)	8.0 dB
Range	41,752 Km (22,500 nmi.)
Modulation: @ Baseband: AM @ RF: FM	
Link Summary	
Transmit Power	19.5 dBw
Transmit Antenna Gain	-10.0 dB
Polarization Loss	-1.0 dB
Free Space Loss	-192 dB
Receive Antenna Gain	53 dB
Receive System Loss	-0.5 dB
Carrier Level @ Receiver Input (C)	-131 dBw
Receiver System Noise Power (N) (T_s = 55K)	-146.4 dBw
C/N @ Detector Input	15.4 dB
Link Margin	7.4 dB
Wideband Improvement	15.5 dB
Output S/N	30.9 dB

3.4.4 SUBSYSTEM OPERATION

Subsystem Functions

The baseline Communications Subsystem provides the communications link functions described below. Table 3.4.6 denote the functional requirements during each mission phase.

Telemetry: The telemetry link provides the means to transmit vehicle system status from the vehicle to the ground for vehicle system performance evaluation. This capability must be maintained during each mission phase. In addition, limited Tug telemetry is made available to the EOS during EOS/Tug rendezvous and docking operations for verification of EOS commands.

Command and Up-Link Data: The command and up-link data link permits the transmission of data from the ground and other control elements, such as the EOS, to the Tug in the form of discrete commands, system contact program updates, and guidance parameter updates. Up-data link capability must be provided for all mission phases.

Tracking and Ranging: The tracking and ranging link provides the capability to receive and retransmit a ground tracking signal for the purpose of periodic ground generation of vehicle ephemeris data. The tracking and ranging link may be activated at any time during ground contact periods.

Television: The television link provides the capability to transmit imagery data from the vehicle to the ground. These data are used for ground observation and control during Tug/Payload rendezvous and docking operations.

System Design

The Space Tug Communications Subsystem shown in Figure 3.4-1 has the capability to transmit and receive all RF information necessary to accomplish the basic Tug mission requirements as described in the Tug Point Design Study criteria document. The function of the Communications Subsystem is to provide the basic transfer of intelligence between the Tug, external interfaces (Figure 3.4-2) and internal interfaces. External interfaces are those which are external to the Tug and are normally associated with RF Link (Figure 3.4-4) between the Tug and the MSFN or the EOS. Internal communication interfaces are those links whithin the Tug, and provide digital data transfer within the space vehicle.

An integrated equipment approach has been used to accomplish the communications functional requirements of telemetry, command and uplink data, tracking and ranging (MSFN), ranging (EOS rendezvous) and television transmission (payload docking).

Table 3.4-6 Communications Functional Requirements

COMM. FUNCTION \ MISSION PHASE	CHECKOUT & LAUNCH OPERATIONS	LAUNCH TO ORBIT	TUG DEPLOYMENT	ON-ORBIT OPERATIONS TUG MISSION	TUG RECOVERY	QUIESCENT	RE-ENTRY LANDING
Command & Uplink Data	X		X	X	X	X	
Telemetry	X		X	X	X		
Tracking & Ranging	X			X			
Television	X			X			
Rendezvous Interrogation*	X		X		X		
Rendezvous Response**	X		X		X		

*Tug/Payload Rendezvous

** Shuttle/Tug Rendezvous

Notes: 1. Command and Uplink Data includes: Commands, computer up-dating & reprogramming.

2. Telemetry includes: Experimental data, vehicle status, computer memory and command verification, G&N, status, & television.

3. Tracking & Ranging includes: Range and range-rate data.

4. Television includes: Imagery data for real-time monitoring of rendezvous and docking maneuvers.

5. Rendezvous Interrogation includes: Radar laser signal for detecting & tracking payload.

6. Rendezvous Response includes: Reception and retransmission of EOS ranging interrogation signal during Tug/EOS rendezvous.

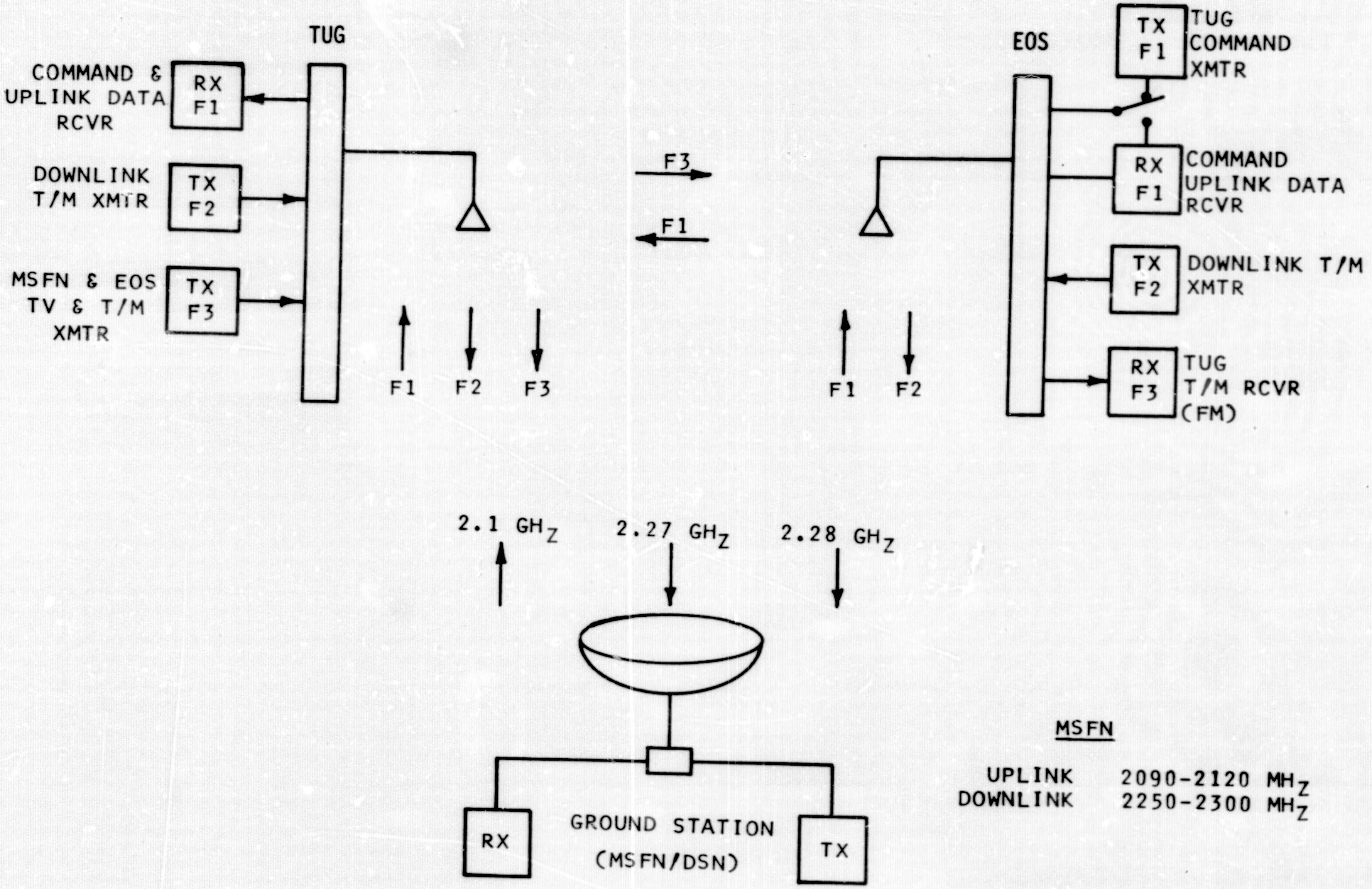


Figure 3.4-4 Communications Subsystem RF Link Interfaces

Telemetry

The Tug will utilize pulse code modulation techniques at S-band frequencies for the transmission of critical spacecraft parameters to MSFN ground facilities. A PCM serial bit stream (52 kbps) from the DMS is applied to the bi-phase modulator. The PCM data is modulated on an internally generated 1.024 MHz subcarrier. The resultant modulated signal is then routed to the transponder for phase modulation onto the down link carrier for subsequent transmission to the MSFN via the two omnidirectional S-band antennas. The transponder transmitter also transmits the turn-around PRN ranging code along with the PCM subcarrier. Both signals are phase modulated directly on the downlink carrier as shown in Figure 3.4-3B.

A limited amount of telemetry data will be transmitted via the FM transmitter to a manned rendezvous vehicle to indicate Tug status prior to docking operations.

Tracking, Ranging, and Commands

MSFN tracking and ranging of the Tug will be accomplished utilizing S-band communications. MSFN simultaneously transmits PRN ranging and uplink data on a 2.1 GHz carrier as shown in Figure 3.4-3A.

Serial binary command words are phase modulated on a subcarrier and linearly mixed with a pseudo-random noise (PRN). The composite signal is then phase modulated onto the up-link carrier. The up-data is received by the Tug's two S-band omni-antennas and is presented to the S-band transponder via the hybrid junction and isolation filter. The S-band transponder phase-lock receiver receives the uplink signal ,demodulates the ranging code, multiplies the signal by a known ratio, and coherently remodulates the signal on a downlink carrier (2.28 MHz), Figure 3.4-3B, as a phase coherent signal with a fixed delay for PRN ranging. Vehicle range rate will be measured employing the double-doppler method, and vehicle ranging by the use of pseudo-random noise techniques.

Command and Uplink Data

The demodulated 30 KC uplink subcarrier from the transponder receiver is routed to the command decoder. The decoder detects the command word bits and subsequently decodes, verifies, and distributes command words to the DMS for processing. In addition, the decoder is designed to provide a limited capability for commands independent of the DMS. Uplink information is in the form of discrete commands, system control program updates, and guidance parameter updates.

On-Board Television

The on-board television system (250 lines, 15 frames/sec.) will be utilized during rendezvous and docking operation with the Tug payload. Prior to commencing the docking operations, the television system will be activated by the on-board computer. During docking operations, the television system will provide a composite video signal to the S-band FM transmitter. The FM transmitter provides the wideband (3 MHz) transmission capability required for video transmission. The video signal is frequency modulated directly on an FM carrier for subsequent transmission to the MSFN.

3.4.5 COMPONENT CHARACTERISTICS

This section presents the specification sheets and component list for the Communications Subsystem design hardware selection. Overall component selection for the Tug Communications Subsystem design will primarily based on the latest technological design concepts and proposed future developments for the mid-1970's. Preliminary component selection was also based on the latest flight qualified equipment available on the market to-date with special emphasis placed on minimum weight requirements during the overall selection. The system design is dependent on two advanced technological components which are presently under development, they are: (1) the proposed Watkins-Johnson S-band 100 watt power amplifier primarily selected to insure an adequate link margin for reliable down-link television reception (the amplifier is a fully integrated package complete with traveling wave tube and featuring small size, light weight, and maximum overall efficiency), and (2) the Motorola Inc. S-band transponder presently under development for the Earth Resources Technology Satellite (ERTS) program. The transponder design is a combination of the Apollo block II design and the SGLS transponder design. The resulting transponder will be an up-to-date unit in reduced size and weight having the advantages of proven circuitry and package design. These two items are forecasted to be space-rated and in production by the mid-1970's.

The physical and performance characteristics of the individual subsystem components are described in the tables defined below:

Component	Table
PM Transponder	3.4-7
Power Amplifier	3.4-8
FM Transmitter	3.4-9
Bi-Phase Modulator	3.4-10
Decoder	3.4-11
S-Band Antenna	3.4-12
RF Switch	3.4-13
RF Multiplexer	3.4-14
Isolation Filter	3.4-15
Hybrid Junction	3.4-16
Directional Coupler	3.4-17

Table 3.4-7. PM Transponder Characteristics Summary

BASELINE SELECTION: MOTOROLA, INC., ERT TRANSPONDER (PRESENTLY UNDER DEVELOPMENT)	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	24 lb.
Size	13 x 6 x 8 in.
Heat Dissipation	35 watts
Operating Voltage	28 $\pm$4 VDC
Operating Current	1.25 amperes @ 28 VDC
Operating Thermal Limits	+32°F to +118°F
Non-Operating Thermal Limits	-45°F to +150°F
Sine Vibration	10 g's
Random Vibration	20 g's RMS (qual. level)
Humidity	Unlimited (unit sealed)
Altitude	Unlimited (unit sealed)
Installation	Standard avionic equipment mounting
Reliability MTBF	40,000 hours

Table 3.4-7. PM Transponder Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS	
Parameter	Performance
PM Transmitter	
Transmit Frequency	2200-2300 MHz
RF Power Output	1 watt min. (at antenna terminals)
Modulation Frequency Range, Wideband	PM: 300 Hz to 3.0 MHz
Peak Deviation Capability	± 4 radians
Deviation Sensitivity	1 radian/volt
Deviation Linearity	10%/2 radians
Receiver	
Receiver Frequency	2020-2120 MHz
Tracking Range	± 90 KHz
Predetection Bandwidth	15 KHz
Threshold Noise Bandwidth	$2\ B_{LO} = 800$ Hz
Strong Signal Noise Bandwidth	$2\ B_{LSS} = 3000$ Hz
Noise Figure (at receiver input)	8 DB
Noise Figure (at antenna terminal)	13 DB
Carrier Tracking Sensitivity (at diplexer input)	-46 to -126 DBM
AGC Time Constant	Approx. 5 msec.

Table 3.4-8. Power Amplifier Characteristics Summary

BASELINE SELECTION: WATKINS-JOHNSON CO., MODEL WJ-1014-1, P/N 50M12673 (PRESENTLY UNDER DEVELOPMENT)	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	19 lb.
Size	12.5 x 6.6 x 5.6 in.
Heat Dissipation	200 watts
Operating Voltage	28 $\pm$4 VDC
Operating Current	7.1 amperes @ 28 VDC
Operating Thermal Limits	-22°F to $+176^{\circ}$F
Filament Time Delay Before Carrier	90 to 150 secs.
Installation	Standard avionic equipment mounting
Reliability MTBF	20,000 hours

Table 3.4-8. Power Amplifier Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Frequency Range	2.2 to 2.4 GHz
Power Output, Saturated	90 watts min.
Efficiency	
Overall Amplifier	20%, min.
Traveling-wave Tube	30%, min.
Power Supply	80%, min.
Power Output Variation	
DB per 100 MHz	± 0.4
DB per 10 MHz	± 0.1
RF Drive for Sat. Output	100 mw. max.
Maximum Load VSWR	Infinity, any phase
Duty Cycle	CW
Phase Linearity	4°/10 MHz BW
Harmonic and Spurious Outputs	
Spurious Coherent	65 DB
In-band Spurious	60 DB
2nd & 3rd Harmonic	60 DB
4th Harmonic	80 DB
Noise Power Output (2.2-10 GHz)	65 DB/MHz BW
On-Off Cycling	10,000, min.

Table 3.4-9. FM Transmitter Characteristics Summary

BASELINE SECTION: MOTOROLA, INC., MTT-501, S-BAND	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	2.3 lb.
Size	4.85 x 2.9 x 3 in.
Heat Dissipation	60 watts
Operating Voltage	28 ± 4 VDC
Operating Current	2.2 amperes @ 28 VDC
Operating Thermal Limits	$-22^{\circ}F$ to $+176^{\circ}F$
Installation	Standard avionic equipment mounting
Acceleration	50 g's, three axes
Vibration	25 g's rms random
Shock	100 g's, 11 milliseconds, survives 2500 g's 0.5 millisecond
Humidity	Unlimited (sealed unit)
Operating Altitude	Unlimited (sealed unit)
Reliability MTBF	50,000 hours

Table 3.4-9. FM Transmitter Characteristics Summary (Cont)

PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Output	
Frequency Range	2200 to 2300 MHz
Frequency Error	±0.003% maximum (includes tolerance and stability)
Output Power	7-watts nominal, 5-watts minimum
Load Impedance	50 ohms: rated power developed with up to 2:1 VSWR; open or short will not damage transmitter.
Antenna Conducted Spurious Outputs	55 +10 Log Pt db below the unmod. carrier
RF Intermodulation Products	40 db below the carrier when an RF in-channel signal 15 db below the carrier is applied to the antenna.
Modulation	
Type	True FM
Frequency Response	DC to 1.0 MHz, ±1 db
Peak Deviation	±500 KHz
Deviation Sensitivity	Up to 500 KHz per volt (optional)
Deviation Sensitivity Stability	±10%
Input Impedance	10K to 50K ohm dc resistance shorted by 30 pf maximum
Distortion	(a) Intermodulation products of two equal test tones adjusted to produce ±500 KHz total peak deviation are at least 34 db below either test tone. (b) Total harmonic distortion of any frequency from dc to 1 MHz adjusted for ±500 KHz peak deviation is no greater than 2%. (c) Pulse response exhibits a 10-90% rise time of 1 microsecond maximum, less than 5% overshoot and less than 5% tilt.

Table 3.4-10. Bi-Phase Modulator Characteristics Summary

BASELINE SELECTION: MOTOROLA, INC. (UNDER DEVELOPMENT)	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	10 ounces
Size	5 x 3 x 0.7 in.
Heat Dissipation	5 watts peak, 3 watts standby
Operating Voltage	28 $\pm$4 VDC
Operating Current	1.8 amperes @ 28 VDC
Operating Thermal Limits	-22°F to $+176^{\circ}$F
Installation	Standard avionic equipment mounting
Reliability MTBF	50,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
PCM Telemetry Interface (Input)	
Signal Type	NRZ
Bit Rate	1 to 256 Kbps maximum
Amplitude	"0" = 0.3V, "1" = 4.5V (nominal)
Rise and Fall Times	2.5% of bit length
Input Impedance	1000 ohms nominal, 100 picofarads nominal
Source Impedance	300 ohms nominal, 1000 picofarads nominal
Composite Baseband Interface (Output)	
Signal Type	Modulator subcarrier
Subcarrier Frequency	1.024 MHz $\pm$0.002% stability $\pm$0.005%
Modulation	Bi-phase $\pm$90 degrees
Bandwidth	270 KHz (3 db bw)
Data	PCM, NRZ, 256 Kbps maximum
Signal Output Level	2 to 10 volts peak-to-peak (adjustable)
Load Impedance	5000 ohms (nominal) 250 picofarads (nominal)

Table 3.4-11. Decoder Characteristics Summary

BASELINE SELECTION: MOTOROLA, INC., MODEL UDD-94990	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	1.5 lb.
Size	5 x 4 x 2 in.
Heat Dissipation	2 watts
Operating Voltage	28 $\pm$4 VDC
Operating Current	70 ma @ 28 VDC
Operating Thermal Limits	-4°F to $+176^{\circ}$F
Altitude	Unlimited
Humidity	Zero to 100%
Installation	Standard avionic equipment mounting
Reliability MTBF	50,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Operating Frequency	Works in conjunction with S-band receiver of transponder
Data Rate	1000 bits/sec

Table 3.4-12. S-Band Antenna Characteristic Summary

BASELINE SELECTION: AMECON DIV., LITTON SYSTEM, INC., P/N MF481-0048	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	2.5 lb.
Size	4.5 x 4.5 x 1.5 in.
Operating Thermal Limits	-125°F to +400°F
Vibration (sine)	(3 mutually perpendicular axes)
5 to 27.5 cps	Plus or minus 1.56 g.
27 to 52 cps	0.43 inch double amplitude
52 to 500 cps	Plus or minus 6.0 g's
Shock	30 g for 11 ±1 milliseconds, in each of 3 mutually perpendicular axes.
Acceleration	7 g's for 147 seconds, 20 g's for 120 seconds
Installation	Flush mounted to external surface
Reliability MTBF	300,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Frequency	2100 to 2300 MHz
Polarization	Right-hand circular polarization (RCP)
Impedance	50 ohms nominal
VSWR	Not greater than 1.5:1

Table 3.4-13. RF Switch Characteristic Summary

BASELINE SELECTION: TRANSCO PRODUCTS, INC., P/N 15610, TYPE W, SPDT, LATCHING	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	8 ounces
Size	2.45 x 2.65 x 1.09 in.
Heat Dissipation	15 watts for 150 ms per actuation
Operating Voltage	28 ±4 VDC
Operating Current	0.54 amperes for 150 ms per switch actuation
Operating Time	10 ms
Contacts	Make-before-break
Operating Thermal Limits	-65°F to +250°F
Vibration	15 g minimum
Latching	
Actuator	
Voltage	28 VDC
Power	3 amps peak
Life	100,000 cycles
Installation	Standard avionic equipment mounting
Reliability MTBF	250,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Impedance	50 ohms nominal
Frequency	7 GC
VSWR	1.25 at 4 GHz
Insertion Loss	0.25 db at 4 GHz
Crosstalk	30 db at 4 GHz, 45 db at 2 GHz

Table 3.4-14. RF Multiplexer Characteristics Summary

BASELINE SELECTION: WAVECOM, INC. - MODEL 217, S-BAND	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	5 ounces
Size	1.2 x 2 x 4 in.
Power Rating	10 watts CW/Chan
Operating Thermal Limits	-65° to +160°F
Installation	Standard avionic equipment mounting
Reliability MTBF	250,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Frequency Band	2200-2300 MHz
Passband BW	5 MHz max.
Passband Ins. Loss	1.3:1 max.
Interchannel Isolation	
(at fo), 25 MHz Sep.	20 DB min.
30 MHz Sep.	25 DB min.
40 MHz Sep.	33 DB min.
50 MHz Sep.	39 DB min.
60 MHz Sep.	44 DB min.
Harmonic Rejection	60 DB min. to 10 GHz
Passband Time Delay Vatiation	±5 nsec.

Table 3.4-15. Isolation Filter Characteristics Summary

BASELINE SELECTION: WAVECOM, INC., P/N S-101, S-BAND	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	0.35 lb. (max.)
Size	7 x 2 x 1.5 in.
Power Rating	40 watts
Operating Thermal Limits	-65°F to +200°F
Installation	Standard avionic equipment mounting
Material	Aluminum
Altitude	Unlimited (sealed)
Shock	50 g's (8 millisec duration) 200 g's (1 millisec. duration)
Acceleration	50 g's
Vibration	
Sinusoidal	5-14 cps - 5% double amplitude 14-400 cps - 10 g zero to peak 400-2000 cps - 20 g zero to peak
Random	20-400 cps - .08 g^2/cps 400-2000 cps - .20 g^2/cps
Reliability MTBF	200,000 hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Frequency	2200-2300 MHz
Passband Bandwidth	5.0 MHz min.
Passband Insertion Loss	0.8 db max.
Passband VSWR	1.30:1 max.
Interchannel Isolation	
25 MHz Separation	20 db min.
30 MHz Separation	25 db min.
40 MHz Separation	33 db min.
50 MHz Separation	39 db min.
60 MHz Separation	44 db min.
Power Handling	40 watts CW/channel
Harmonic Rejection (thru 3rd Harmonic)	60 db min.
Time Delay Variation	±5 nsec

Table 3.4-16. Hybrid Junction Characteristics Summary

BASELINE SELECTION: TRANSCO PRODUCTS, INC., P/N 40020, S-BAND	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	1 lb. (max.)
Size	2.45 x 2.98 x 1.87 in.
Operating Thermal Limits	-65°F to +250°F
Installation	Standard avionic equipment mounting
Pressure	Hermetically sealed unit
Reliability MTBF	200,000
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Frequency	2900-3260 MHz
VSWR	1.20 max., arm 2 over frequency range
Bandwidth	over 10%
Insertion Loss	3.2 db max., arms 2 to 1 and 2 to 3
Isolation	20 db over frequency range, arm 1 to 3 and 2 to 4

Table 3.4-17. Directional Coupler Characteristics Summary

BASELINE SELECTION: MICROLAB/FXR, PART NO. CB-68N	
PHYSICAL CHARACTERISTICS	
Item	Characteristic
Weight	7.3 oz.
Size	2.5 x 4.12 x 0.75 in.
Operating Thermal Limits	-65°F to +212°F
Installation	Standard avionic equipment mounting
Reliability MTBF	500,000 Hours
PERFORMANCE CHARACTERISTICS	
Parameter	Performance
Coupling Accuracy	±1 DB
Maximum Insertion Loss*	0.2 DB
Impedance	50 ohms
Power Rating, Average	100 watts
Power Rating, Peak	3 kilowatts
*Above coupling loss	

3.4.6 Alternate Design Approach

An alternate design approach in the Communications Subsystem design would be the use of individual solid state amplifiers for the telemetry and television links instead of the baseline configuration of a single wideband TWT amplifier.

The Microwave Semiconductor Corp. (MSC) has recently introduced the 91,000 series of all-transistorized MIC power amplifiers delivering up to 30 watts at 2.2-2.3 GHz in an attempt to satisfy requirements for high power S-band amplifiers in the telemetry industry. The solid state devices hold promise of improved reliability in space vehicle applications where long operating life is necessary. The latest advances in transistor and microwave integrated circuits techniques have been used by MSC to develop the new line of S-band amplifiers.

The advantages of the alternate design approach are improved reliability, greater overall input power efficiency and a reduction in system weight of approximately 12 pounds.

The disadvantage of the 91,000 solid state amplifiers is the maximum output power limitation. The Tug baseline Communications Subsystem presently requires an amplifier output power of 100 watts obtainable by utilizing a hybrid design of solid state circuitry and a traveling-wave output tube. The total output power is shared by both the telemetry and television links. The MSC solid state amplifier is presently limited to 30 watts, but it is anticipated that greater output power levels will be available within the operational time period of the Tug.

3.5 INSTRUMENTATION

The Instrumentation Subsystem is composed of transducers and signal conditioning equipment in the variety and quantity necessary to satisfy the vehicle systems measurement requirements as identified in Table 3.5-1. To obtain high accuracy and reliability in the measurement of the most numerous parameters, strain gage pressure transducers and platinum wire resistance temperature transducers are used. Dedicated, remotely located signal conditioning modules provide sensor excitation and output amplification to a 0-5 VDC level for DMS input compatibility. Measurement of position parameters is made with potentiometric transducers excited to provide a 0-5 VDC output. Voltage and current measurements are made using a sensor-electronic package combination to provide a 0-5 VDC output. The measurement of fluid flow is made using a turbine type flowmeter that provides a pulse output at a level compatible with the DMS input requirements. The liquid level point sensors provide a discrete output to the DMS.

No multiplexing of measurement channels is performed in the Instrumentation Subsystem. All channels are individually routed to the DMS DAU's where multiplexing is accomplished as a normal subsystem function.

For the measurement of main engine parameters (not identified in Table 3.5-1), the main engine control package provides self contained signal

Table 3.5-1. Instrumentation Measurement List

Measurement Title	Range	Transducer; Sig. Conditioner
THRUST VECTOR CONTROL SYSTEM		
Reservoir Oil Temp	M65-300°F	Hi Level Probe; Reg ±28 VDC Pwr Supply
Cold Spot Oil Temp	M100-300°F	Hi Level Probe; Reg ±28 VDC Pwr Supply
Accumulator Oil Press	0-3500 PSIA	Potentiometer; Reg 5 VDC Power Supply
Accumulator Gas Press	0-3500 PSIA	Potentiometer; Reg 5 VDC Power Supply
Reservoir Piston Position	0-3 Inches	Potentiometer; Reg 5 VDC Power Supply
Pitch Actuator Piston Position	±1.5 Inches	Potentiometer; Reg 5 VDC Power Supply
YAW Actuator Piston Position	±1.5 Inches	Potentiometer; Reg 5 VDC Power Supply
FUEL CELL AND THERMAL CONTROL SYSTEM		
Radiator #1 Coolant Inlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #1 Coolant Outlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #2 Coolant Inlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #2 Coolant Outlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #3 Coolant Inlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #3 Coolant Outlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #4 Coolant Inlet Temp	M150-200°F	Plat. Probe; Amplifier
Radiator #4 Coolant Output Temp	M150-200°F	Plat. Probe; Amplifier
Condensor Subcooler Inlet Temp	M150-200°F	Plat. Probe; Amplifier
Condensor Subcooler Outlet Temp	M150-200°F	Plat. Probe; Amplifier
Freon Pump Inlet Press	0-150 PSIA	SG Xducer; Amplifier
Freon Pump Outlet Press	0-200 PSIA	SG Xducer; Amplifier
Freon Flow Rate	0-300 PPH	Flowmeter
POWER GENERATION SYSTEM		
Safing Battery Temp	0-160°F	Plat. Surf. Xducer; Amplifier
Safing Battery Voltage	26-32 VDC	N/A; Attenuator
Safing Battery Current	TBD	Current Sensor; Current Sensor Electronics
Fuel Cell Stack Voltage	26-32 VDC	N/A; Attenuator
Fuel Cell Stack Current	TBD	Current Sensor; Current Sensor Electronics
Main Power Distributor Voltage	26-32 VDC	N/A; Attenuator
Main Power Distribution Current	TBD	Current Sensor; Current Sensor Electronics

Table 3.5-1. Instrumentation Measurement List (Cont)

Measurement Title	Range	Transducer; Sig. Conditioner
PRESSURIZATION SYSTEM		
LOX Tank Internal Temp	M350-200°F	Plat. Probe; Amplifier
LOX Tank Internal Temp	M350-200°F	Plat. Probe; Amplifier
LOX Tank Internal Temp	M350-200°F	Plat. Probe; Amplifier
LOX Tank Internal Temp	M350-200°F	Plat. Probe; Amplifier
APS LOX Tank Internal Temp	M350-200°F	Plat. Probe; Amplifier
Main Engine GOX Pressurization Supply Temp	M200-200°F	Plat. Probe; Amplifier
LH_2 Tank Internal Temp	M425-M200°F	Plat. Probe; Amplifier
LH_2 Tank Internal Temp	M425-M200°F	Plat. Probe; Amplifier
LH_2 Tank Internal Temp	M425-M200°F	Plat. Probe; Amplifier
LH_2 Tank Internal Temp	M425-M200°F	Plat. Probe; Amplifier
APS LH_2 Tank Internal Temp	M425-M200°F	Plat. Probe; Amplifier
Main Engine GH_2 Pressurization Supply Temp	M360-M60°F	Plat. Probe; Amplifier
LOX Tank Ullage Press	0-35 PSIA	SG Xducer; Amplifier
APS LOX Tank Press	0-35 PSIA	SG Xducer; Amplifier
LOX Pressurization Reg Out Press	0-35 PSIA	SG Xducer; Amplifier
Main Eng GOX Pressurization Supply Press	0-4000 PSIA	SG Xducer; Amplifier
LH_2 Tank Ullage Press	0-35 PSIA	SG Xducer; Amplifier
APS LH_2 Tank Press	0-35 PSIA	SG Xducer; Amplifier
LH_2 Pressurization Reg Out Press	0-35 PSIA	SG Xducer; Amplifier
Main Eng GH_2 Pressurization Supply Press	0-3000 PSIA	SG Xducer; Amplifier
LOX Pressurization Reg Stage #1 Position	0-100 PCT	Potentiometer; Reg 5 VDC Pwr Supply
LOX Pressurization Reg Stage #2 Position	0-100 PCT	Potentiometer; Reg 5 VDC Pwr Supply
LH_2 Pressurization Reg Stage #1 Position	0-100 PCT	Potentiometer; Reg 5 VDC Pwr Supply
LH_2 Pressurization Reg Stage #2 Position	0-100 PCT	Potentiometer; Reg 5 VDC Pwr Supply
SYSTEM		
LH_2 Turbopump GG Injector Temp	0-2500°F	Plat. Probe; Amplifier
Flow Controller GOX Supply Temp	M100-0°F	Plat. Probe; Amplifier
Flow Controller GH_2 Supply Temp	M300-M200°F	Plat. Probe; Amplifier
LOX Turbopump GG Injector Temp	0-2500°F	Plat. Probe; Amplifier
GH_2 Accumulator Temp	M350-M150°F	Plat. Probe; Amplifier
GOX Accumulator Temp	M150-30°F	Plat. Probe; Amplifier

Table 3.5-1. Instrumentation Measurement List (Cont)

Measurement Title	Range	Transducer; Sig. Conditioner
SYSTEM (Cont)		
LH_2 Turbopump GC Injector Press	0-350 PSIA	SG Xducer; Amplifier
LH_2 Pump/GH_2 Accumulator Delta Press	0-400 PSID	SG Xducer; Amplifier
LH_2 Pump Output Press	0-1500 PSIA	SG Xducer; Amplifier
Flow Controller GOX Supply Press	0-600 PSIA	SG Xducer; Amplifier
Flow Controller GH_2 Supply Press	0-600 PSIA	SG Xducer; Amplifier
GH_2 Heat Exchanger GG Inject Press	0-350 PSIA	SG Xducer; Amplifier
GOX Heat Exchanger GG Inject Press	0-350 PSIA	SG Xducer; Amplifier
LOX Turbopump GG Injector Press	0-350 PSIA	SG Xducer; Amplifier
LOX Pump/GOX Accumulator Delta Press	0-400 PSID	SG Xducer; Amplifier
LOX Pump Output Press	0-1500 PSIA	SG Xducer; Amplifier
GH_2 Accumulator Press	0-2000 PSIA	SG Xducer; Amplifier
GOX Accumulator Press	0-2000 PSIA	SG Xducer; Amplifier
GH_2 Inter Reg Press	0-1000 PSIA	SG Xducer; Amplifier
GH_2 Inter Reg Press	0-1000 PSIA	SG Xducer; Amplifier
GOX Inter Reg Press	0-1000 PSIA	SG Xducer; Amplifier
GOX Inter Reg Press	0-1000 PSIA	SG Xducer; Amplifier
GH_2 Reg Output Press	0-1000 PSIA	SG Xducer; Amplifier
GOX Reg Output Press	0-1000 PSIA	SG Xducer; Amplifier
Thruster #1 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #2 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #3 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #4 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #5 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #6 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #7 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #8 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #9 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #10 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #11 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #12 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #13 Injector Press	0-350 PSIA	SG Xducer; Amplifier
Thruster #14 Injector Press	0-350 PSIA	SG Xducer; Amplifier

Table 3.5-1. Instrumentation Measurement List (Cont)

SAFING/VENT SYSTEM

Measurement Title	Range	Transducer; Sig. Conditioner
LH_2 Shuttle Valve Inlet Temp	M425-200°F	Plat. Probe; Amplifier
LH_2 Relief Valve Outlet Temp	M425-200°F	Plat. Probe; Amplifier
LH_2 Orbital Vent Valve Inlet Temp	M425-200°F	Plat. Probe; Amplifier
LOX Shuttle Valve Inlet Temp	M360-200°F	Plat. Probe; Amplifier
LOX Relief Valve Outlet Temp	M360-200°F	Plat. Probe; Amplifier
LOX Orbital Vent Valve Inlet Temp	M360-200°F	Plat. Probe; Amplifier
LH_2 Vent Valve Outlet Temp	M425-200°F	Plat. Probe; Amplifier
LH_2 Fill/Drain Valve Outlet Temp	M425-200°F	Plat. Probe; Amplifier
LOX Vent Valve Outlet Temp	M360-200°F	Plat. Probe; Amplifier
LOX Fill/Drain Valve Outlet Temp	M360-200°F	Plat. Probe; Amplifier
Insulation Purge Gas Temp	M100-250°F	Plat. Probe; Amplifier
LH_2 Fill/Drain Valve Outlet Press	0-50 PSIA	SG Xducer; Amplifier
LOX Fill/Drain Valve Outlet Press	0-50 PSIA	SG Xducer; Amplifier
LOX He Sys Orifice Downstream Press	0-1800 PSIA	SG Xducer; Amplifier
LOX He Sys Orifice Upstream Press	0-3500 PSIA	SG Xducer; Amplifier
LH_2 He Sys Orifice Downstream Press	0-1800 PSIA	SG Xducer; Amplifier
LH_2 He Sys Orifice Upstream Press	0-3500 PSIA	SG Xducer; Amplifier
Insulation Purge Gas Press	0-75 PSIA	SG Xducer; Amplifier
Helium Receiver Bottle Press	0-3500 PSIA	SG Xducer; Amplifier

conditioning and pulse code modulation circuitry to perform internal processing that outputs data at the engine/stage interface in a serial digital wavetrain format. The output is routed to a DAU where the data becomes available for use by the DMS computer.

3.5.1 Requirements

A basic design requirement governing the Instrumentation Subsystem definition was that it be compatible with the DMS. The Instrumentation Subsystem as defined consists of the transducers and/or signal conditioners necessary to provide a 0-5 VDC data signal to the DMS for selected measurement parameters. Discrete measurements will be wired directly to the DMS from the valve, relay, or other device within the operating subsystem. The multiplexing of measurements to meet response characteristics of the measurements will be controlled by the DMS.

The instrumentation measurement requirements were established as a result of Electrical and Mechanical Subsystem design analysis. These requirements are identified in Table 3.5-1 with an abbreviated summary of quantities by subsystem shown in Table 3.5-2. The measurements defined in Table 3.5-1 and Table 3.5-2 reflect the Tug operational configuration and do not include the main engine measurements. The main engine measurements were assumed to be the responsibility of the engine manufacturer and as such the data from the engine will be made available to the DMS in a serial digital wavetrain format at the engine/stage interface.

For the purposes of this study, all discrete measurements are identified in Table 3.1-3. The direct nature of their interface with the data management data acquisition units prompted the selection of this data presentation approach.

3.5.2 Subsystem Operation

Functional Operation

The functional operation for each type of analog measurement listed in Table 3.5-1 is defined below with a functional block diagram:

Temperature

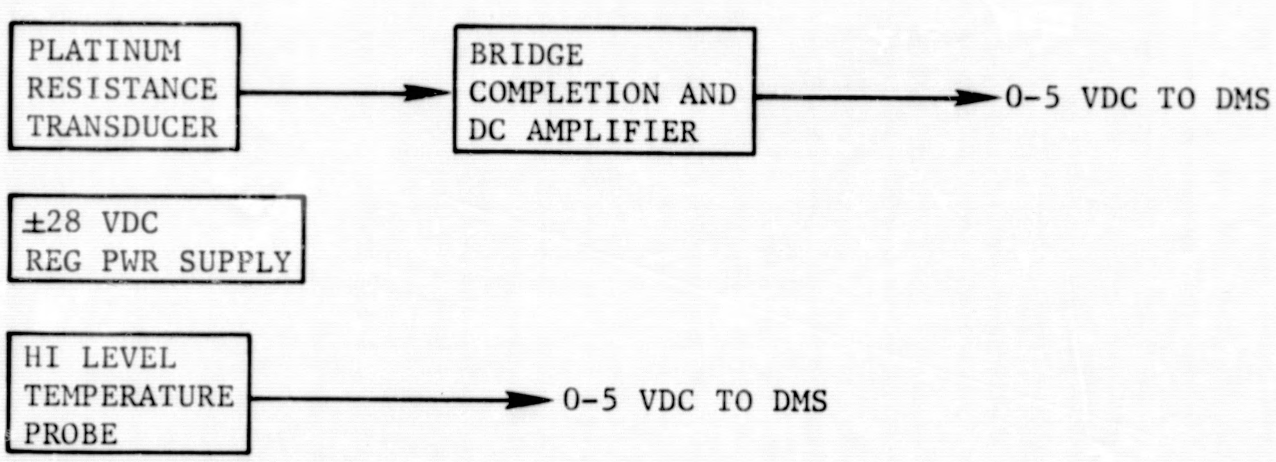


Table 3.5-2. Operational Measurement Summary

Subsystem**	PRESSURE	TEMPERATURE	POSITION	VOLTAGE	CURRENT	FLOW	POINT SENSORS*	TOTAL
THRUST VECTOR CONTROL	2	2	3	-	-	-	-	7
FUEL CELL AND THERMAL CONTROL SYSTEM	2	10	-	-	-	1	-	13
POWER GENERATION SYSTEM	-	1	-	3	3	-	-	7
FEED, FILL, DRAIN AND VENT SYSTEM	-	-	-	-	-	-	22	22
PRESSURIZATION SYSTEM	8	12	4	-	-	-	-	24
APS SYSTEM	32	6	-	-	-	-	-	38
SAFING/VENT SYSTEM	8	11	-	-	-	-	-	19
TOTAL	52	42	7	3	3	1	22	130

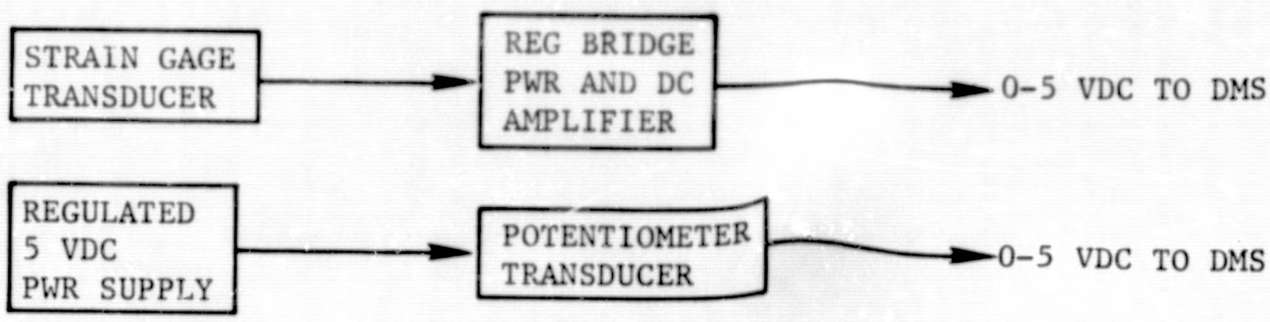
*Measurement titles are identified in Table 3.1-3

**Main engine measurements are made available to the DMS in serial digital format by the engine control package and are not identified in this summary.

Temperature measurements with the exception of TVC measurements will be made utilizing platinum resistance sensors. The sensor will change over a prescribed range. The sensor is the active leg of an unbalanced bridge circuit. The bridge completion and DC amplification necessary for a 0-5 VDC output to the DMS is integrated into a single package.

The two TVC temperature measurements will be made utilizing a high level temperature probe similar to the one utilized on the Saturn S-II. The probe is a half-bridge design using a regulated ±28 VDC power supply to provide a 0-5 VDC output over a specified temperature range.

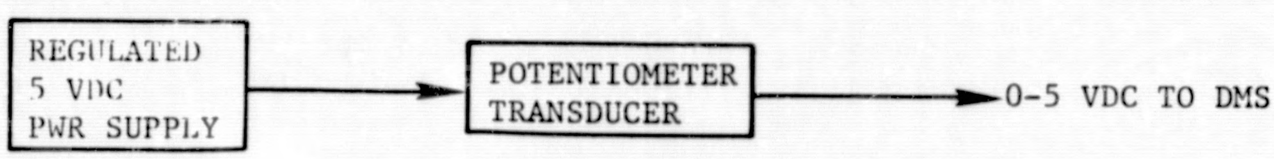
Pressure



The pressure measurements will be made utilizing two types of transducers; potentiometers and strain gages. The potentiometer is an electromechanical device containing a resistance element which is contacted by a movable slider. With a regulated 5 VDC across the resistance element, the movable slider provides a 0-5 VDC output that is a direct function of the pressure sensed by the transducer.

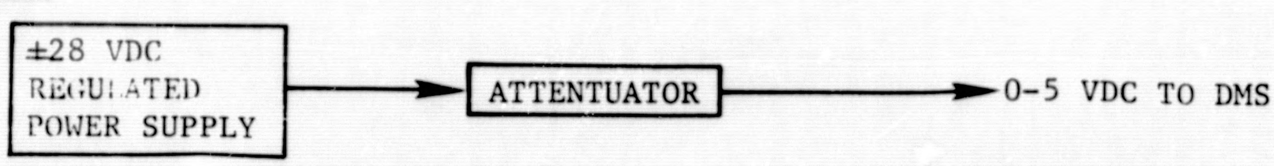
The strain gage sensors will be four active arm bridges, each arm changing length and consequently resistance as a result of the change in applied pressure to the transducer. Bridge voltage and output signal amplification will be integrated into a single package to provide a 0-5 VDC output signal to the DMS.

Position



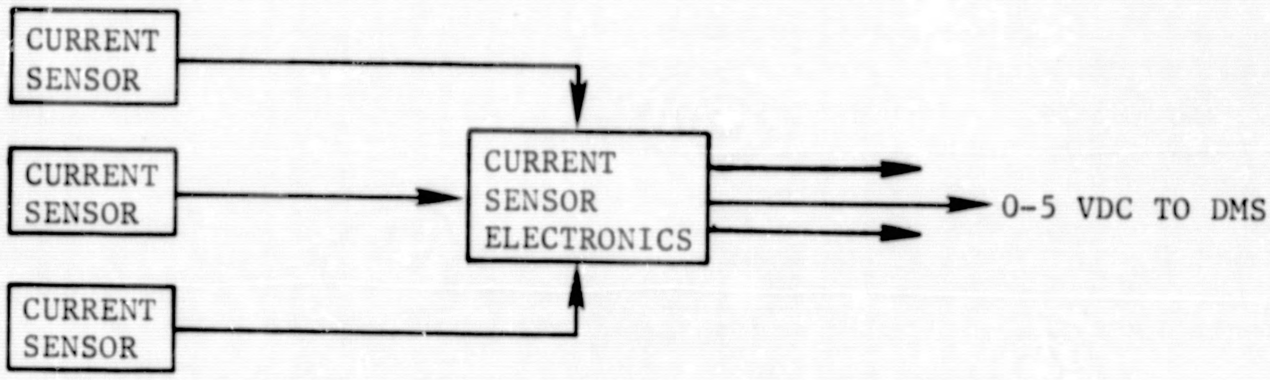
The position measurements will be made with potentiometers. The operation is the same as the pressure potentiometer except that the output is a function of valve or piston position.

Voltage



The attenuator is a network of zener diodes and resistors using a regulated ±28 VDC. Zero volts output is equivalent to 26 VDC and five volts output is equivalent to 32 VDC.

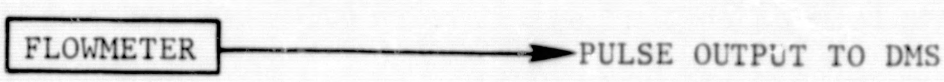
Current



The current monitor measurements will be made utilizing the present Saturn S-II system. A central electronics package and three sensors comprise the system. The sensors are flux linkage devices using the saturable reactor principle. The sensor constitutes one winding on a torroid core. The saturable reactor has another winding on the torroid that is driven by an AC source. The torroid is driven into the saturation region by the current in the bus, and this current change in the current carrying bus is reflected as impedance change in the secondary winding.

The current change as a result of this impedance change is full wave rectified and this DC current passing through an adjusted resistor provides a 0-5 VDC output to the DMS.

Flow



The flow measurements will be made using a turbine type non-magnetic pickup flowmeter. The pulse output will be generated by a non-magnetic cryotronic pickup.

Discrete (LOX and LH_2 Point Sensors)

The point sensors for monitoring cryogenic liquid level consists of a gold-plated platinum wire grid through which a minute constant current is maintained. The change in media causes an almost instantaneous change in resistance. This resistance change is used to operate a solid state switch. The control units output is zero when the sensor is wet and 27 volts when the sensor is dry.

Table 3.5-3 Component Characteristics

TRANSDUCERS

Characteristics	Pressure	Temperature	Current	Flowmeter	Pt Sensor
Unit Weight	Strain Gage - 2.5 oz. Pot - 10 oz.	8 oz.	8 oz.	16 oz.	1.4 oz.
Unit Size	SG - 1" dia x 2" Pot - 2"x3"x1-1/4"	3"x.25" dia.	3"x3"x3"	6-1/4"x2" dia	1" dia.x1.2"
Operating Voltage	SG - 10 VDC Pot - 5 VDC	N/A	24-32 VDC	N/A	6.3 VDC max.
Operating Current	SG - 30 MA Pot - Neg.	20 MA max.	300 MA max.	N/A	200 MA (LH_2) 100 MA (LO_X)
Operating Thermal Limitations	SG - M423-300°F Pot - M100-280°F	M450°F-630°F	M65°F-160°F	M452°F-460°F	M425°F-165°F
Non-Operating Thermal Limitations	SG-M423-300°F Pot - M100-280°F	M450°F-630°F	M65°F-160°F	M452°F-460°F	M425°F-160°F
Installation Requirements	SG - Boss Mounted Pot	Boss Mounted	Screw Mounted	In-Line Mounted	Screw Mounted

Typical Manufacturers

Transducer Type	Manufacturer	Model Number
Pressure	Bell and Howell (CEC) Statham Instruments	4-354-0132 PA822
Temperature	Rosemount Engr. Co. RDF Corp.	134FT 21005
Current	Pioneer Magnetics, Inc.	PM-1253
Flowmeter	Quantum Dynamics, Inc.	QFL(XX)-VWR-15C
Pt. Sensor	Acoustica Associates, Inc.	STS-505

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Table 3.5-4 Component Characteristics

SIGNAL CONDITIONERS

Characteristic	Press. Sig. Cond. Amp.	Temp. Sig. Cond. Amp.	5 VDC Reg. Pwr. Supply	Current Sensor Electronics	Attenuator	Pt. Sensor Electronics	$\pm$28 VDC Pwr Supply
Unit Weight	4 oz.	4 oz.	2.5 oz.	64 oz.	5 oz.	7.5 oz.	32 oz.
Unit Size	1"x2"x3"	1"x2"x3"	1"x1-1/2"x1-1/4"	6"x3"x3"	2"x2-1/2"x1/3"	2"x2"x2-1/2"	4"x3"x3"
Operating Voltage	28 $\pm$4 VDC	28 $\pm$4 VDC	28 $\pm$4 VDC	28 $\pm$4 VDC	18-35 VDC	28 $\pm$4 VDC	28 $\pm$4 VDC
Operating Current	25 MA	22 MA	250 MA	300 MA max	70 MA	LO_X 150 MA LH_2 250 MA	500 MA
Operating Thermal Limitations	M65°F-212°F	M65°F-212°F	M40°F-175°F	M65°F-160°F	0-150°F	M65°F-165°F	M65°F-160°F
Non-Operating Thermal Limitations	M65°F-212°F	M65°F-212°F	M40°F-175°F	M65°F-160°F	M65-150°F	M65°F-165°F	M65°F-160°F
Installation Requirements	Screw Mounted	Screw Mounted	Screw Mounted	Screw Mounted	Screw Mounted	Screw Mounted	Screw Mounted

Typical Manufacturers

Signal Conditioner	Manufacturer	Model Number
Press. Sig. Cond. Amp.	Rosemount Engr. Co.	510 BH
Temp. Sig. Cond. Amp.	Rosemount Engr. Co.	510 BH
5 VDC Reg. Pwr. Supply	K-West	870
Current Sensor Electronics	Pioneer Magnetics, Inc.	PM-1459
Attenuator	NR	V7-750407
Pt. Sensor Electronics	Acoustica Associates, Inc.	TCU 426-1 (LH_2) TCU 425-1 (LO_X)
$\pm$28 VDC Pwr. Supply	NR	V7-750324

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Table 3.5-5 Instrumentation Subsystem Power and Weight Summary

Subsystem Component(s)	Quantity	Power Required (watts)	Total Weight (oz.)
Strain Gage Transducer/Amplifier	50	35.0	325
High Level Temp. Probe	2	1.1	16
Plat. Transducer/Amplifier	40	24.6	480
Potentiometers	9	N/A	90
Current Sensor (3)/Electronics	1	8.4	64
Current Sensors	3	N/A	24
Pt Sensor/Controllers	22	123.2	196
Reg. 5 VDC Power Supply (Potentiometers)	1	7.0	3
±28 VDC Reg. Pwr. Supply (Attenuators and Hi Level Temp. Transducers)	1	14.0	32
Attenuators	3	N/A	15
Flowmeter	1	2.5	16
	Total	215.8 watts	1261 oz. (78.8 lb.)

3.5.3 Component Characteristics

The component characteristics for the Instrumentation Subsystem hardware selected for the baseline are identified in Tables 3.5-3 and 3.5-4. The characteristics reflect state-of-the-art hardware presently available from numerous vendors. Typical vendor part numbers are indicated below each Table. The large number of instrumentation component manufacturers in existence coupled with the Tug study limited time span precluded a comprehensive search for component candidates.

The power and weight requirements for the Instrumentation Subsystem are summarized in Table 3.5-5. The hardware identified will be powered by the Communications and Instrumentation Distributor and essentially reflects continuous power usage from this bus. The weight as summarized does not include external component harnesses.

3.5.4 Alternate Design Approaches

Two areas appear to merit further study in terms of weight and power reduction considering the present state-of-the-art and projected industry improvement. These areas are the amplifiers utilized for the temperature and pressure measurements and the point sensor controllers. The specifications identified in the previous section reflect individual components. Improved packaging could combine multiple amplifiers and controllers utilizing state-of-the-art processes; however, this would require complete new component designs. These new designs would have to be weighed against cost, maintainability, logistics, etc. of the present designs.

3.6 AVIONICS SUBSYSTEM CHECKOUT

An analysis of avionics subsystem checkout requirements and a discussion of the checkout methods is included in this section. The special consideration of software development for the DMS is closely related to the general subject of checkout and is also included in this section.

3.6.1 Data Management

Software Development

The computer software development process will consist of the following phases:

- Definition Phase - The establishment of the concepts for the computer programs and the detailed requirements for each program and sub-program.

- Program Development Phase - The incorporation of the requirements into the computer program and the generation of the computer programs.

- Verification Phase - Verification that the computer program fulfills the design requirements.

- Maintenance Phase - Updating of the computer programs to meet new or changed requirements that are identified after release.

During the definition phase the overall program software requirements will be identified. The major software programs to be developed will be identified and the philosophy to be utilized in the construction of the individual program elements defined. The level of program language will be established and the specifications governing the writing of computer programs will be prepared.

With the completion of the definition phase, the sequence will enter the actual development of the computer programs. During this phase all aspects of the program will be analyzed to establish detailed requirements of the individual software programs. Included in this analysis will be:

- Overall program requirements and objectives
- Vehicle performance requirements
- Mission requirements
- Vehicle subsystem component capabilities
- Software program requirements
- Documentation requirements
- Test and flight data requirements
- Computer capabilities
- Program language requirements

With the integration of the requirements imposed upon the software by all the program elements, the design of individual software elements will be accomplished. The function of the design effort will be to provide a detailed document for each module that specifies sufficient information so that the actual coding of the software program modules may begin. The coding phase will begin upon completion of the detailed design. Coding is the process of transforming the required specification in the detailed design document into machine code by means of symbolic statements. These statements produced will be the first manifestation of the final deliverable computer program.

The verification phase will commence with the production of the first coded program. The software modules will be debugged as individual units prior to integrating the modules into the completed computer program.

Certain types of tests will be followed in debugging computer programs. Among these tests will be:

- Unit tests
- Integration tests
- System tests
- Acceptance tests

Unit testing is the testing of the lowest program module to identify and remove errors prior to combining the units into packages.

Integration testing is the testing of Tug program units commencing with the lowest level element and expanding in scope until complete computer programs are tested as a functional element. Integrating testing of the functionally related modules assumes the units within the modules have been unit tested.

System testing is the testing of the Tug program systems by an independent testing group. The testing will be accomplished in two environments: (1) simulation, and (2) flight hardware. The major objectives are validation (meeting requirements) and verification (implementation of design is correct). These tests assure the software program does the job required by the Tug program.

Acceptance tests are the tests applied to the system testing programs under direction and coordination of the customer. The testing will be accomplished after the system testing has been accomplished.

With the completion of the acceptance testing, the software development will enter the maintenance phase. During this phase the software programs will be maintained current with vehicle requirements. Any modifications necessary to the software will necessitate the programs being recycled through the program development and verification phase.

Subsystem Functional Checkout

The checkout requirements for the DMS will evolve from the initial development testing and integration of the Tug avionics subsystem components. The computer, memory, data bus and other hardware elements will be integrated with the software to form an operational system.

The objectives of the checkout are as follows:

1. CPU - Verify the capability to address storage, fetch and store data, process algorithmic and logical information, execute programmed instructions and handle communications between memory and the input/output (I/O) devices.

2. I/O Units - Verify ability to process communication between CPU and data bus for all vehicle subsystems.

3. Memory Units - Verify ability to store and access required memory locations in specified time intervals.

4. Data Bus - Verify capability to handle communications between major elements of the DMS and the other Tug subsystems.

Checkout of the DMS will be conducted on two levels, (1) detailed checkout prior to installation of the Tug in the EOS, and (2) limited checks while mated to the EOS.

On-board and Ground Checkout Prior to Installation in EOS

The checkout accomplished prior to installation in the EOS will consist of self-test program contained in the DMS computer and more detailed subsystem verification programs contained in a ground support computer.

The self-test programs contained in the DMS computer will be executed under sole control of the DMS computer. The results of all self test will be evaluated in the DMS computer and a go/no-go status provided through the status and control panel or via telemetry.

Failure detection and isolation of a faulty DMS element are accomplished by a combination of three elements, (1) software self-tests, (2) software comparison, and (3) built-in test hardware.

The self-test programs check the functional hardware for proper operation by using a predetermined sequence of instructions and selected bit patterns.

The isolation and detection of failures in the data bus elements are accomplished by wrap around test in which control stimuli are fed back into the monitoring circuitry to verify the functional integrity of the control response loop, and by parity check of information that is transmitted on the data bus. The operation of the IU/DAU is verified under computer control by comparing the DAU's performance with that of the instructed operation. This is accomplished by monitoring each command as it is issued from the DAU and by maintaining a command status reference profile. This status profile will continually be compared with actual status in order to detect any unscheduled change of status. The DAU contains built in test equipment to evaluate the status of each command as it is issued and relays a go/no-go status back to the computer.

The checkout of the DMS as an integral portion of the Tug avionic system will be accomplished with the aid of a ground computer and other support equipment required to close the loop for the uplink and telemetry systems. This support equipment also provides the simulation necessary to perform a simulated mission test on the complete Tug vehicle.

In this configuration the operation of the DMS computer will be verified by the ground system. The DMS computer will be controlled through the ground equipment interface. The DMS computer will be loaded and verified

by the ground system. It will be tested under control of special functional test programs in the ground equipment. The checkout will expand out from the computer to the interface between the DMS data bus elements and the various subsystems. With completion of verification of the DMS elements the checkout programs will commence a detailed evaluation of the various subsystem operation. This will be accomplished by directing the DMS computer to issue stimuli to the subsystem and monitoring the system performance to these instructions. The ground controlled checkout will be executed through both a direct interface with the DMS computer and through the utilization of the uplink as the means of communication. For those functions which cannot be accomplished in a non-space environment, stimulation will be provided using software or hardware as required.

This ground controlled testing will culminate in a test simulating actual functions as performed during the flight mission.

On-board Checkout After EOS Mating

The limited tests which will be performed while the Tug is mated to the EOS will consist of portions of the DMS self-test augmented with statusing tests of the individual subsystem. These tests will consist primarily of monitoring routines developed to evaluate the vital signs of the individual subsystems. They will not contain routines which cause the subsystems to be manipulated except for those elements which have a self-test capability built into the basic design. This testing will consist extensively of go/no-go statusing of the subsystem. The results of these tests will be provided via the status and control panel or via telemetry.

3.6.2 Guidance, Navigation and Control

Functional performance tests would be performed on all GN&C components at the bench level prior to installation in the spacecraft. These would be end-to-end tests using star and horizon simulators and rate tables to excite the input axes of optical and inertial sensors, respectively. Test tolerances would be such as to accommodate the interchanging of components within the subsystem and also accommodate degradation of performance associated with mission time and environmental extremes. Operational measurement and stimuli points would normally be utilized with more detailed measurements made available on test connectors.

The IMU and the star tracker would be mounted on a common navigation base closely aligned in the bench area and would be considered a single integral unit for installation in the spacecraft.

As GN&C components are installed in the Tug, the respective interfaces would be verified through the use of test routines in the DMS computer. Self test capabilities would be utilized where appropriate and special targets would be used to excite the inputs of the star tracker and horizon tracker. Earth rates would be used to verify proper operation and polarity of the gyros.

Combined systems and integrated system tests would be performed after component installation was completed. This would be accomplished primarily through test routines contained in the DMS computer. These tests provide

stimuli to the various elements of the subsystem with specific inputs calling for verification of appropriate outputs. Thresholds can be verified by providing stimuli at levels both below and above the specified threshold values and verifying through appropriate switching, gimballing, etc.

The effects of earth rates and gravity can be effectively used to excite the inputs of inertial devices in the IMU and thus provide effective end-to-end continuity and polarity tests while in the vehicle.

Once the Tug/Payload is loaded into the Shuttle bay, the capability to perform GN&C checkout is limited. The GN&C would be unpowered (except for heaters where required) during Shuttle ascent to low earth orbit and would only be powered up again after deployment from the Shuttle bay. When the Tug is being physically held by the Shuttle in a semi-deployed position, the physical alignment between the Shuttle and Tug will be very crude from G&C standards and therefore correlation between the two vehicles' attitude reference systems can only be used for gross malfunction detection. Furthermore, the obstruction to the Tug's optical instruments' fields-of-view will probably restrict any full operation of these devices. Consequently, self-test capabilities and artificial stimuli would probably be employed for subsystem verification at deployment.

3.6.3 Rendezvous and Docking

The Rendezvous and Docking Subsystem checkout includes the test of electrical, optical, visual and RF interface paths as well as the internal operation in all functional modes.

Checkout Prior to Installation in the EOS

Ground checkout of the Rendezvous and Docking Subsystem involves the use of target and ground station substitutes; all of which can be of a relatively simple, nearly stationary configuration. Figure 3.6-1 describes checkout equipment functions. The primary objectives of the ground tests are to assess signal quality and response to command.

Four laser reflectors are necessary to check the scanning capability of the laser; three of which simulate a payload docking configuration in the normal path direction, and the fourth simulates the EOS acquisition reflector in a path perpendicular to the other three. During the test, commands are given in a sequence similar to the actual flight sequence. The first command is flight checkout, where the laser measures an internal test reflector with known position and characteristics. Next, one of the simulated payload reflectors is acquired. All three payload reflectors are then scanned to provide three-axis attitude tracking data. Finally, the internal mirror is rotated so that the laser can acquire the EOS reflector. These tests produce single-point response data which contain all of the characteristics of actual flight. Detection of motion phenomena can be tested by rotating the payload reflector simulator mount at a fixed angular rate during the tracking test. Response data must meet accuracy and signal strength criteria to be acceptable.

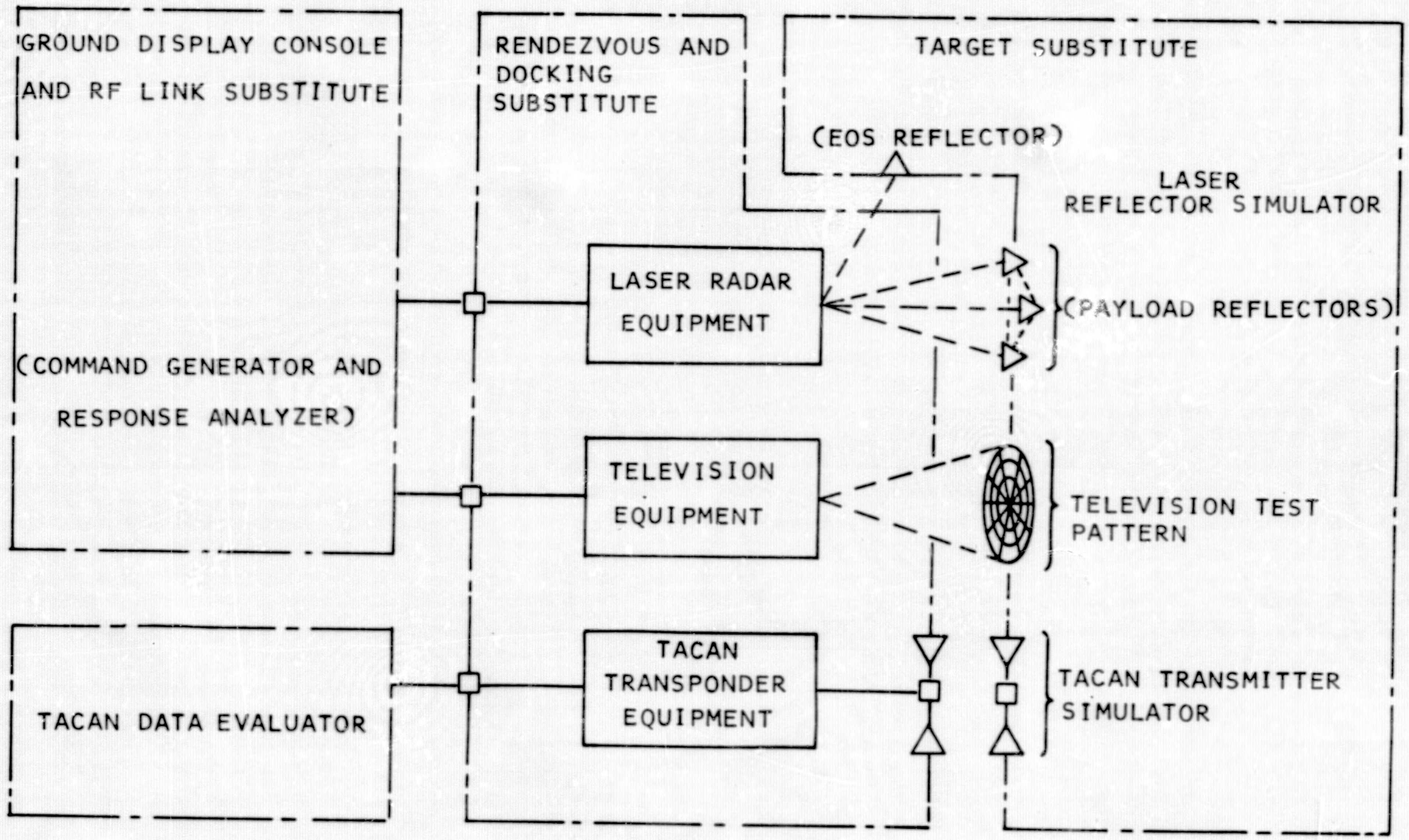


Figure 3.6-1 Rendezvous and Docking Subsystem Ground Checkout Equipment Functions

The television camera is tested by viewing a test pattern under specified lighting conditions. The picture is displayed on a television monitor and matched against picture quality test criteria.

All of the frequency range of the TACAN system is used in testing the transponder.

The ground display console and RF link substitute merely reads the laser data and displays the television picture. Also, the TACAN data evaluator reads the data output of the TACAN transponder.

Checkout While Mated to the EOS

When the Tug is inside the EOS cargo bay, Rendezvous and Docking Subsystem checkout is similar to flight checkout. The laser is tested by using internal equipment. A television picture of either the mated payload or EOS cargo bay features can be obtained with the television lights on. The TACAN transponder can be tested since antenna coupling equipment is provided.

Checkout During Flight

Television and television lighting can be completely tested at any time during flight when there is a payload attached. At other times, partial tests using starfield or earth background can be obtained without television lighting, if needed. No checks can be made of the TACAN transponder, once it is out of range of the EOS. Therefore, prior to the first injection burn of the Tug, the transponder should be tested.

The laser radar can always be tested during flight, using a method proposed by ITT. The laser return pulse is replaced with pulses of known timing from the system master time counter to produce readouts of known range, range rate, angle and angular rate. Substituting known fixed delays in the laser return pulse and checking the range reading for the various delays will also check the master time counter and the frequency of the system master oscillator. These tests check all of the system except the optical portion. The optical pickoff of the transmitted pulse provides a constant monitoring of output power. To check the optics, the rotating mirror is positioned at 45 degrees to deflect the transmitted beam into a folded optical path to a small corner reflector at a known range and angle from the system boresight. The return pulse is reflected back along the same path to the receiver. The system will read out the range and angle position of the corner reflector and can be compared to the known range and angle position.

3.6.4 Communications

The Communications Subsystem will utilize only components that have undergone a separate functional test to insure compliance to system requirements prior to installation. After component installation, checkout will be conducted on two levels, (1) detailed checkout prior to installation of the Tug in the EOS and (2) limited checks while mated to the EOS.

Detailed Checkout

A detailed checkout at the subsystem level will be performed under data management subsystem control. Each transmitter will be checked for the proper power, center frequency, and deviation. The overall transmitting subsystem will be checked to determine the total forward and reflected power so the voltage standing wave ratio (VSWR) can be calculated to verify requirement compliance. The PCM subcarrier oscillator will be checked for the proper center frequency and deviation.

Limited Checkout

After the Tug is installed in the EOS, only a limited go/no-go status check will be performed under data management subsystem control. The checkout will include all Tug communications links.

3.6.5 Instrumentation

The Instrumentation Subsystem hardware will be designed to include component test points and test connectors for detailed, bench level checkout prior to component installation in the Tug vehicle. After component installation, subsystem checkout will be performed using remote calibration circuitry for analog measurements and built-in checkout circuitry for discrete measurements. The checkout capabilities of the two types of circuitry are discussed in the following paragraphs.

Analog Measurement Checkout

The hardware used for vehicle temperature and pressure analog measurements will be designed to respond to a one point, externally (GSE) activated calibration check. This check will be used to verify subsystem operation during integrated tests performed before the Tug is installed in the EOS. After Tug installation in the EOS, and immediately prior to EOS launch, the calibration check will again be performed. Once liftoff of the EOS has occurred, the one point calibration check capability will no longer exist. Should more detailed studies indicate the need for an in-flight calibration capability, the subsystem design can be modified with a minor weight increase to provide that capability.

Discrete Measurement Checkout

The controllers used for point sensor level monitoring discrete measurements will contain internal circuitry capable of providing simulated outputs on command from the Data Management Subsystem. The controllers will respond to three checkout commands: (1) simulate wet, (2) simulate dry, and (3) simulate open. Application of the first command will produce an output corresponding to a liquid condition at the point sensor. The second command will result in a response simulating the absense of liquid at the point sensor. An output simulating an open circuit condition in the point sensor will be provided by the third command. The discrete measurement calibration capability will be used during both ground and flight checkout operations.

3.7 GROUND AND SHUTTLE ORBITER ELECTRICAL INTERFACES

The Tug avionics system will interface with the ground during checkout and prelaunch activities. There are two umbilical connectors provided for these operational modes. One connector is mounted on the vehicle forward umbilical carrier plate while the other is located on the aft umbilical carrier plate.

During checkout, prelaunch, and on orbit activities, the Tug avionics system will interface with the EOS through a single umbilical connector. The connector is located in the aft end of the Tug vehicle and affects separation at the disconnect plane of the Tug/EOS docking mechanism.

3.7.1 Requirements

The basic requirements for ground and shuttle orbiter interfaces were defined in the NASA Study Plan and are identified as follows:

1. The avionics system baseline should include a digital multiplexing technique to interconnect the subsystems with the data management. This system is also utilized as a ground and on-orbit checkout scheme.

2. In a mission abort mode while the Tug is still in the Shuttle payload bay, the Tug will be capable of dumping its propellants and safing the subsystems for a safe reentry and landing.

3. The only orbital operations to be assumed are Tug undocking and redocking with Shuttle and payload and minimum functional test of the Tug prior to its separation from the shuttle.

4. Upon redocking of the Tug with the Shuttle, the Tug/Shuttle interface must reestablish the fluid (vent, purge) and electrical (power, safety monitoring) interfaces for a safe reentry and landing.

5. The Tug and payload will have their detailed checkouts performed prior to installation in the Shuttle. Only limited functional tests will be performed at the pad and in orbit while the Tug/payload are attached to the Shuttle. These functional checkouts will be self tests by Tug and payload and will present only a go-no-go status to the Shuttle crew via hardline or RF link.

6. The Tug avionics system shall provide for monitoring of Tug critical functions for Shuttle crew safety to be monitored by the Shuttle crew at all times the payload is attached to the Shuttle.

7. The Tug will be provided a navigation update from the Shuttle prior to Shuttle/Tug separation.

8. Sustaining power to Tug during ascent and descent is Shuttle furnished.

3.7.2 Interface Connectors

The connectors selected for the interfaces will be capable of tolerating a maximum of .090" mechanical misalignment and will mate when the interface carrier plates are engaged during tug-orbiter docking. The connectors are designed to be self-aligning and the contacts do not engage until the shell alignment key has entered the keyway of the connector receptacle. An interfacial compression spring is provided as an integral part of the connector assembly to insure that the mating halves of the connector maintain the proper amount of compression for sealing. The connector is also designed to compensate for up to .50" of overtravel of the carrier plates without inducing damage. (See Figures 3.7-1 and 3.7-2.)

The connector mating force will be 50 lb. maximum while the demating force will be 15 lb. maximum.

The connector shell size will be approximately a #24 shell size of a MIL-C-26482 type connector. The insert arrangement for the connectors is dependent upon a more detailed design analysis.

3.7.3 Interface Functions

Tug/Ground

The Tug interfaces with the GSE through two umbilical links that pass through the outer shell ofthe EOS. One umbilical is located in the aft area of the Tug and the second is located in the forward area.

The functions that are routed through the GSE umbilicals are those which are necessary to provide ground power and control for propellant loading, vehicle safing, computer initialization and verification, and vehicle checkout and statusing during prelaunch/launch operations. The majority of the critical command functions are redundant to command functions wired to the EOS status and control panel panel. This allows ground operations to be carried out independent of EOS crew support.

The functions to be carried through the aft ground interface connector are identified in Table 3.7-1. The functions to be carried through the forward ground interface connector are identified in Table 3.7-2.

Shuttle Orbiter/Tug

Interface between the EOS and Tug is achieved through a single umbilical connector mounted in the aft end of the Tug. The connector is located at the EOS/Tug docking mechanism separation plane and remains mated during the swing-out of the Tug from the EOS cargo bay. It is disconnected after initial Tug systems on-orbit start up and checkout have been completed and the Tug is demated from the EOS to perform its mission.

The functions that are routed through the EOS/Tug umbilical are those which are necessary to provide power and control for vehicle safing, navigational state vector handoff, thermal requirements, vehicle checkout and statusing, and computer take-over while the EOS and Tug are mated.

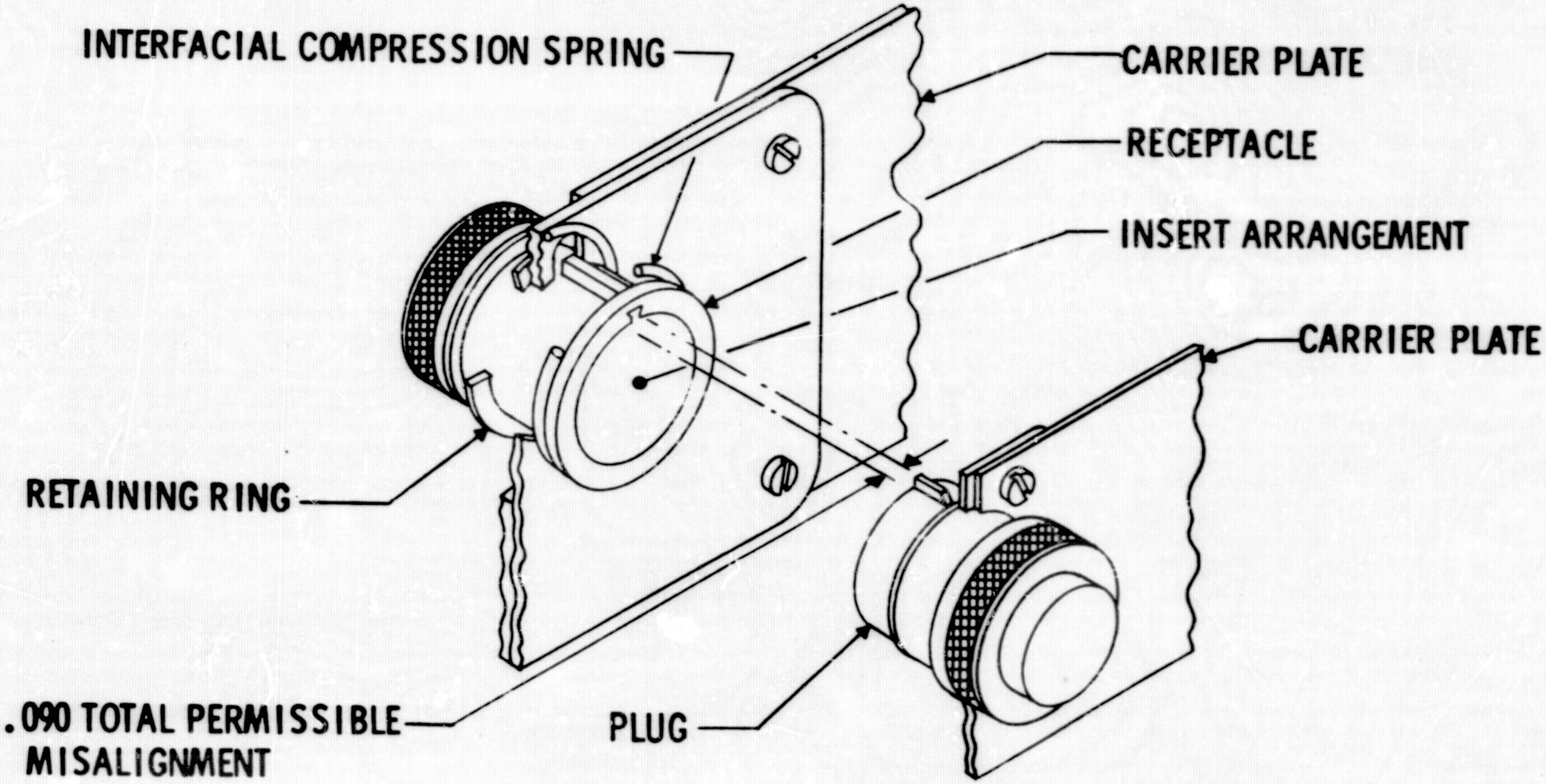


Figure 3.7-1 Electrical Umbilical Connector Design

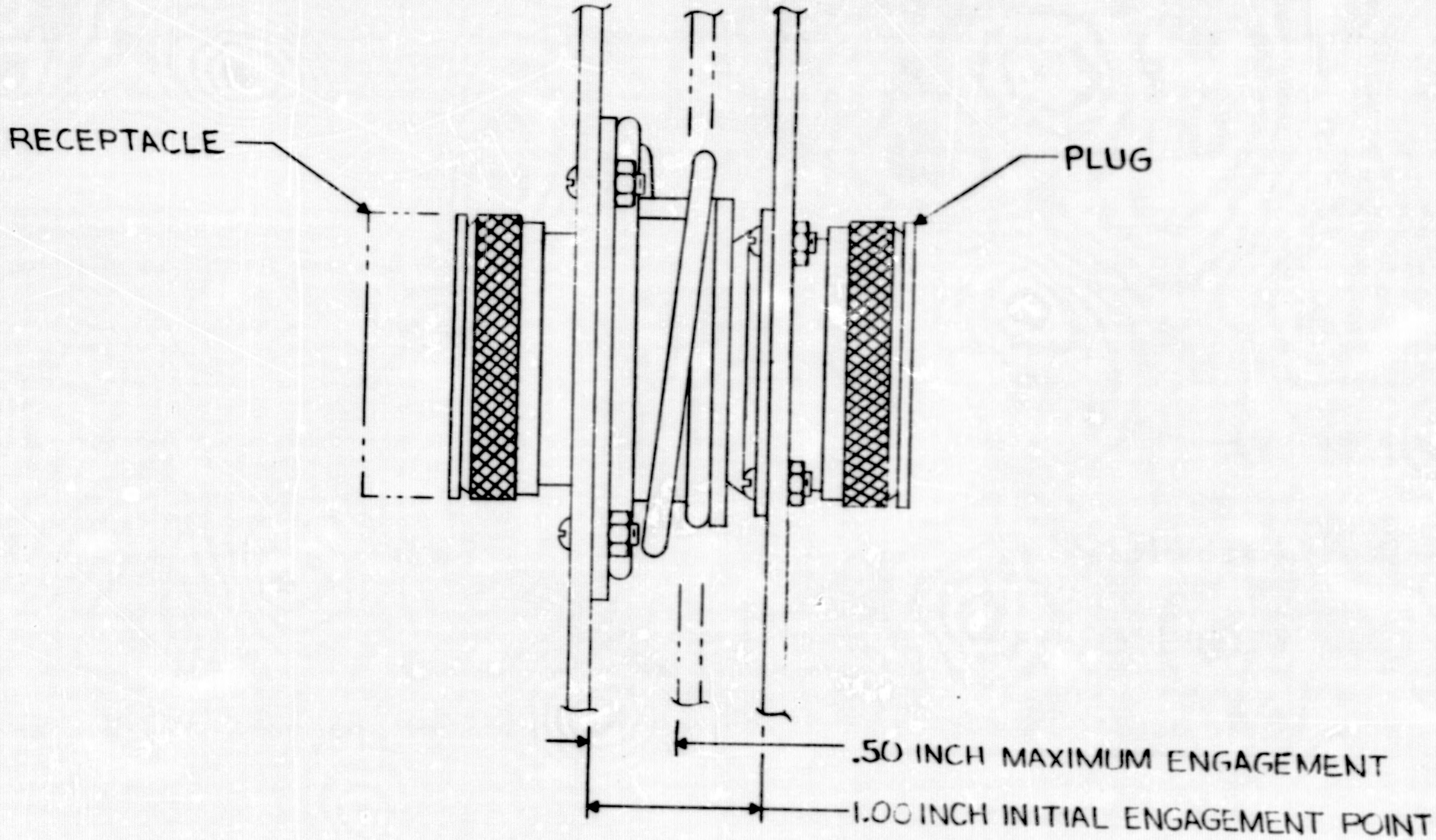


Figure 3.7-2 Electrical Umbilical Connector Engagement

The functions required to be routed from the EOS status and control panel to Tug docking mechanism located components are identified in Table 3.7-4.

Table 3.7-1 Ground Interface - Aft Connector Function Listing

Command Functions
Power Transfer Switch to EOS Power Transfer Siwtch to Tug Power Transfer Switch to GSE LOX Tank Vent Selector Valve No. 1 Open LOX Tank Vent Selector Valve No. 1 Close LOX Tank Vent Selector Valve No. 2 Open LOX Tank Vent Selector Valve No. 2 Close LOX Tank Vent Valve Open LOX Tank Vent Valve Close LOX Fill & Drain Valve No. 1 Open LOX Fill & Drain Valve No. 1 Close LOX Fill & Drain Valve No. 2 Open LOX Fill & Drain Valve No. 2 Close LOX Tank Helium Purge Control Valve Open LOX Prevalve Open LOX Prevalve Close APS GOX Isolation Valve Open APS GH_2 Isolation Valve Open APS GOX Isolation Valve Close APS GH_2 Isolation Valve Close *APS LOX Feed Isolation Valve Open *APS LOX Feed Isolation Valve Close *LOX Aux Prop Tank Fill Valve Open *LOX Aux Prop Tank Vent Control Valve Open Fuel Cell GH_2 Reactant Control Valve Open Fuel Cell GOX Reactant Control Valve Open Command Signals Return
Response Functions
LOX Tank Helium Purge Pressure Switch On/Off LOX Tank Capacitance Probe (COAX)
Power Functions
GSE Power (28 VDC) *These commands are not redundant to EOS/Tug interface commands

Table 3.7-2 Ground Interface - Forward Connector Function Listing

Command Functions
LH_2 Tank Vent Selector Valve No. 1 Open
LH_2 Tank Vent Selector Valve No. 1 Close
LH_2 Tank Vent Selector Valve No. 2 Open
LH_2 Tank Vent Selector Valve No. 2 Close
LH_2 Tank Vent Valve Open
LH_2 Tank Vent Valve Close
LH_2 Fill & Drain Valve No. 1 Open
LH_2 Fill & Drain Valve No. 1 Close
LH_2 Fill & Drain Valve No. 2 Open
LH_2 Fill & Drain Valve No. 2 Close
LH_2 Tank Helium Purge Control Valve Open
LH_2 Prevalve Open
LH_2 Prevalve Close
*APS LH_2 Feed Isolation Valve Open
*APS LH_2 Feed Isolation Valve Close
*LH_2 Aux Prop Tank Fill Valve Open
*LH_2 Aux Prop Tank Vent Control Valve Open
Command Signals Return
Response Functions
LH_2 Tank Helium Purge Pressure Switch On/Off
LH_2 Tank Capacitance Probe (COAX)
Communication Subsystem Checkout Link (COAX)
Rendezvous Transponder Checkout Link (COAX)
Data Management Functions
Data Bus Terminals
Computer Initialization & Verification
1. Input Data Lines (16 lines)
2. Output Data Lines (16 lines)
3. Control lines
a. Clock
b. Computer Ready to Accept Data
c. Computer Ready to Output Data
d. System Reset
e. Computer Program Start
*These commands are not redundant to EOS/Tug interface commands.

All functions that are required for actuation of Tug-mounted components that are operationally sequenced only while mated to the EOS are also routed through the EOS/Tug interface.

With the exception of electrical power sources, all functions routed through the EOS/Tug umbilical are interfaced with the Data Management Subsystem status and control panel which is installed on the EOS. Switches and displays on the status and control panel allow the EOS crew to control and monitor operations while the EOS and Tug are mated.

The functions to be carried through the Shuttle Orbiter interface connector are those identified in Table 3.7-3.

Shuttle Orbiter/Tug Docking Adapter

Certain components which require power and control are installed on the EOS/Tug docking adapter. With the exception of power sources, the functions for control of and response from these components are routed directly to the status and control panel. Table 3.7-4 identifies these functions.

3.8 AVIONICS EQUIPMENT INSTALLATION

The avionics system components are installed on open panels mounted in the forward and aft skirt areas of the Tug vehicle. Data management, primary GN&C, communications, and rendezvous and docking components are installed in the forward skirt. Power generation components are located in the aft skirt. GN&C engine control and backup stabilization assemblies are installed in the aft end of the Tug to minimize wire runs to the components they serve. Instrumentation, power distribution, and data management data acquisition units are installed in both the forward and aft areas in close proximity to the components they service. Power and signal transmission between forward and aft located components is via cabling routed through two electrical systems conduit runs.

3.8.1 Requirements

The requirements adhered to in the equipment installation design are a combination of NASA guidelines and NR design analysis constraints. The NR design analysis installation constraints are identified in the sections of this report describing the various avionics subsystems and are not reiterated in this section. The NASA guidelines affecting equipment installation are as follows:

1. The Tug will be designed as an integral vehicle with none of the subsystems designed to be removable as a kit or single unit.

2. The Tug will be designed for a mission life of 20 missions. Refurbishment of subsystems after each mission is acceptable.

3. The EOS cargo bay shall be vented during launch and entry phases and operate unpressurized during the orbital phase of the mission.

Table 3.7-3 Shuttle Orbiter Interface Connector Function Listing

Command Functions
Power Transfer Switch to EOS
Power Transfer Switch to Tug
Power Transfer Switch to GSE
LH_2 Tank Vent Selector Valve No. 1 Open
LH_2 Tank Vent Selector Valve No. 1 Close
LH_2 Tank Vent Selector Valve No. 2 Open
LH_2 Tank Vent Selector Valve No. 2 Close
LOX Tank Vent Selector Valve No. 1 Open
LOX Tank Vent Selector Valve No. 1 Close
LOX Tank Vent Selector Valve No. 2 Open
LOX Tank Vent Selector Valve No. 2 Close
LH_2 Tank Vent Valve Open
LH_2 Tank Vent Valve Close
LOX Tank Vent Valve Open
LOX Tank Vent Valve Close
LH_2 Fill & Drain Valve No. 1 Open
LH_2 Fill & Drain Valve No. 1 Close
LH_2 Fill & Drain Valve No. 2 Open
LH_2 Fill & Drain Valve No. 2 Close
LOX Fill & Drain Valve No. 1 Open
LOX Fill & Drain Valve No. 1 Close
LOX Fill & Drain Valve No. 2 Open
LOX Fill & Drain Valve No. 2 Close
LH_2 Tank Helium Purge Control Valve Open
LOX Tank Helium Purge Control Valve Open
LH_2 Prevalve Open
LH_2 Prevalve Close
LOX Prevalve Open
LOX Prevalve Close
APS GH_2 Isolation Valve Open
APS GH_2 Isolation Valve Close
APS GOX Isolation Valve Open
APS GOX Isolation Valve Close
Fuel Cell GH_2 Reactant Control Valve Open
Fuel Cell GOX Reactant Control Valve Open
Fuel Cell Start
Fuel Cell Stop
Aft Data Acquisition Unit No. 1 PCS On
Forward Data Acquisition Unit No. 1 PCS On
Command Signals Return
Response Functions
LH_2 Tank Helium Purge Pressure Switch On/Off
LOX Tank Helium Purge Pressure Switch On/Off

Table 3.7-3 Shuttle Orbiter Interface Connector Function Listing (Cont'd)

Power Functions
EOS Sustaining Power (28 VDC)
Data Management Functions
Data Bus Terminals
Computer Program Start

Table 3.7-4 Shuttle Orbiter/Tug Docking Mechanism Interface Function Listing

Command Functions
EOS/Tug Latches Engage
EOS/Tug Latches Disengage
EOS/Tug Probes Extend
EOS/Tug Probes Retract
Helium Purge Isolation Valve Open
Insulation Purge Shut Off Valve Open
Command Signals Return
Response Functions
EOS/Tug Latch No. 1 Engaged/Disengaged
EOS/Tug Latch No. 2 Engaged/Disengaged
EOS/Tug Latch No. 3 Engaged/Disengaged
EOS/Tug Latch No. 4 Engaged/Disengaged
EOS/Tug Latch No. 5 Engaged/Disengaged
EOS/Tug Latch No. 6 Engaged/Disengaged
EOS/Tug Latch No. 7 Engaged/Disengaged
EOS/Tug Latch No. 8 Engaged/Disengaged
EOS/Tug Latch No. 9 Engaged/Disengaged
EOS/Tug Latch No. 10 Engaged/Disengaged
EOS/Tug Latch No. 11 Engaged/Disengaged
EOS/Tug Latch No. 12 Engaged/Disengaged
EOS/Tug Latch No. 13 Engaged/Disengaged
EOS/Tug Latch No. 14 Engaged/Disengaged
EOS/Tug Latch No. 15 Engaged/Disengaged
EOS/Tug Latch No. 16 Engaged/Disengaged
EOS/Tug Latch No. 17 Engaged/Disengaged
EOS/Tug Latch No. 18 Engaged/Disengaged
EOS/Tug Latch No. 19 Engaged/Disengaged
EOS/Tug Latch No. 20 Engaged/Disengaged
EOS/Tug Latch No. 21 Engaged/Disengaged
EOS/Tug Latch No. 22 Engaged/Disengaged
EOS/Tug Latch No. 23 Engaged/Disengaged
EOS/Tug Latch No. 24 Engaged/Disengaged
EOS/Tug Probe No. 1 Extended
EOS/Tug Probe No. 1 Retracted

Table 3.7-4 Shuttle Orbiter/Tug Docking Mechanism Interface Function Listing (Cont'd)

Response Functions (Cont'd)
EOS/Tug Probe No. 2 Extended EOS/Tug Probe No. 2 Retracted EOS/Tug Probe No. 3 Extended EOS/Tug Probe No. 3 Retracted Helium Purge Isolation Valve Open/Closed Insulation Purge Shut Off Valve Open/Closed
Power Functions
EOS/Tug Latch Actuation Power (115 VAC, 3 , 400 Hz) EOS/Tug Latch/Probe/Valve Position Indicators Power (28 VDC)

4. The EOS cargo bay internal wall temperature thermal environment is defined as follows:

Condition	Temperatures (°F) Minimum	Maximum
Pre-launch	-100	+120
Launch	-100	+200
On-orbit (door closed)	-100	+200
On-orbit (door open)	Undefined	Undefined
Entry and post landing	-100	+200

5. The Tug is to be designed for groundbased operation with all propellant loading, payload/Tug assembly, maintenance, repair and refurbishment to be done on the ground.

6. The Tug should be designed to be installed with the EOS either horizontal or vertical.

7. A passively cooled system is baselined for the avionics equipment. Heaters and louvers are to be considered passive devices.

8. The outer vehicle mold line dimension (15 ft. max. diameter) is to be considered a hard requirement.

9. The DOD should be provided the structural and mechanical details of the area allotted for the communication subsystem in order that they can determine the physical characteristics of equipment that they add to the communication subsystem.

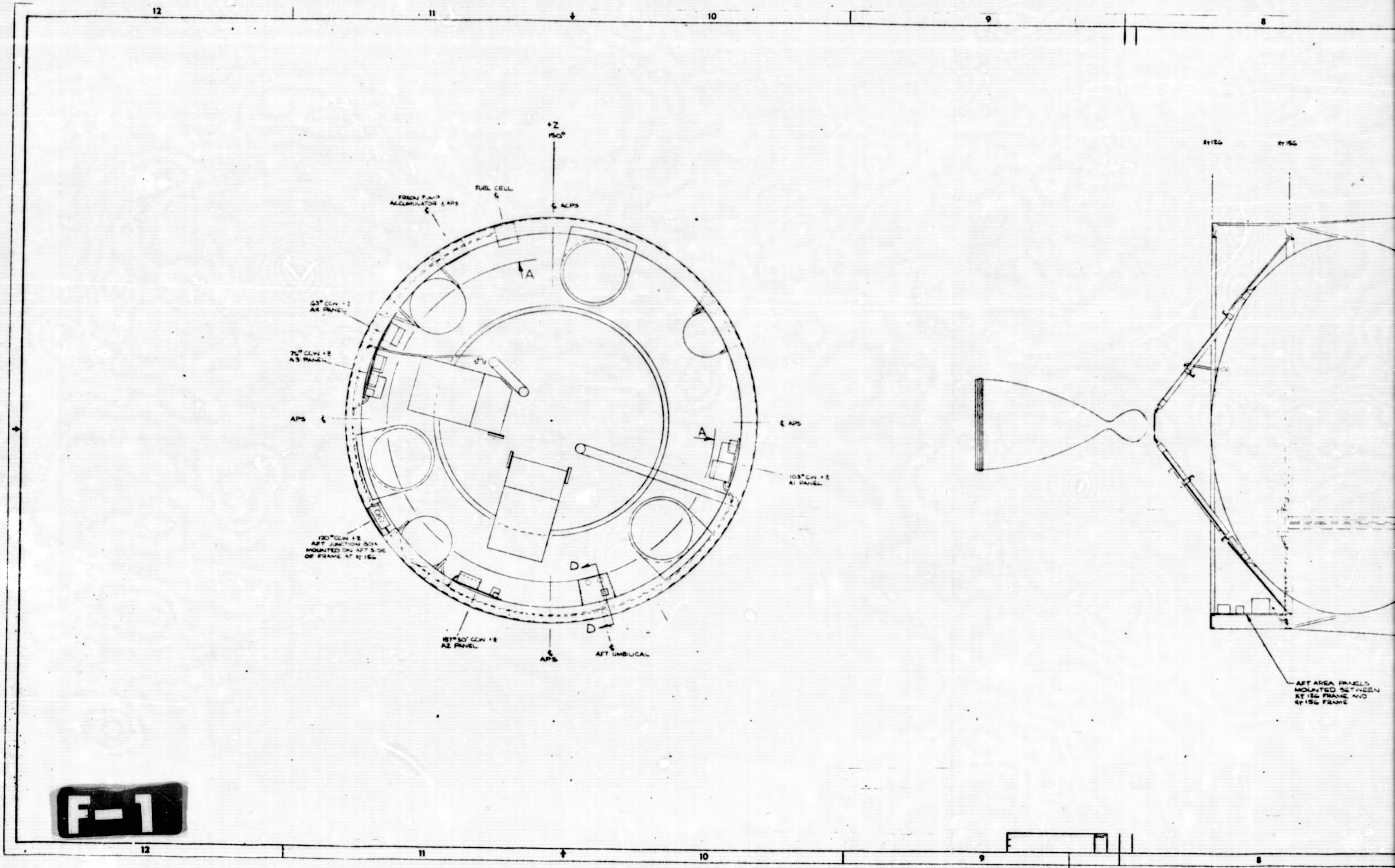
3.8.2 Component Mounting

Panels

The avionics equipment will be mounted on honeycomb panels and located in two areas on the vehicle as shown on Avionic Equipment Packaging and Layout Installation V7-975403, Reference Figure 3.7-1. The panels will be approximately 20" x 30" x 1/2" in size, and constructed of 1/2" thick honeycomb core sandwiched between 0.020 gauge aluminum sheet. Threaded inserts will be bonded in the panels for the installation of equipment mounting fasteners. Eight panels will be installed between the ring frames in the forward skirt. These panels will be mounted inboard of the mold line on angles attached to ring frames. The equipment located on the forward skirt panels is listed in Table 3.8-1. Four panels are installed between the ring frames in the aft skirt mounted on inboard caps. The equipment located on the aft skirt panels is listed in Table 3.8-1.

Thermal Control Structure

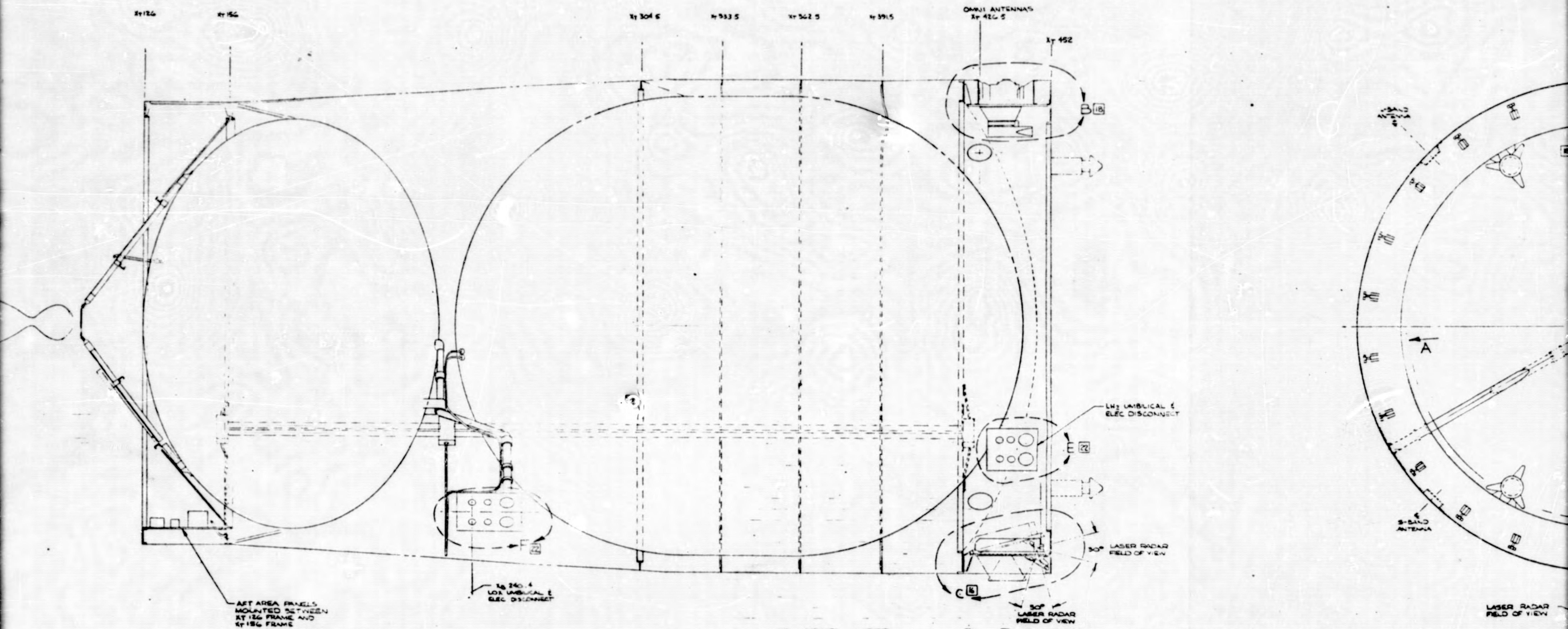
The equipment panels will provide a thermal conduction path for transferring heat. Since the equipment will be designed for heat sink, the panel will provide a means of stabilizing the component temperatures. Glass-fiber spacers will be used to insulate the plates from the structure. Electrical heaters for individual panels will be used to maintain a safe minimum operating temperature. Thermal control louvers will be used where



12
11
10
9
8
+Z
180°
FUEL CELL ℄
FREON PUMP ACCUMULATOR ℄ APS
℄ ACPS
A
63° CCW +Z A4 PANEL
76° CCW +Z A3 PANEL
APS ℄
℄ APS
A
103° CCW +Z A1 PANEL
120° CCW +Z AFT JUNCTION BOX MOUNTED ON AFT SIDE OF FRAME AT X_T 156
D
D
157° 30' CCW +Z A2 PANEL
℄ APS
℄ AFT UMBILICAL
X_T 126
X_T 156
AFT AREA PANELS MOUNTED BETWEEN X_T 126 FRAME AND X_T 156 FRAME
12
11
10
9
8

F-1

8 7 6 5 4



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CV7-975403

8 7 6 5 4

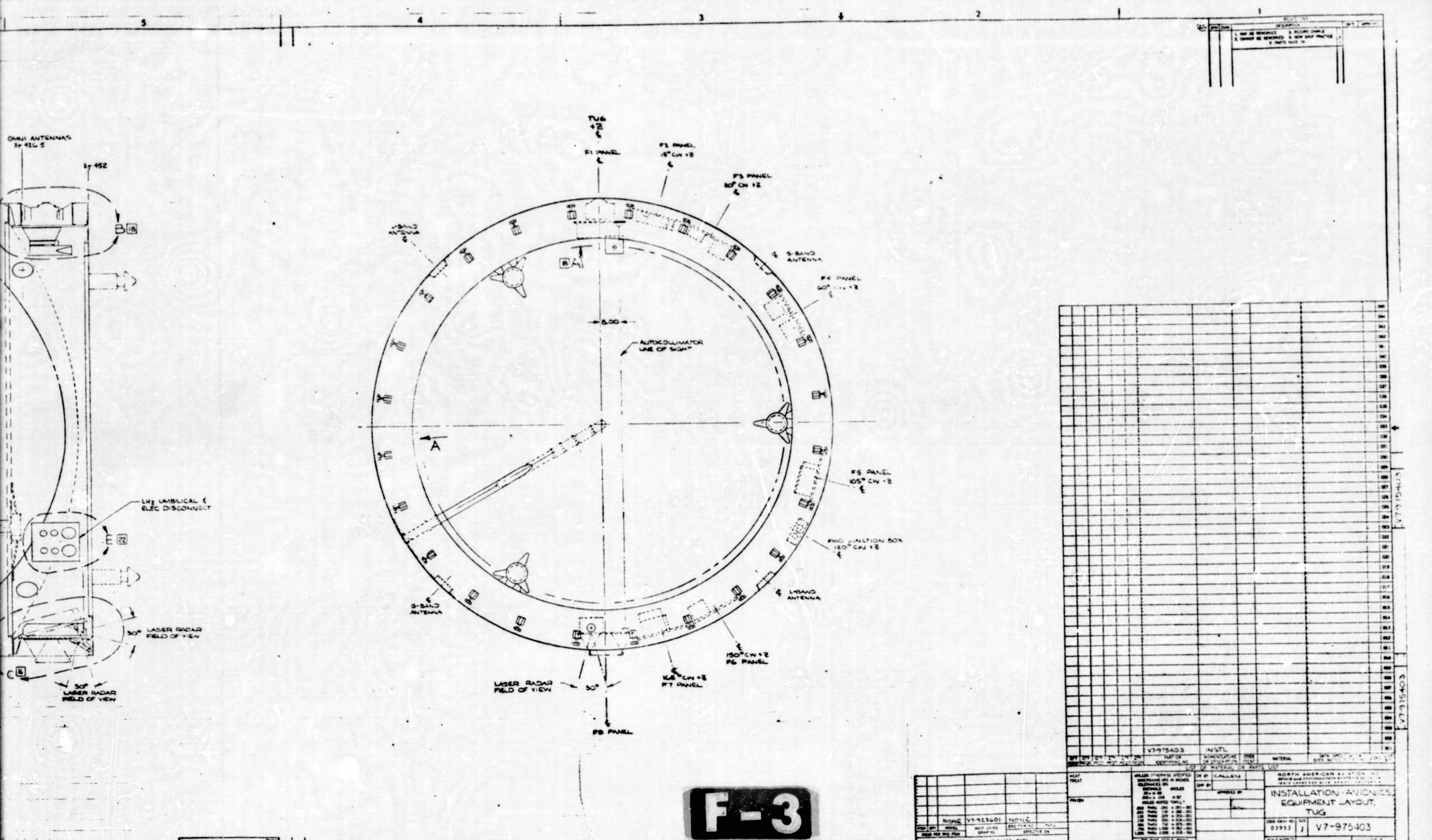

Figure 3.8-1 V7-975403 Installation - Avionics, Equipment Layout, Tug (Sheet 1 of 3)

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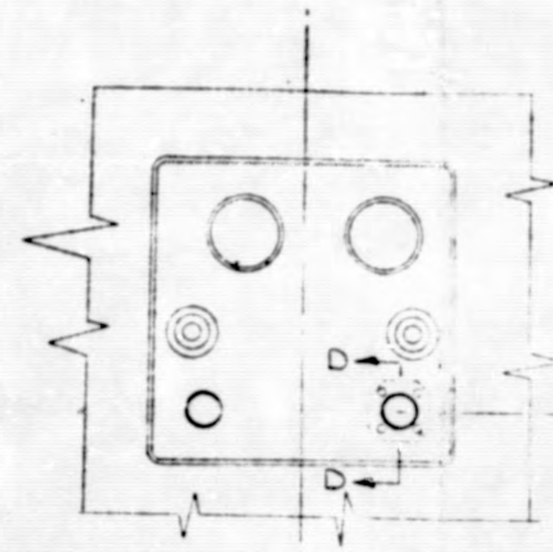
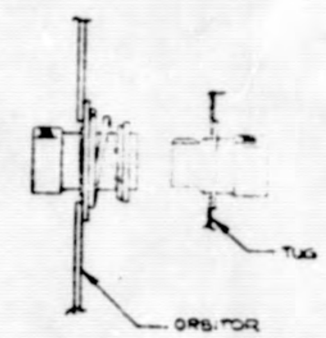


DETAIL E
SCALE: 1/4
LH_2 UMBILICAL



SECTION D-D
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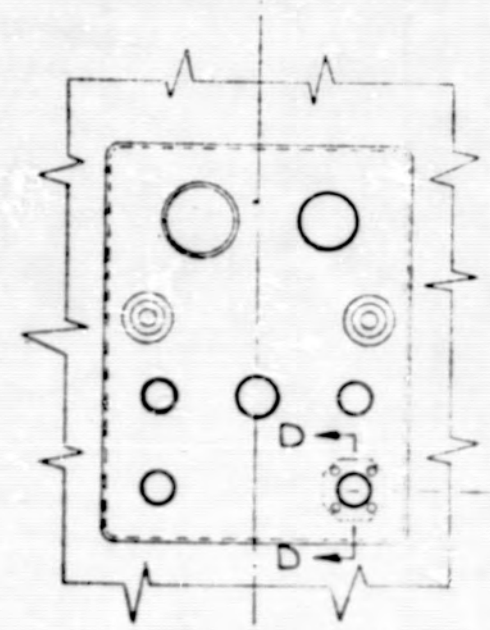
ELECTRICAL DISCONNECT



DETAIL F
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LOX UMBILICAL

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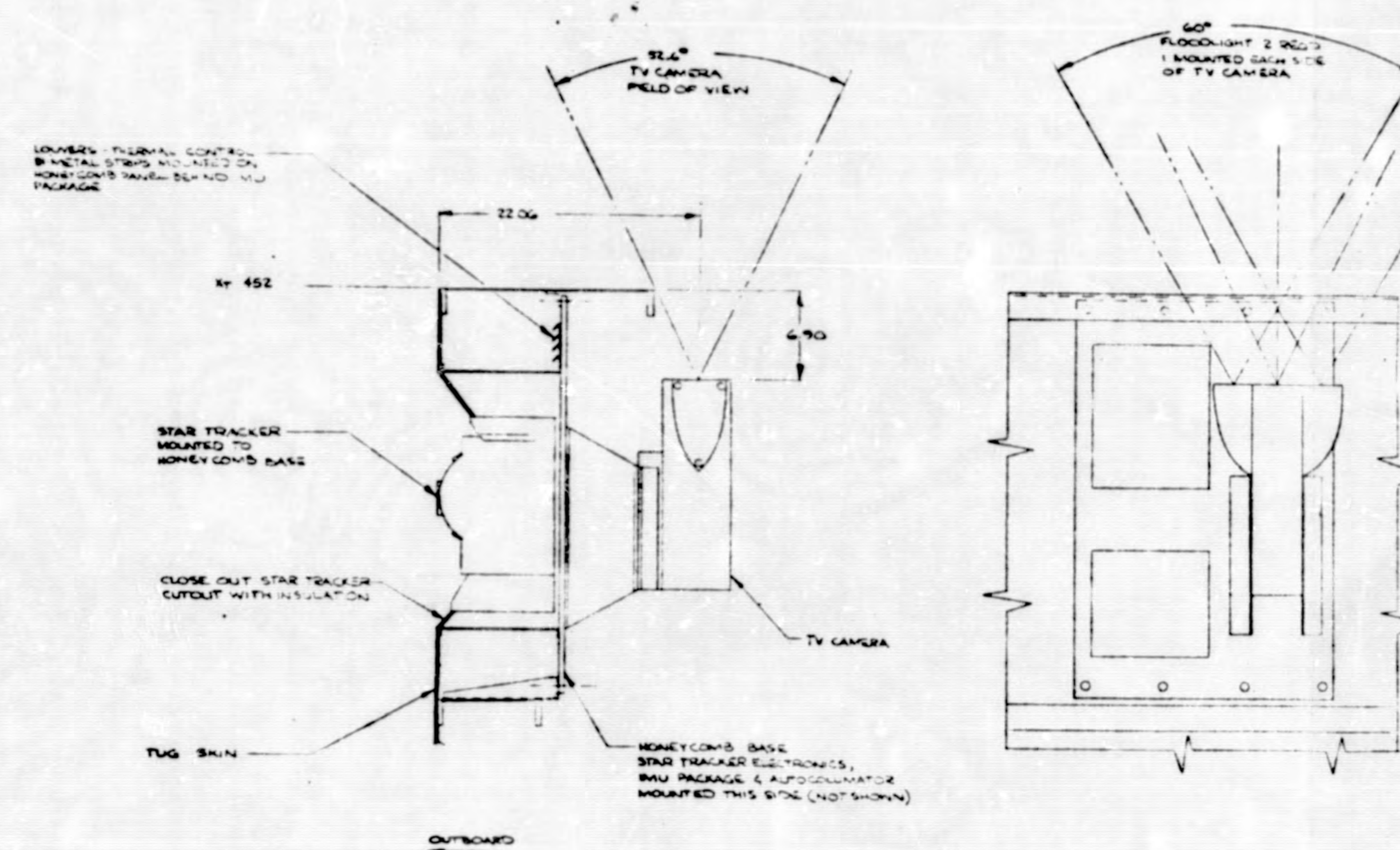
VT-975403 2

20 19 18 17 16



DETAIL B
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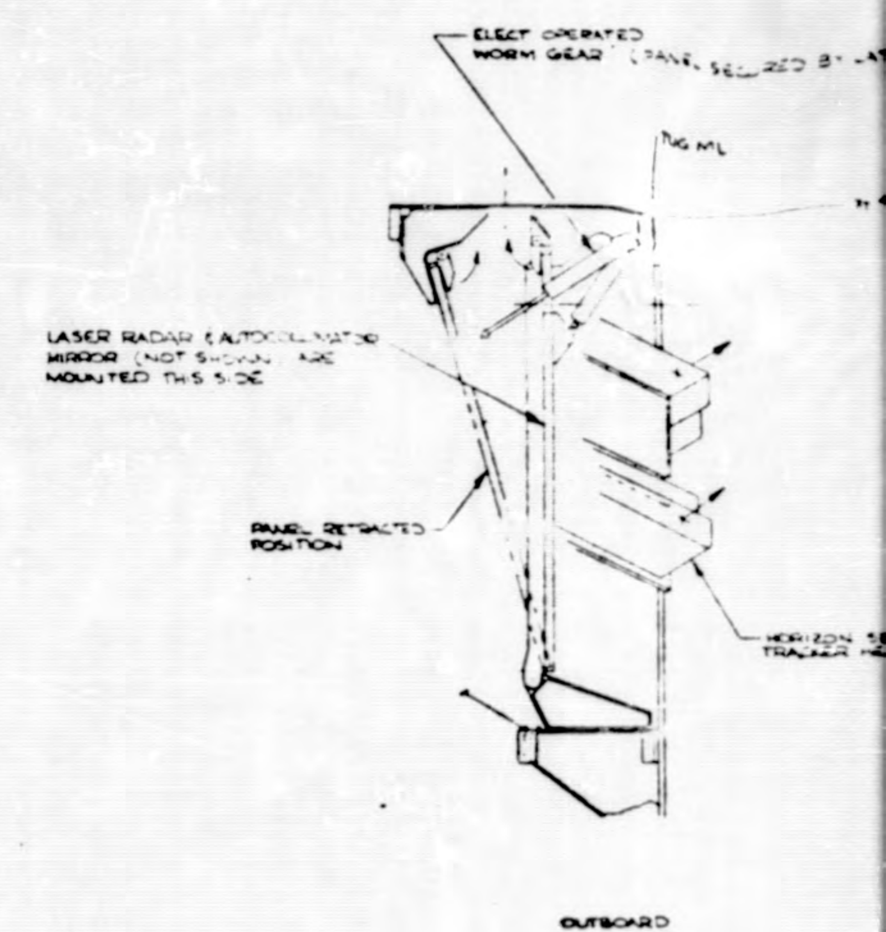

DETAIL C
SCALE: 1/4

F-2

VT-975403 2

20 18 17 16

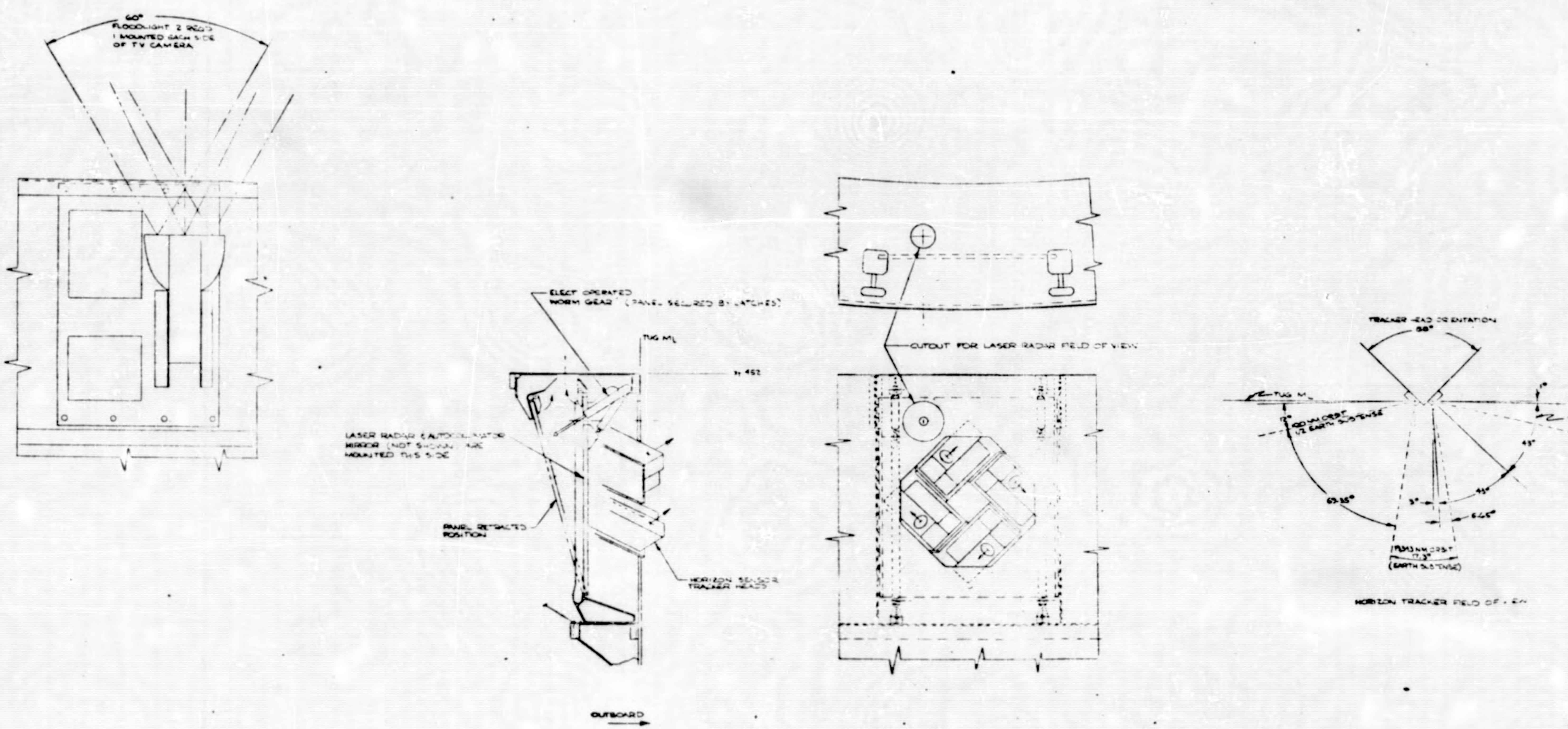

Figure 3.8-1 V7-975403 Installation - Avionics, Equipment Layout, Tug (Sheet 2 of 3)

V7-975403

F-3

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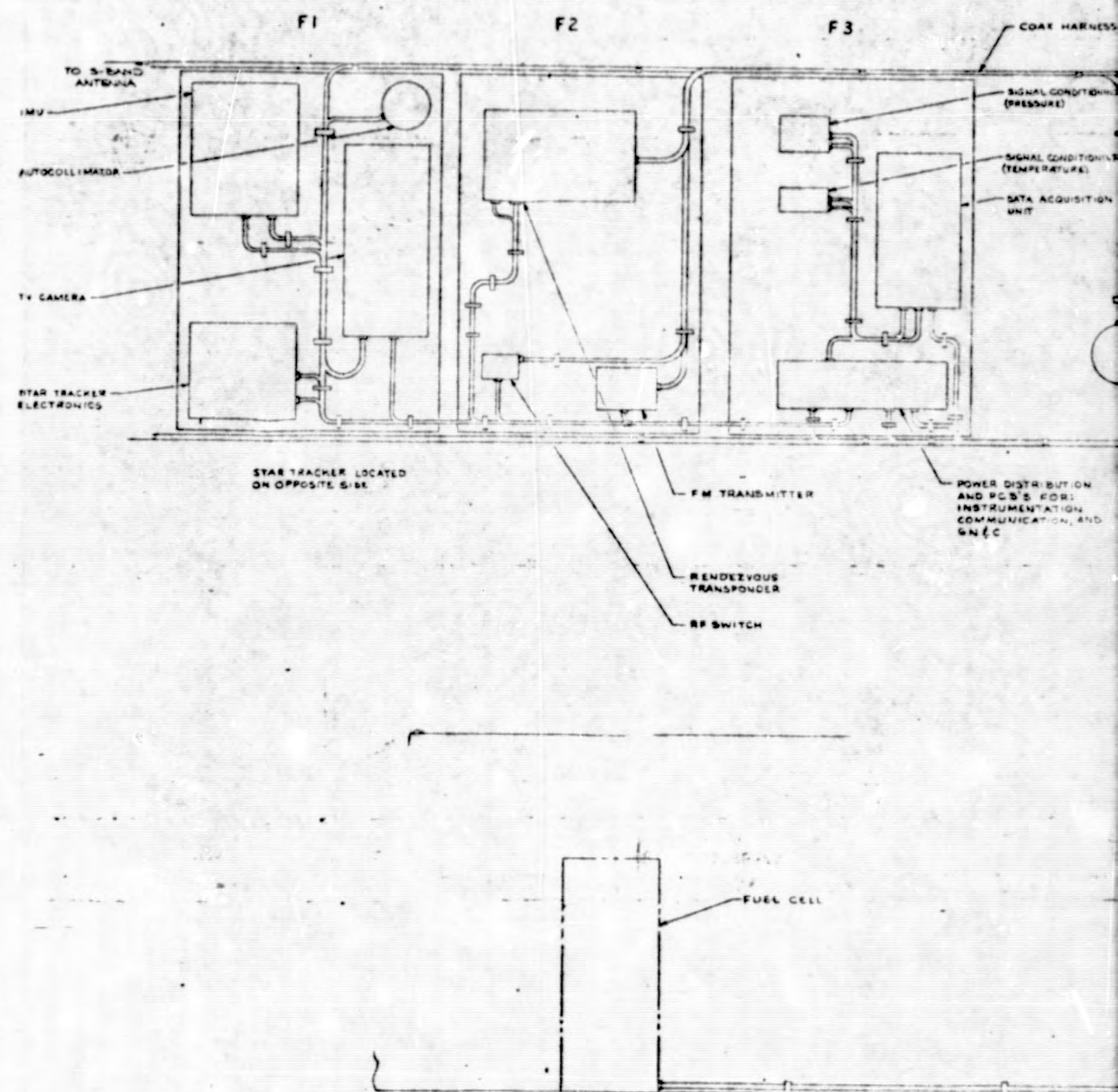
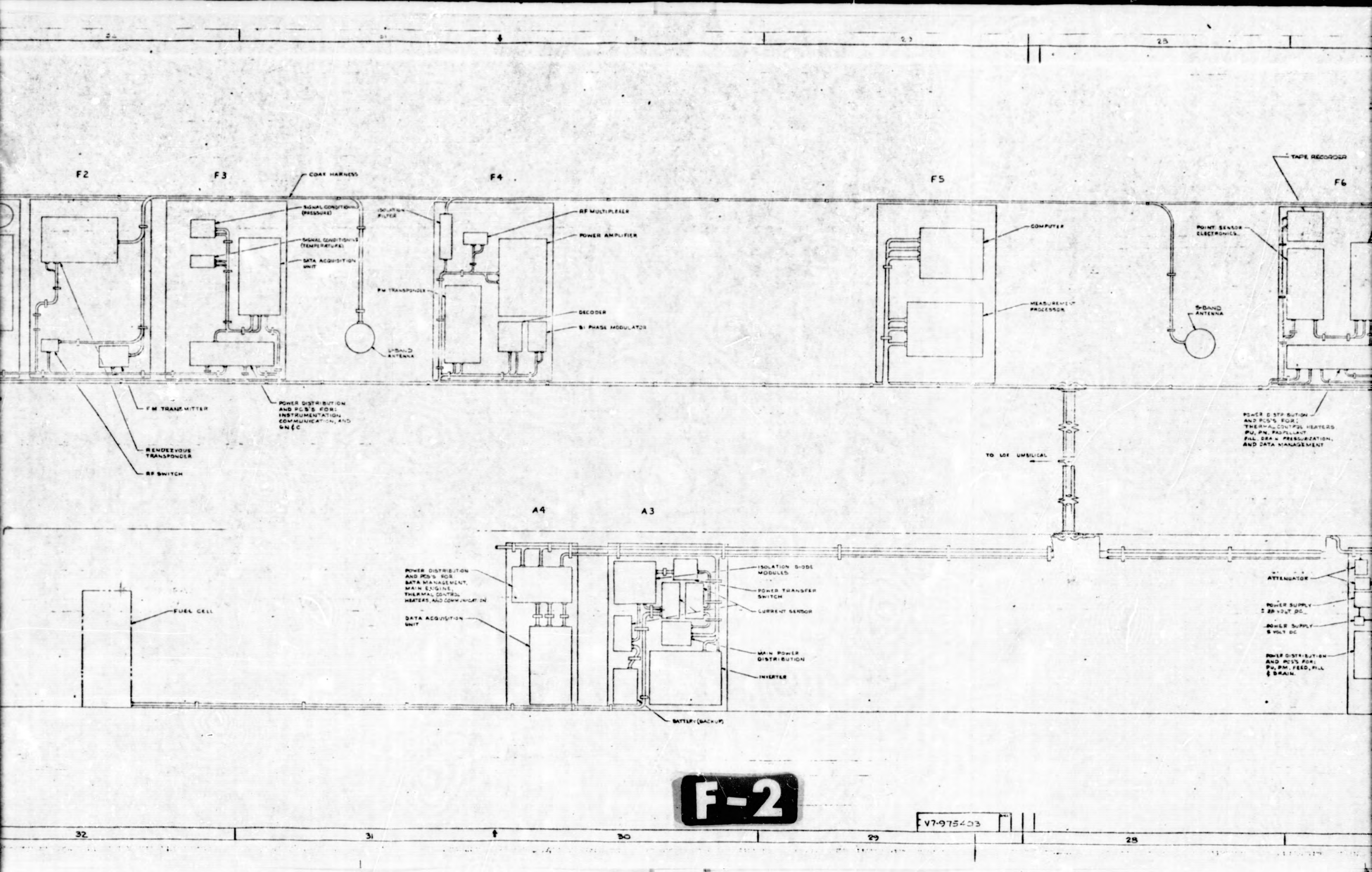

F1
F2
F3
COAX HARNESS
TO S-BAND ANTENNA
IMU
AUTOCOLLIMATOR
TV CAMERA
STAR TRACKER ELECTRONICS
SIGNAL CONDITIONING (PRESSURE)
SIGNAL CONDITIONING (TEMPERATURE)
DATA ACQUISITION UNIT
STAR TRACKER LOCATED ON OPPOSITE SIDE
FM TRANSMITTER
POWER DISTRIBUTION AND PCB'S FOR: INSTRUMENTATION COMMUNICATION, AND GN&C
RENDEZVOUS TRANSPONDER
RF SWITCH
FUEL CELL

F-1


F2
F3
F4
F5
F6
COAX HARNESS
SIGNAL CONDITIONING (PRESSURE)
SIGNAL CONDITIONING (TEMPERATURE)
DATA ACQUISITION UNIT
ISOLATION FILTER
RF MULTIPLEXER
POWER AMPLIFIER
PM TRANSPONDER
DECODER
BI PHASE MODULATOR
S-BAND ANTENNA
COMPUTER
MEASUREMENT PROCESSOR
TAPE RECORDER
POINT SENSOR ELECTRONICS
S-BAND ANTENNA
FM TRANSMITTER
POWER DISTRIBUTION AND PCB'S FOR: INSTRUMENTATION, COMMUNICATION, AND GN&C
RENDEZVOUS TRANSPONDER
RF SWITCH
TO LOX UMBILICAL
POWER DISTRIBUTION AND PCB'S FOR: THERMAL CONTROL HEATERS, PU, PM, PROPELLANT FILL, DRAIN, PRESSURIZATION, AND DATA MANAGEMENT
A4
A3
FUEL CELL
POWER DISTRIBUTION AND PCB'S FOR DATA MANAGEMENT, MAIN ENGINE, THERMAL CONTROL HEATERS, AND COMMUNICATION
DATA ACQUISITION UNIT
ISOLATION DIODE MODULES
POWER TRANSFER SWITCH
CURRENT SENSOR
MAIN POWER DISTRIBUTION
INVERTER
BATTERY (BACKUP)
ATTENUATOR
POWER SUPPLY 28 VOLT DC
POWER SUPPLY 5 VOLT DC
POWER DISTRIBUTION AND PCB'S FOR: PU, PM, FEED, FILL & DRAIN
V7-975403
32
31
30
29
28

F-2

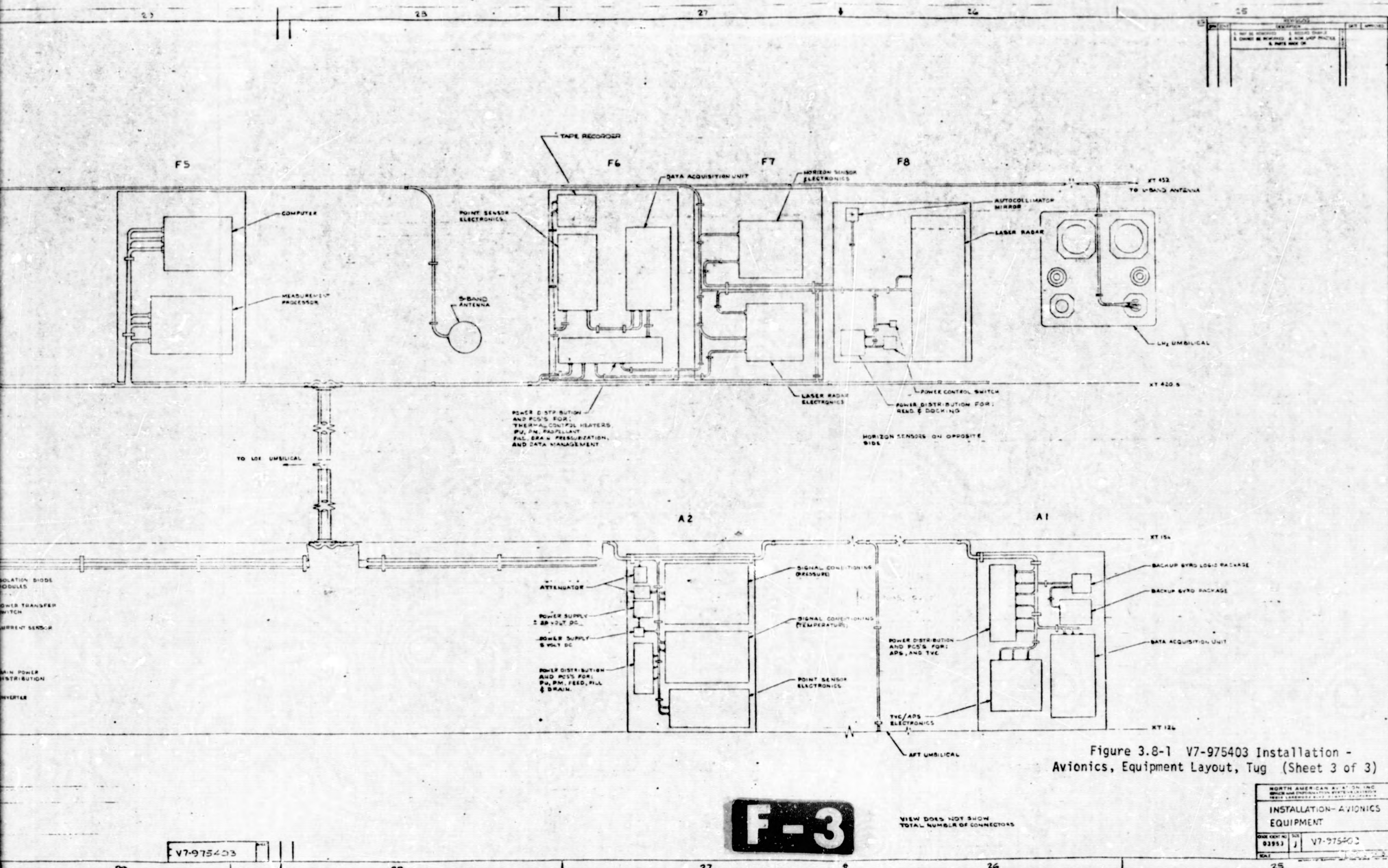


Figure 3.8-1 V7-975403 Installation - Avionics, Equipment Layout, Tug (Sheet 3 of 3)

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Table 3.8-1. Component Mounting Equipment Summary

Panel No.	Equipment	Weight/Pounds Panel, Mounting Hardware & Louvers
	FORWARD	
F1	GN & C, rendezvous, and docking	10
F2	GN & C, communication, rendezvous and docking	2
F3	Instrumentation, Data Management, communication, GN & C, and power distribution	3
F4	Communication	8
F5	Data Management	2
F6	Data Management, propellant monitor, and power distribution	3
F7	GN & C, rendezvous and docking	2
F8	GN & C, rendezvous, docking, and power distribution	8.5
	AFT	
A1	GN & C, data management, power distribution	2
A2	Instrumentation	5
A3	Power distribution	3
A4	Data management and power distribution	2
		Total 50.5 lbs.

required to maintain a safe maximum operating temperature. The thermal control louver assemblies will be mounted on the panel facing the outer vehicle skin. Heat will be conducted from the equipment to the panel and to an actuator housing. The heat is then radiated to bimetallic actuators. As the temperature increases, the bimetallic actuators contract. This creates a torque which opens the louver blades. Heat from the skink escapes by radiation through the open blades and through the vehicle skin to space. The panels requiring the installation of the thermal control louvers are F1 and F4.

Horizon Tracker

Panel F8, where the horizon tracker is installed, will require deployment during orbit because a proper field of view for the horizon tracker can only be attained by extending the horizon tracker external of the outer mold line of the vehicle. See Zone 16 on V7-975403. The panel is hinged at the base and when stowed in the orbiter bay will be retracted. Upon deployment of the Tug from the orbiter bay the panel will be deployed and locked into operational position.

The actuation system to be used for deployment of the panel will be an electrically operated worm gear. The panel when deployed will be mechanically latched. A limit switch will be used in conjunction with the latch to limit the actuator extension or retraction. While the panel is retracted it will be mechanically latched to restrain the panel during launch or that period of high mechanical vibration. When the actuator is energized for extension or retraction the latch locking mechanism will disengage and allow the panel to extend or retract. The horizon tracker is mounted on a machined part which will provide a minimum tolerance alignment installation surface. The laser radar and autocollimator will be mounted on the inboard side of the same panel.

Star Tracker

The F1 panel will have the star tracker mounted to the outboard side of the panel. See Zone 18 on V7-975403. A cutout will be provided in the outer sidewall to allow proper field of view. The panel will be installed inboard enough to allow the star tracker an adequate field of view and not protrude beyond the mold line. The IMU will be mounted on the inboard side of this panel, and will require thermal control louvers for cooling. The TV camera will also be installed on the inboard side of this panel. A mounting bracket will be used to set the camera inboard from the side wall to allow the proper field of view.

Chassis

Chassis assemblies will be designed for the installation of signal conditioning modules, power distribution modules, and isolation diode modules. Modules will be installed in such a manner which will provide the least amplification during launch.

3.8.3 Electrical Interconnections

Wire Selection (Excluding Coax)

The wire selected for use in the baseline configuration is MIL-W-81381/8-10. This wire type consists of nickel plated copper conductors with two wraps of fluorinated ethylene-propylene (FEP) coated Kapton and a polyimide enamel top coat. The wire is light weight, possesses small outside diameters, and has a high nick resistance.

Wire Harnesses

Wire harnesses will be continuous between panels as shown on V7-975403, Sheet 3. Clamps to secure the harnesses will be installed on the panels, or on the frames in the area where panels do not exist. In the forward skirt area the coax cables will be routed on the forward end of the equipment panels, and the remaining wire will be routed on the aft end. This approach will provide some measure of signal separation while minimizing coax cable handling during maintenance, repair and refurbishment.

In general, the avionics harnesses will be a round configuration, with wire in a parallel lay. Exceptions to this will be the harness crossing the main engine gimbal area and the EOS/Tug attachment pivot area. For those two areas, helical lay wire will be used for flexibility. In as much as helical layed wire requires more wire than parallel layed wire, the extensive use of parallel lay wire will be weight effective.

Conduits

The forward and aft skirt wire interconnection will be accomplished by routing the wires through two conduits which extend from the forward skirt to the aft skirt. The conduits will be routed through the ring frames. A breakout in the conduits will exist between the vehicle LOX and LH_2 tanks to allow the passage of the wiring from the tanks to the aft panels. The conduits will be approximately two inches in diameter and will be constructed of thin wall aluminum.

Table 3.8-2 summarizes quantities and weights of the electrical interconnection components.

Table 3.8-2. Electrical Interconnection Components Summary

Component	Qty.	Unit Wt. (lbs)	Total Wt. (lbs)
Forward-to-Aft Conduit	2	5	10.0
Connectors	*500	0.125	62.5
Wire & Wire Harness Clamps	-	-	125.0

* Quantity based on estimation .

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Space Division
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3.3-1 Chiarappa, D., Analysis and Design of Space Vehicle Flight Control Systems, Volume III - Rendezvous and Docking. NASA CR-827 (July 1967).

3.3-2 Orbit-to-Orbit Shuttle (Chemical) Feasibility Study. SAMSO-TR-71-238, Vol. IVB (Oct. 1971).

3.3-3 "IES Lighting Handbook," 4th Edition, 1968.

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4.0 THERMAL CONTROL SUBSYSTEMS

The systems for controlling the temperature of the TVC hydraulics, the Fuel Cell and the Avionics were designed independent of each other. A weight savings may be realized by integrating all thermal requirements and it is recommended that further study be authorized in this area. For instance, the heat rejected (venting of steam) by the fuel cell, the APS heat exchangers and turbopumps possibly could be used as the heat source for the fuel cell reactants, fuel cell water storage systems and the TVC hydraulics. The complications of transferring the heat from one location to another and the time phasing requirement would be the main considerations in determining the practicality of an integrated thermal control system.

4.1 THRUST VECTOR CONTROL

Thermal control of each (two) actuator will be achieved through use of electric heaters and an insulation jacket. The jacket is to be of molded foam material with a white outside surface. For access the jacket is to be clamped in two pieces on each actuator. Interconnecting lines are permanently insulated in a similar manner. Heaters for the lines are not necessary, provided periodic circulation of hydraulic fluid, as is now planned, remains feasible. Thermal switches in redundant pairs at temperature sensitive locations will be used to control the heaters and fluid circulation.

The thermal switches will be set to turn on the heaters when the fluid temperature decreases to -10 deg. F and turn off when the temperature increases +10 deg. F. After the heaters are switched off, the electric motor driven pump will be energized to equalize temperatures throughout the system. For the worst case vehicle attitude (nose-to-sun) the heaters will be on approximately 20 minutes out of each hour. The heaters will be rated at 200 watts; thus, the average power consumption will be approximately 70 watts.

4.2 FUEL CELL

4.2.1 Introduction

This section of the report presents the results of the study performed to design a cooling system that will reject sufficient heat from the fuel cell condenser to maintain the condenser outlet water temperature within the limits specified in the data received from Pratt & Whitney. The design heat load was based upon a specific operating mode for the fuel cell (initiation of open cycle cooling at 1100 watts power output). Recent discussions with P & W personnel indicate open cycle cooling could be initiated at a lower power output; thus, decreasing the heat rejection requirement for the external thermal control system. This method of operation would require additional water for the cell. However, the weight increase due to added water is expected to be less than the weight saving due to the smaller thermal control system.

4.2.2 Study Criteria

In order to meet the objectives of this study to select, define, and locate a thermal control system it was necessary to comply with the applicable ground rules as set forth in the study directive. These ground rules are listed as follows:

a. Tug mission life is 20 mission. Refurbishment of subsystems after each mission is acceptable.

b. On-orbit stay time shall be six days unattached to the Shuttle.

c. Power to Tug during ascent and descent shall be Shuttle furnished.

d. Thermal Control System (TCS) of the Tug shall not provide thermal control to payload systems.

e. TCS of the Tug shall not require selective orientation in orbit to perform the thermal control functions.

f. Active thermal control to Tug systems is not required from the orbiter.

4.2.3 Design Requirements

a. Environmental Requirements

The TCS as a part of the Tug vehicle will be subjected to the pressure, temperature, vibration, and acoustic environment of the Space Shuttle Cargo Bay during launch and during re-entry and return to earth.

(1) Maximum acceleration loads will be $\pm$ 3 g

(2) Internal cargo bay wall temperature will vary from -100°F to +200°F.

b. Performance Requirements

(1) The fuel cell TCS must be designed to reject the heat versus power output as shown in Figure 4.2-1. Maximum heat rejection is 2500 Btu/HR at a power output of 1.1 KW.

(2) The water vapor entering the condenser shall be a maximum of 183°F.

(3) The liquid water temperature at the condenser outlet shall be a maximum of 140°F.

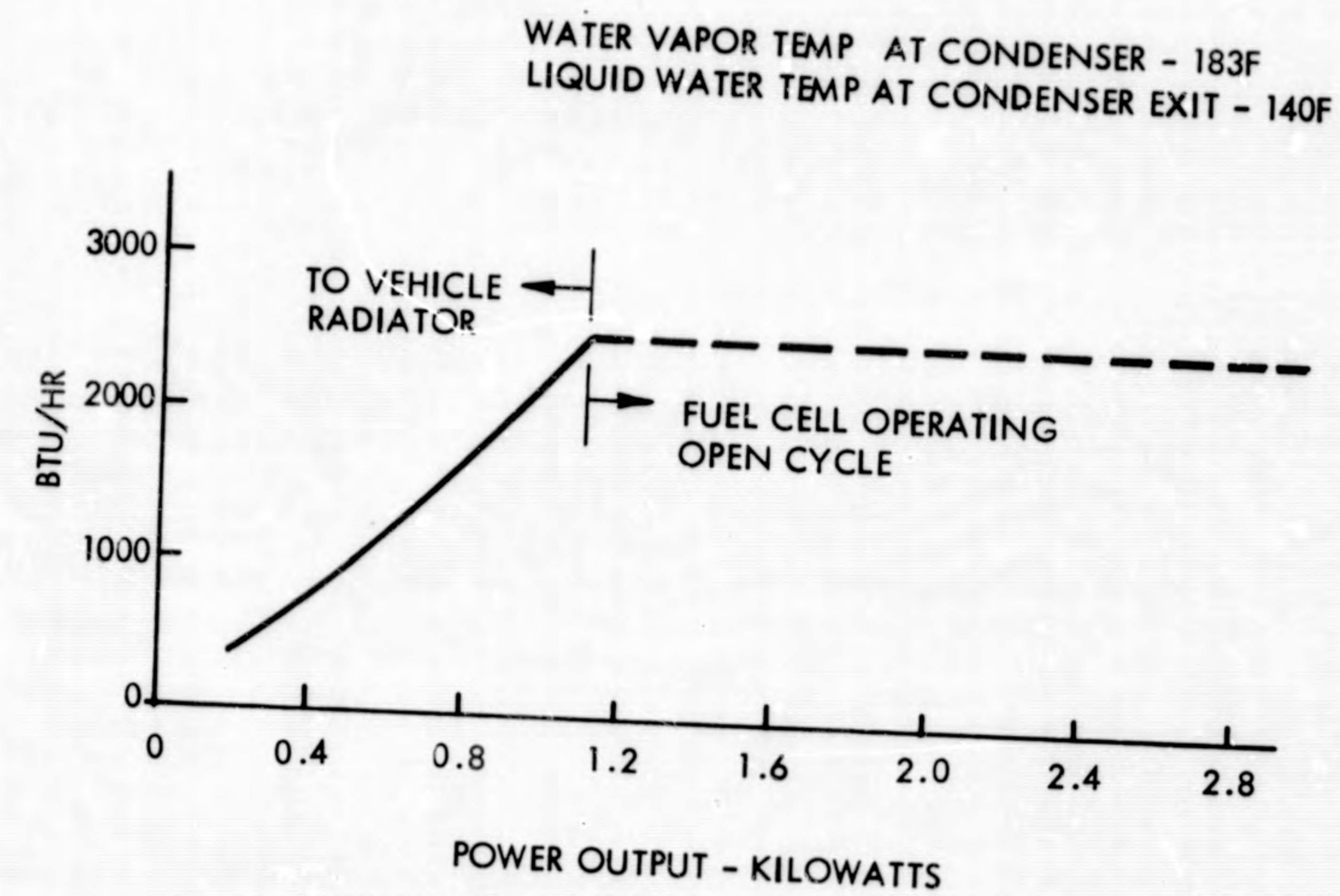

Figure 4.2-1 Fuel Cell Heat Rejection

4.2.4 System Selection

Two thermal control concepts were evaluated to meet the design and performance requirements for the Tug vehicle. They are as follows:

a. Direct coupling of fluid loop system to the spacecraft radiators (active system), Reference Figure 4.2-2.

b. Heat pipes (passive system), Reference Figure 4.2-3.

The pumped fluid loop system was selected as the baseline design for this application. The liquid circulating system has been used extensively in other space applications (Gemini, Apollo CSM) and has an excellent reliability record. The ability to achieve close tolerance control with a bypass thermostatic control which diverts fluid to the radiator, provides the required component temperatures. Confidence in this system and its ability to closely control temperatures has led to its choice as the baseline system. Thermal control systems comparative data are presented in Table 4.2-1. System weight data are presented in Table 4.2-2.

4.2.5 System Description (Baseline Concept)

The fuel cell condenser and TCS schematics are shown in Figure 4.2-2. The installation details are shown in Figures 4.2-4 and 4.2-5. The fuel cell power plant converts hydrogen and oxygen to electric power upon demand. Potable water and waste heat are produced as by-products of the power production process. The heat and water management subsystem in the power plant is a circulating water loop which provides thermal control and by-product water removal. While operating between the specified power range of 200 to 1000 watts, the power plant is cooled by the spacecraft radiators. When operating above 1100 watts, the power plant cooling is supplemented by cooling water supplied from the fuel cell water storage system. This water which cools the power plant by boiling in the fuel cell stack is automatically vented overboard together with a portion of the product water.

The fuel cell Thermal Control System (TCS) consists of a Freon-21 pump and accumulator, four space radiators connected in series and the required tubing and fittings to complete the loop and connect to the fuel cell condenser or subcooler. The condenser outlet temperature is maintained at +140°F by a temperature sensor and a subcooler temperature control valve within the fuel cell which is positioned to control the flow of coolant fluid to the condenser. The Freon-21 pump maintains continuous flow through the radiator system.

4.2.6 Subsystem Performance (Baseline Concept)

The heat rejection rate at a power output of 1.1 KW is 2500 Btu/HR. which requires a Freon flow rate through the condenser of 250 pounds per hours. The pump power requirements at this flow rate with an assumed system ΔP of 10 psi and a pump efficiency of 13 percent is 12 watts. The pump is driven by 115 volt, 3 Ø 400 cps AC electrical motor. A static inverter will be required to change the fuel cell DC output to AC for the electrical motor. The Freon-21 accumulater volume will be approximately 500 cubic inches. A typical pump schematic is shown in Figure 4.2-6.

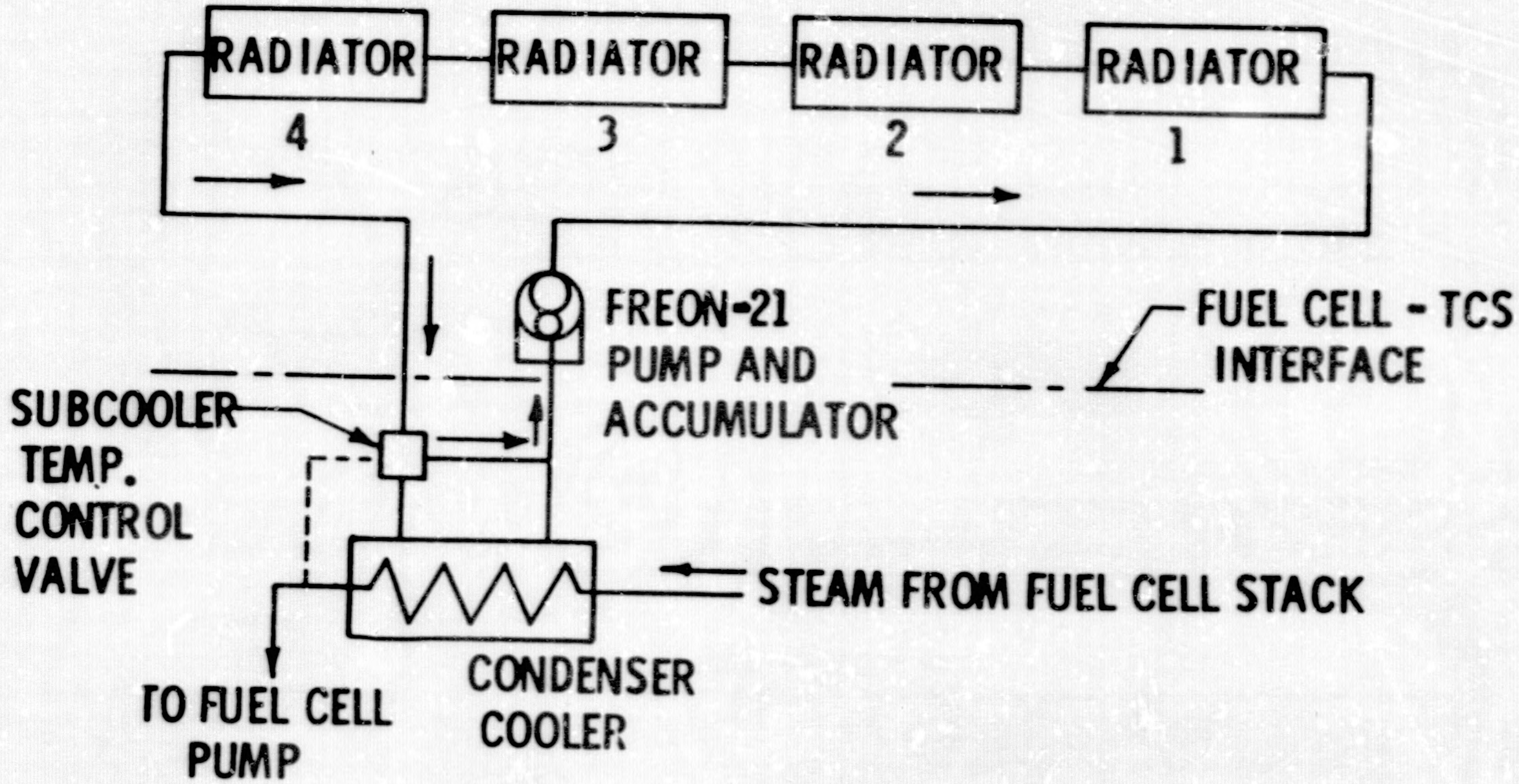

Figure 4.2-2 Fuel Cell Thermal Control System Schematic

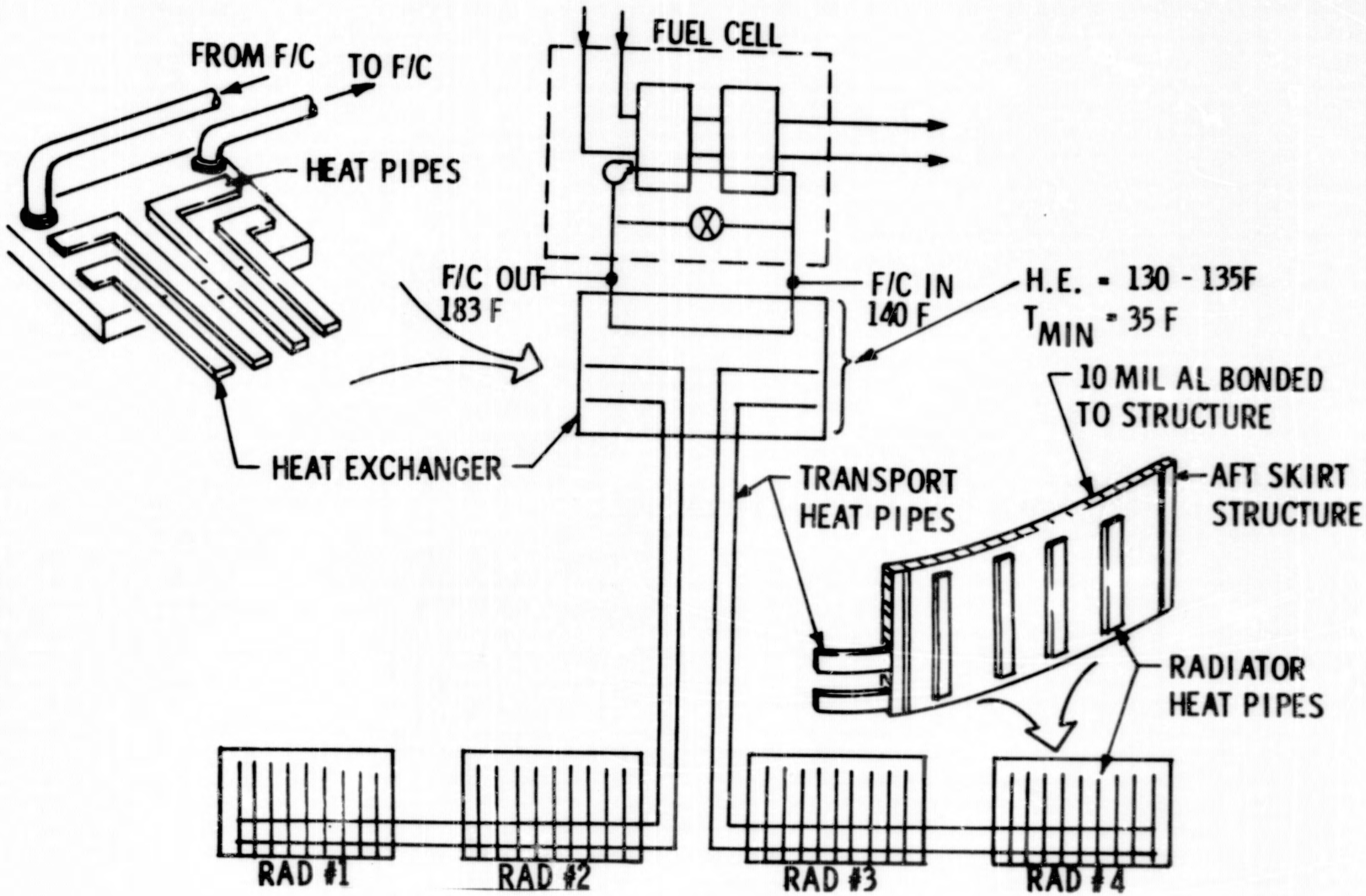

Figure 4.2-3 Fuel Cell Heat Pipe Thermal Control System Schematic

Table 4.2-1 Thermal Control Comparative Data

Option	Relative Weight	Operational Characteristics: Advantages	Operational Characteristics: Disadvantages
Fluid Loop	0	Good temperature control. No vehicle orientation required. Used on other space vehicles, (Apollo CSM)	Potential source of leakage. Moving parts subject to wear such as pumps, valves, etc.
Heat Pipes	-41%	No electrical power required. Light weight. Inherent reliability. Passive system.	No proven flight history. Requires system development.

Table 4.2-2. Fuel Cell Thermal Control

Weight Summary

1. Fluid System

Component	Weight (Estimated)
Radiator Panel 0.5 #/FT^2	22
Freon Pump and Accumulator	12
Freon Fluid	45
Tubing and fittings	5
	84

2. Passive System

Total system weight 35

The heat generated by the fuel cell will be rejected to space by means of four radiator panels as shown on Figure 4.2-2. Each panel is approximately 11.2 ft^2 in area. They are equally spaced around the aft skirt of the spacecraft. The fuel cell power plant is also located on the aft skirt. This location minimizes line length to the radiators and also provides access for servicing and maintenance requirements. The radiator panels will be coated with Z-93 (zinc oxide pigmented potassium silicate) or equivalent which produces good thermal properties of high emmittance (ϵ) and low absorptance (α_s). An $\alpha s/\epsilon$ ratio of 0.3 was used in sizing the radiators. The solar absorptivity

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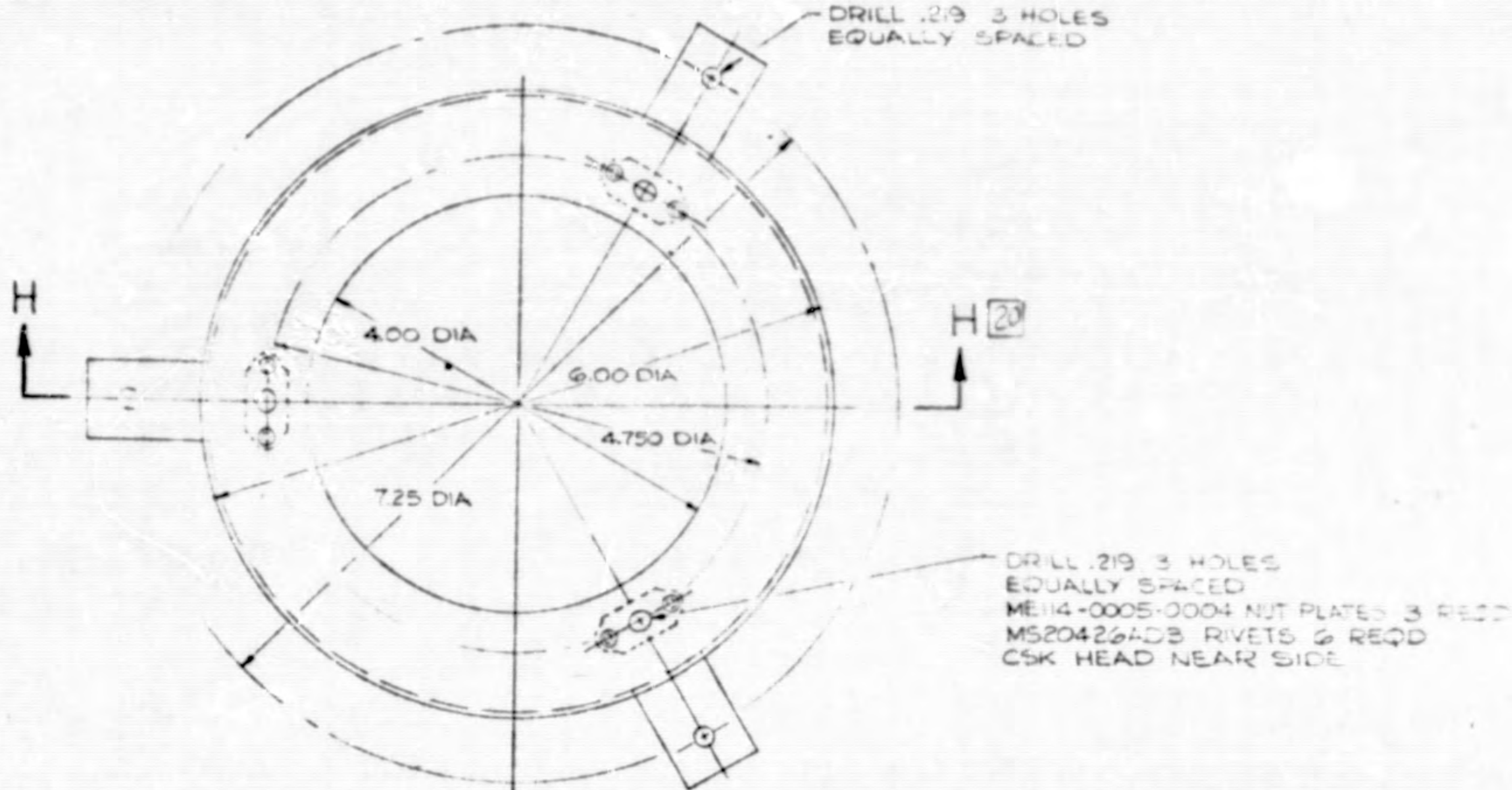
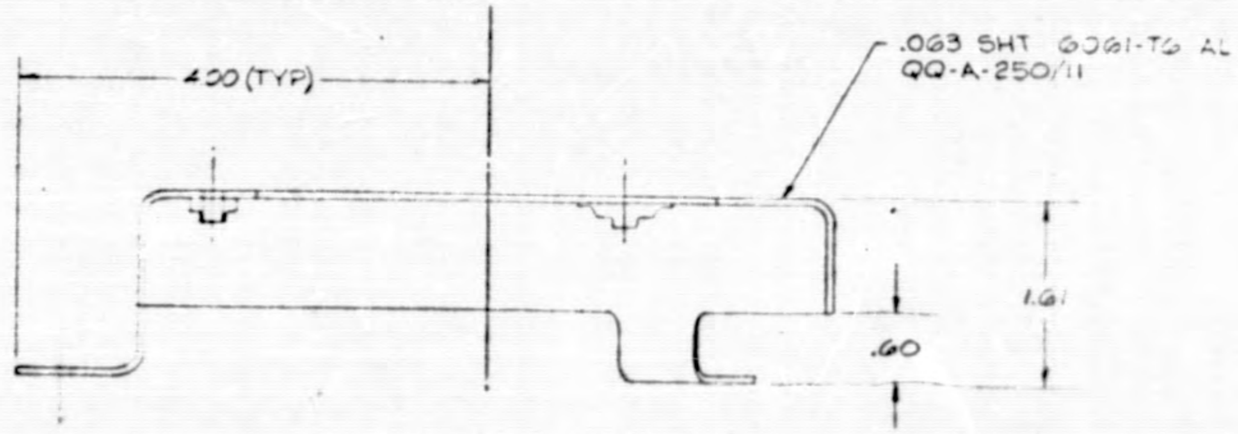


DETAIL G 17
MANIFOLD SUPPORT BASE
SCALE 1/1



SECT H-H 19
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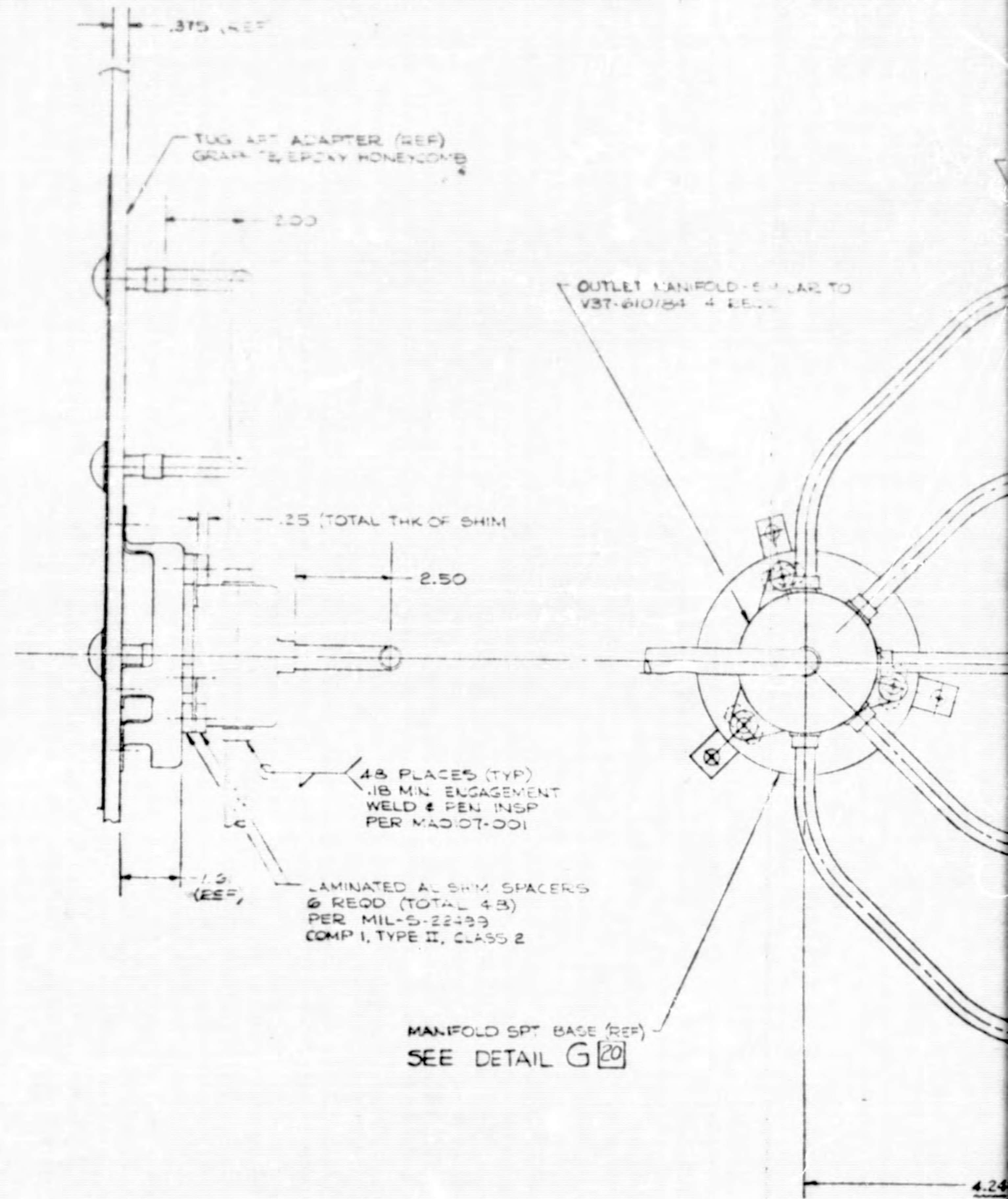

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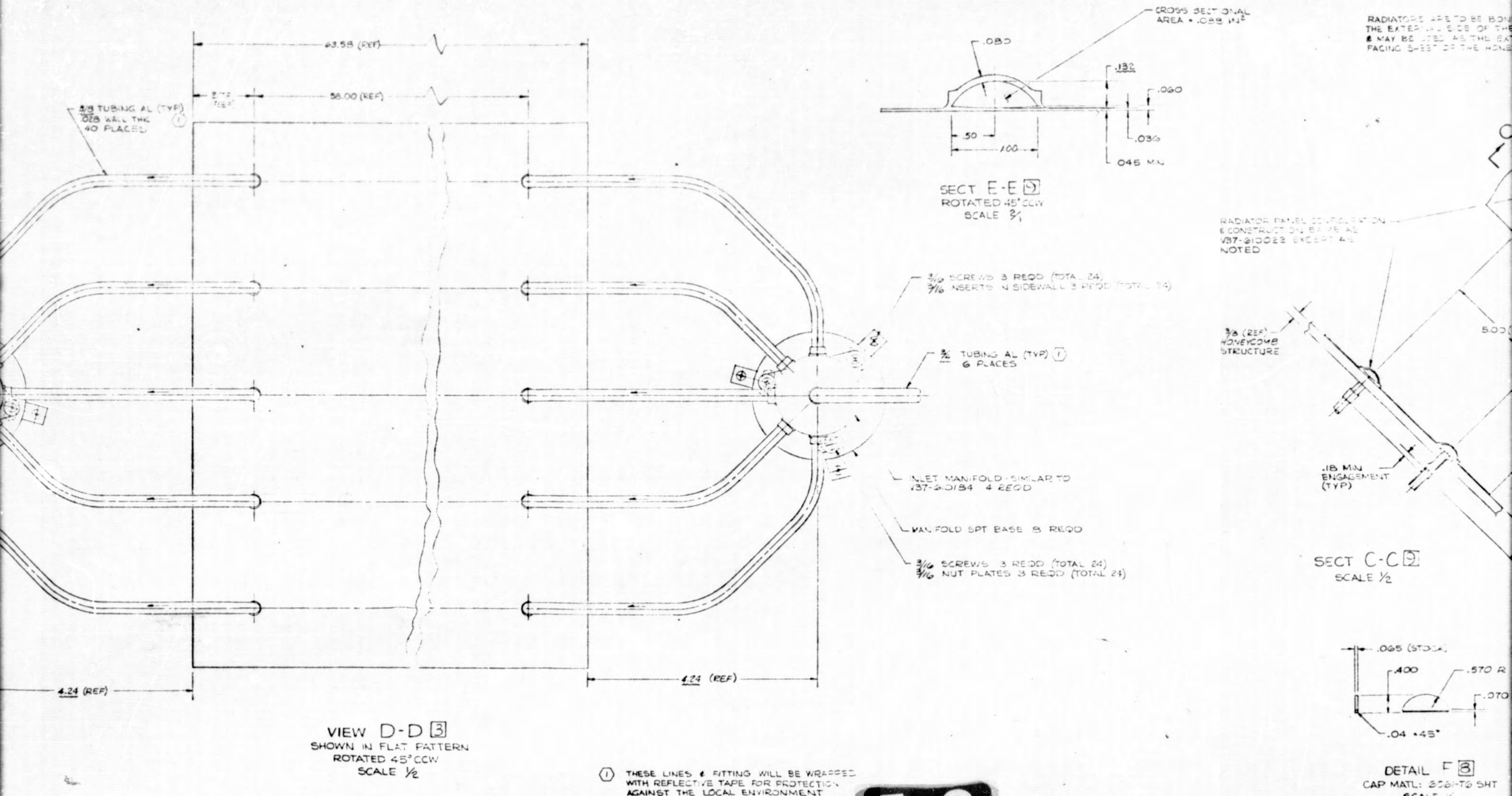
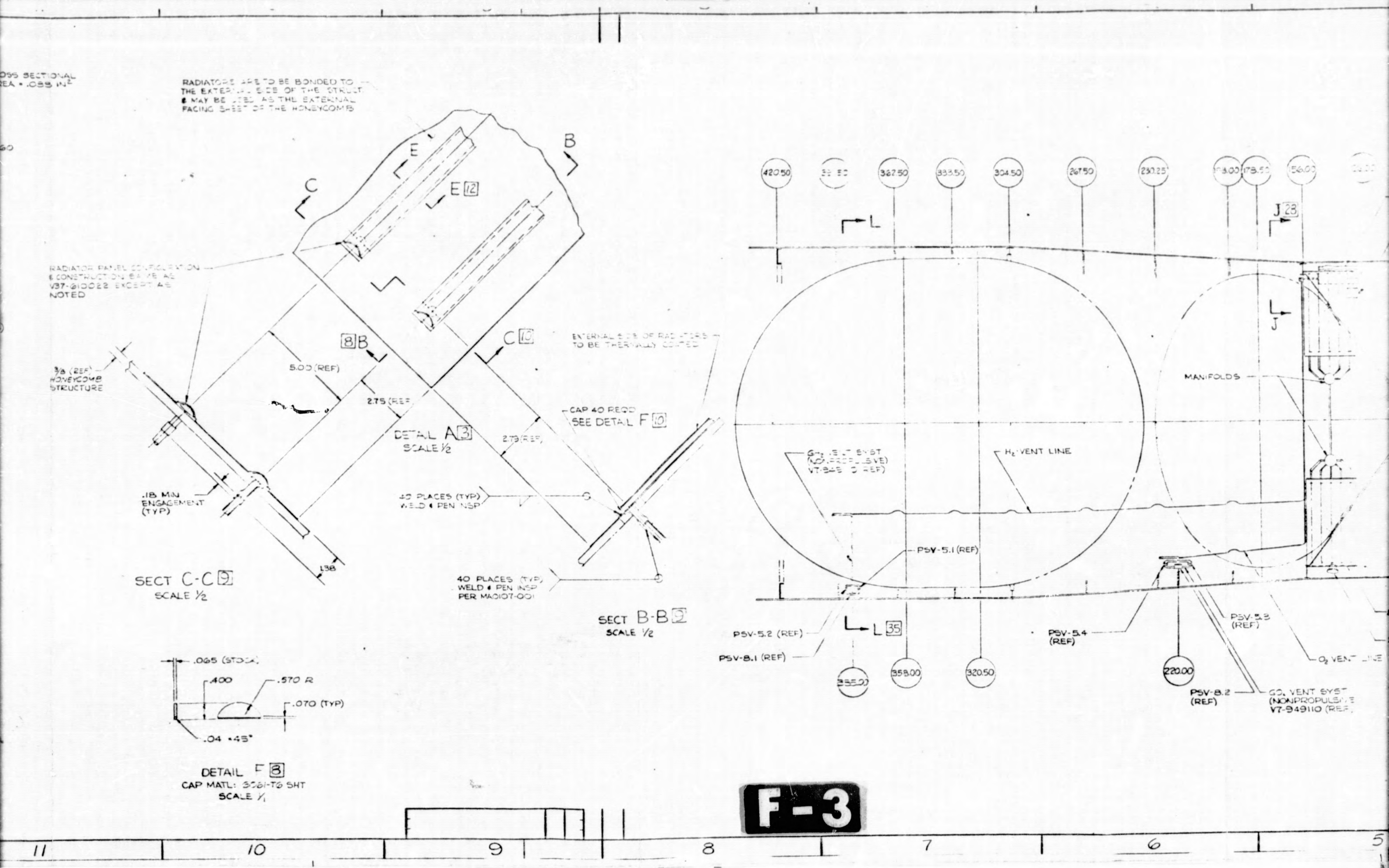


THESE LINES & FITTING WILL BE WRAPPED WITH REFLECTIVE TAPE FOR PROTECTION AGAINST THE LOCAL ENVIRONMENT

F-2

16 | 15 | 14 | 13 | 12 | 11 | 10


OSS SECTIONAL
REA = .035 IN²
RADIATORS ARE TO BE BONDED TO THE EXTERNAL SIDE OF THE STRUCT. & MAY BE USED AS THE EXTERNAL FACING SHEET OF THE HONEYCOMB
RADIATOR PANEL CONFIGURATION & CONSTRUCTION SAME AS V37-610022 EXCEPT AS NOTED
3/8 (REF) HONEYCOMB STRUCTURE
.18 MIN ENGAGEMENT (TYP)
5.00 (REF)
2.75 (REF)
2.79 (REF)
1.38
DETAIL A 3
SCALE 1/2
SECT C-C 9
SCALE 1/2
EXTERNAL SIDE OF RADIATORS TO BE THERMALLY COATED
CAP 40 REQD
SEE DETAIL F 10
40 PLACES (TYP)
WELD & PEN INSP
40 PLACES (TYP)
WELD & PEN INSP
PER MA0107-001
SECT B-B 9
SCALE 1/2
.065 (STOCK)
.400
.570 R
.070 (TYP)
.04 × 45°
DETAIL F 8
CAP MATL: 6061-T6 SHT
SCALE 1/1
420.50
367.50
333.50
304.50
267.50
230.25
178.00
178.50
56.00
GH₂ VENT SYST (NONPROPULSIVE) V7-949110 (REF)
H₂ VENT LINE
MANIFOLDS
PSV-5.1 (REF)
PSV-5.2 (REF)
PSV-8.1 (REF)
PSV-5.4 (REF)
PSV-5.3 (REF)
PSV-8.2 (REF)
GO₂ VENT SYST (NONPROPULSIVE) V7-949110 (REF)
O₂ VENT LINE
355.00
358.00
320.50
220.00
J 28
L 35
F-3
11
10
9
8
7
6
5

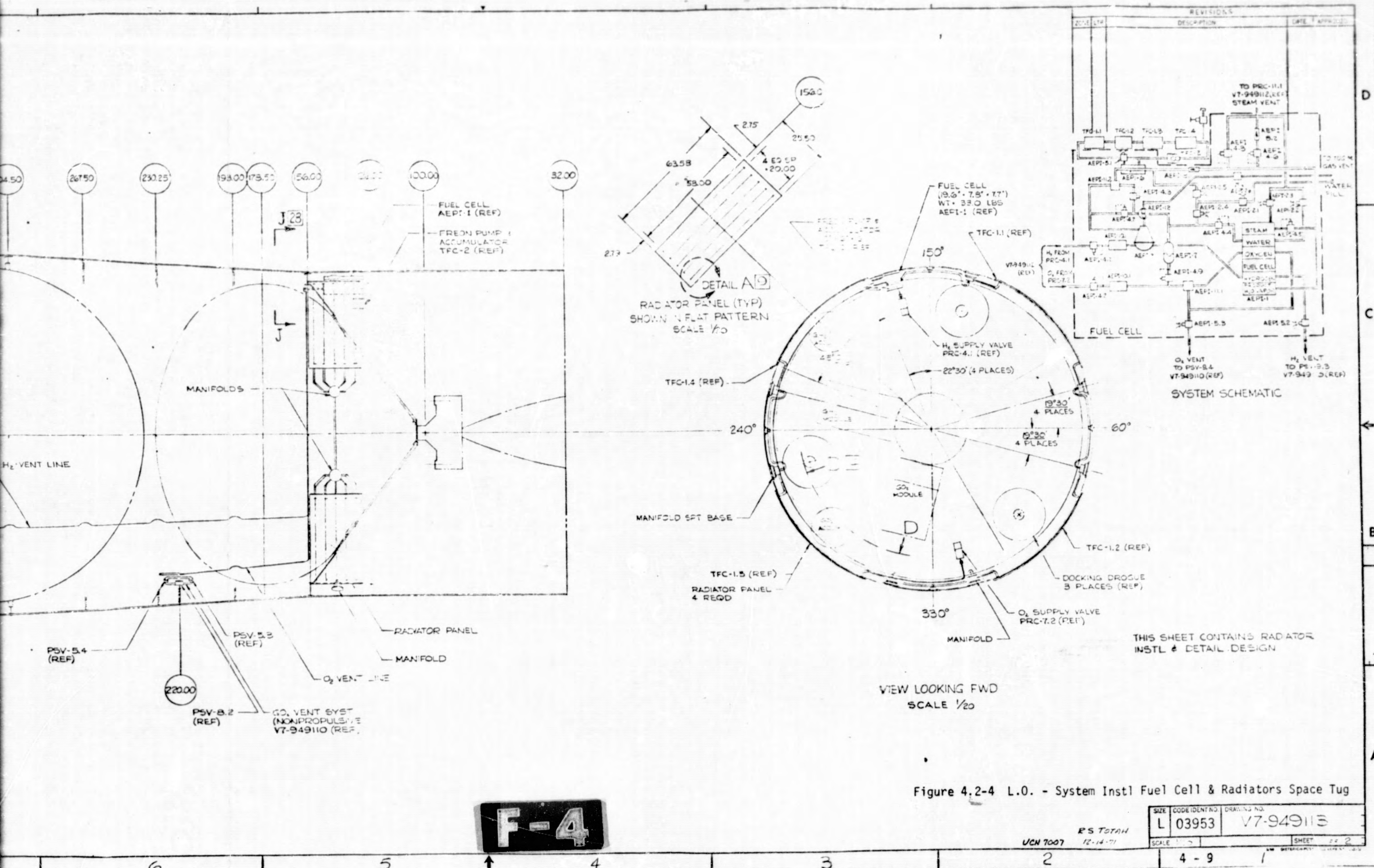


Figure 4.2-4 L.O. - System Instl Fuel Cell & Radiators Space Tug

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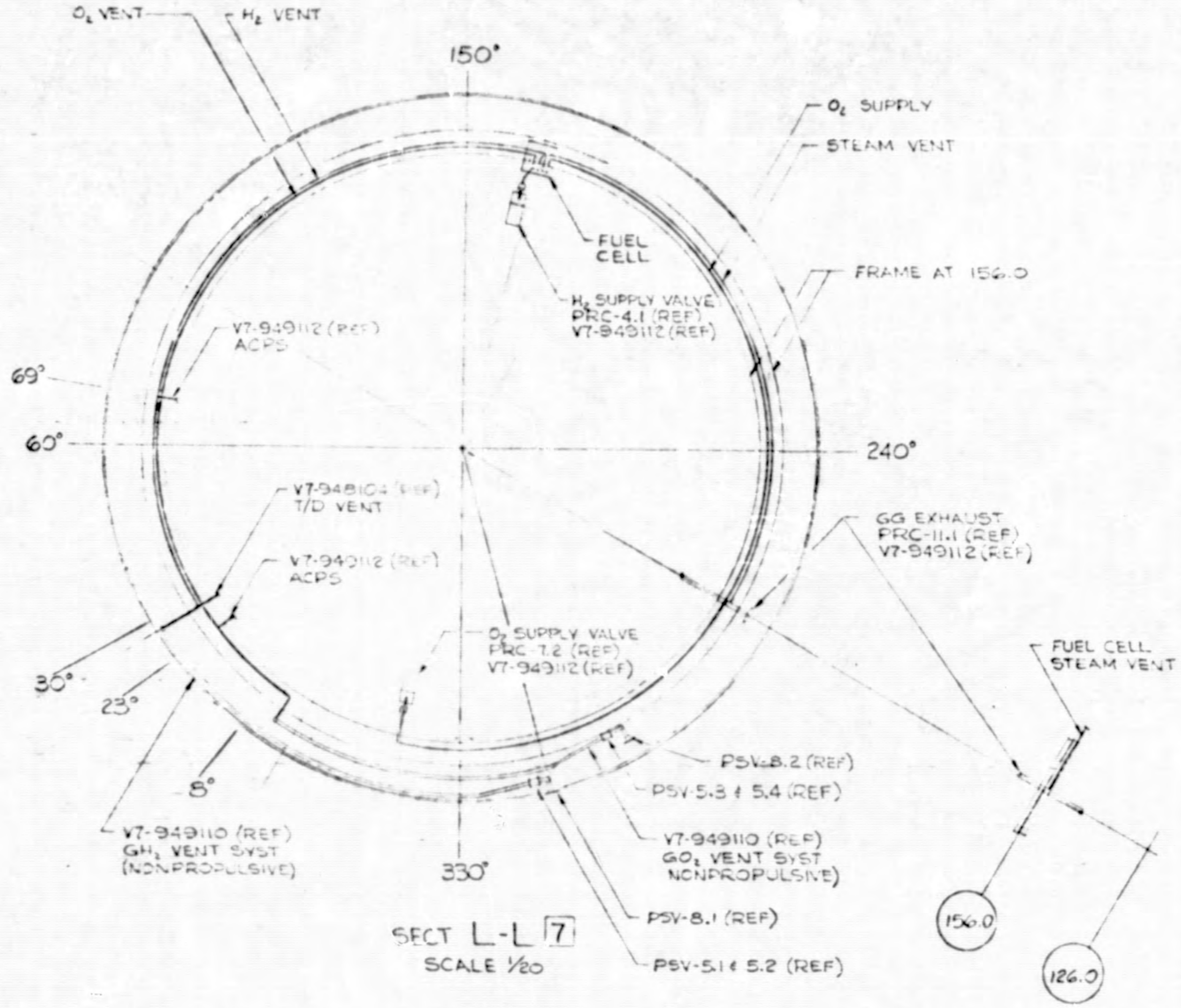
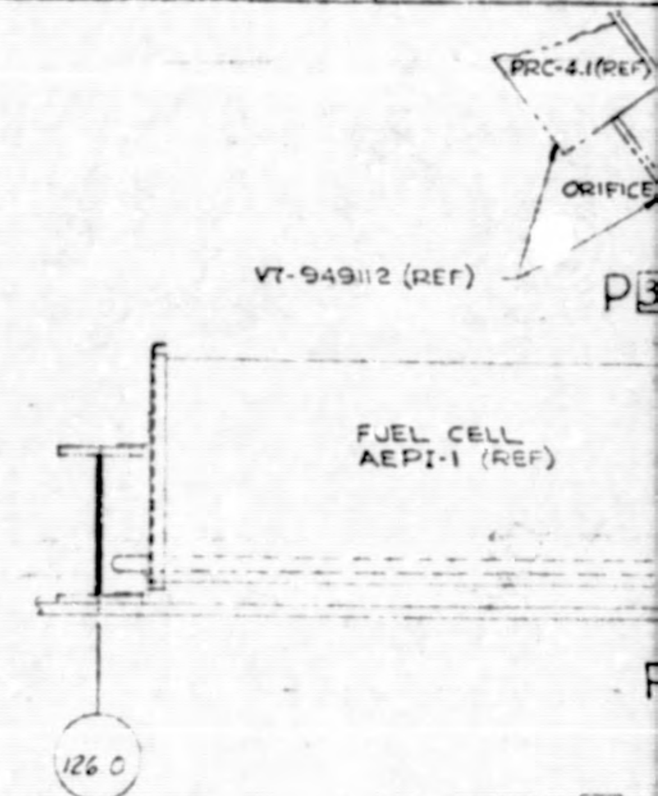
37 36 35 34 33 32



SECT L-L [7]
SCALE 1/20



SECT M-M [29]
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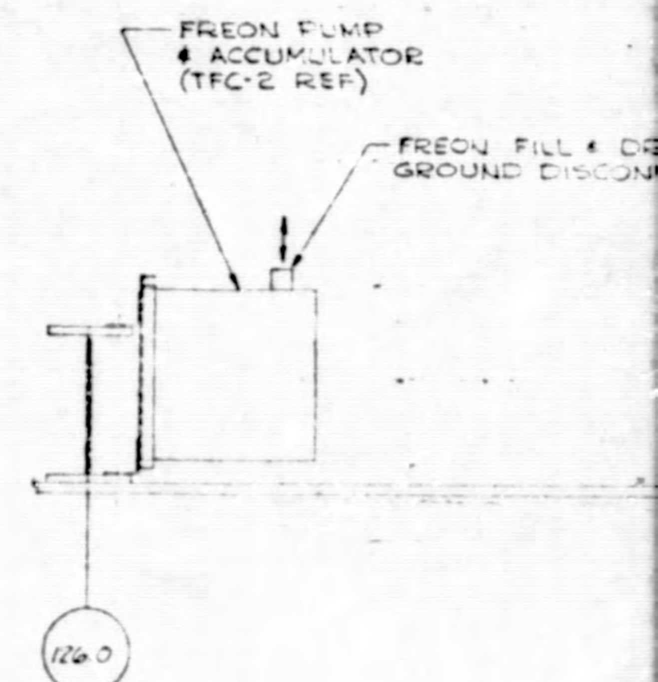
SCALE 1/4



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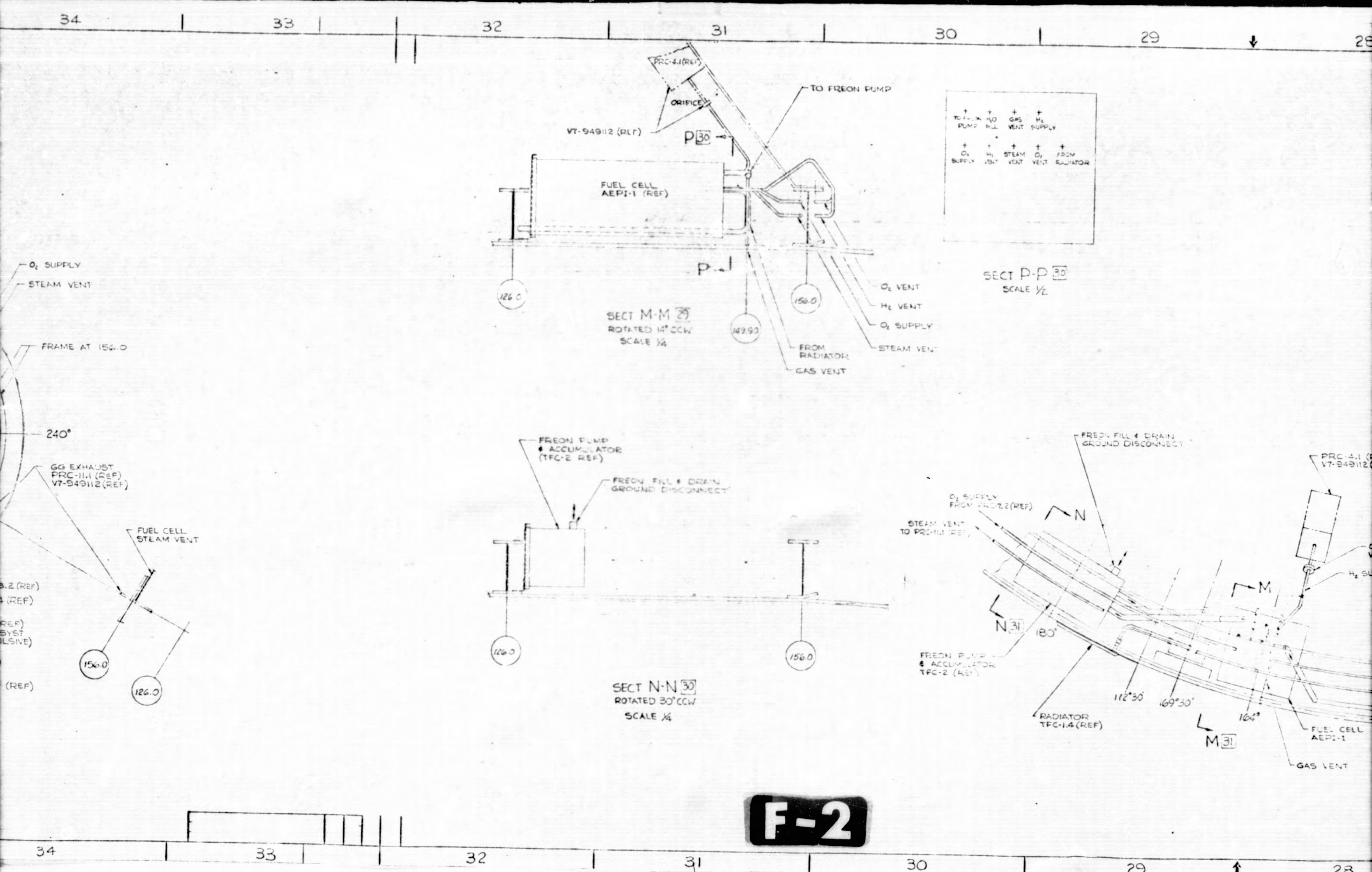
SCALE 1/4

F-1

37 36 35 34 33 32



34 33 32 31 30 29 28
PRC-4.1 (REF)
ORIFICE
V7-949112 (REF)
TO FREON PUMP
P 30
FUEL CELL
AEPI-1 (REF)
P
126.0
156.0
149.93
SECT M-M 29
ROTATED 14° CCW
SCALE 1/4
O2 VENT
H2 VENT
O2 SUPPLY
STEAM VENT
FROM RADIATOR
GAS VENT
TO FREON PUMP
H2O FILL
GAS VENT
H2 SUPPLY
O2 SUPPLY
H2 VENT
STEAM VENT
O2 VENT
FROM RADIATOR
SECT P-P 30
SCALE 1/2
O2 SUPPLY
STEAM VENT
FRAME AT 156.0
240°
GG EXHAUST
PRC-11.1 (REF)
V7-949112 (REF)
FUEL CELL
STEAM VENT
(REF)
156.0
126.0
FREON PUMP
& ACCUMULATOR
(TFC-2 REF)
FREON FILL & DRAIN
GROUND DISCONNECT
126.0
156.0
SECT N-N 30
ROTATED 30° CCW
SCALE 1/4
FREON FILL & DRAIN
GROUND DISCONNECT
PRC-4.1
V7-949112
O2 SUPPLY
STEAM VENT
N
M
N 31
180°
FREON PUMP
& ACCUMULATOR
TFC-2 (REF)
112°30'
169°30'
164°
RADIATOR
TFC-1.4 (REF)
M 31
FUEL CELL
AEPI-1
GAS VENT
F-2
34 33 32 31 30 29 28

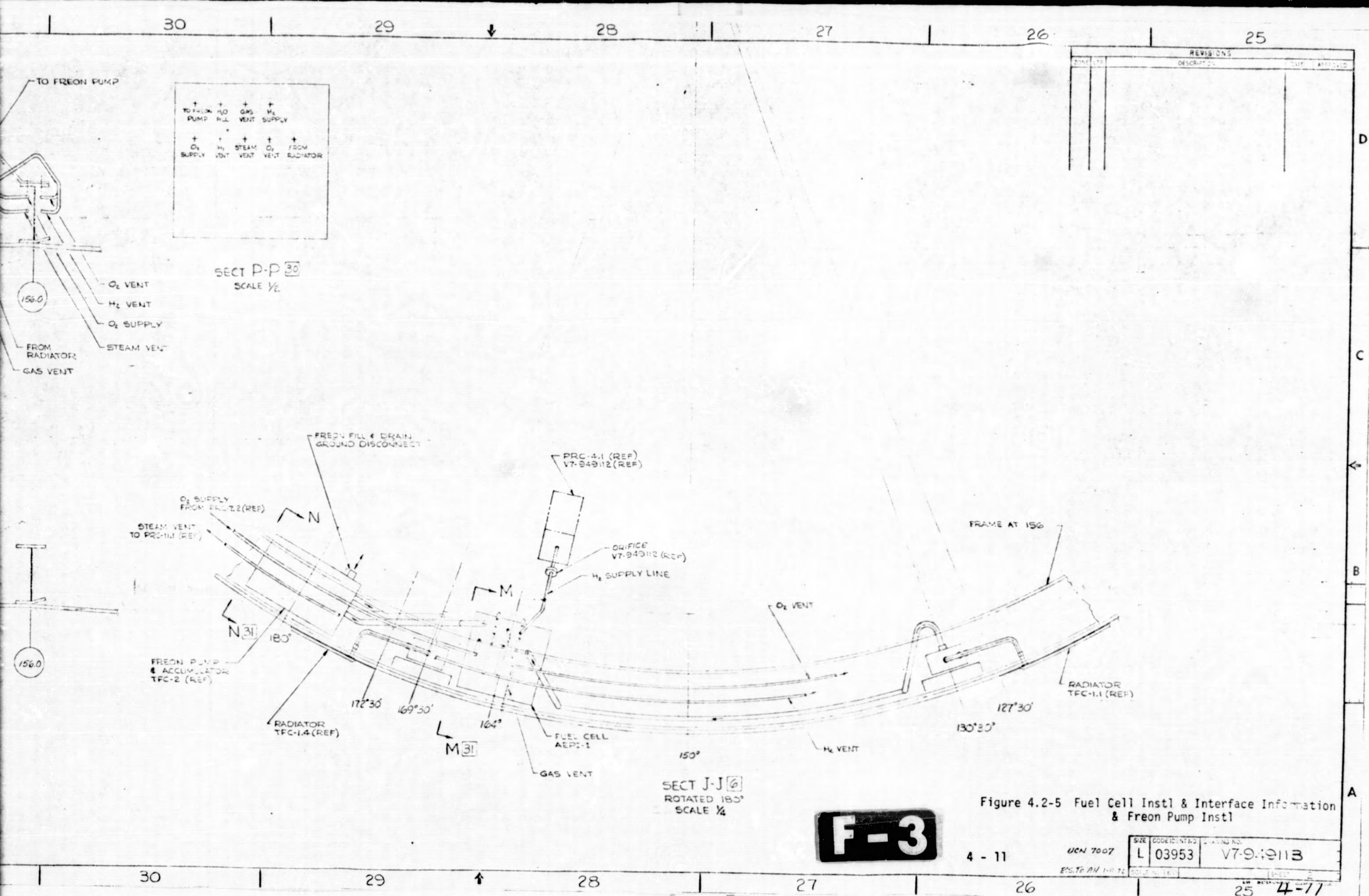


Figure 4.2-5 Fuel Cell Instl & Interface Information & Freon Pump Instl

F-3

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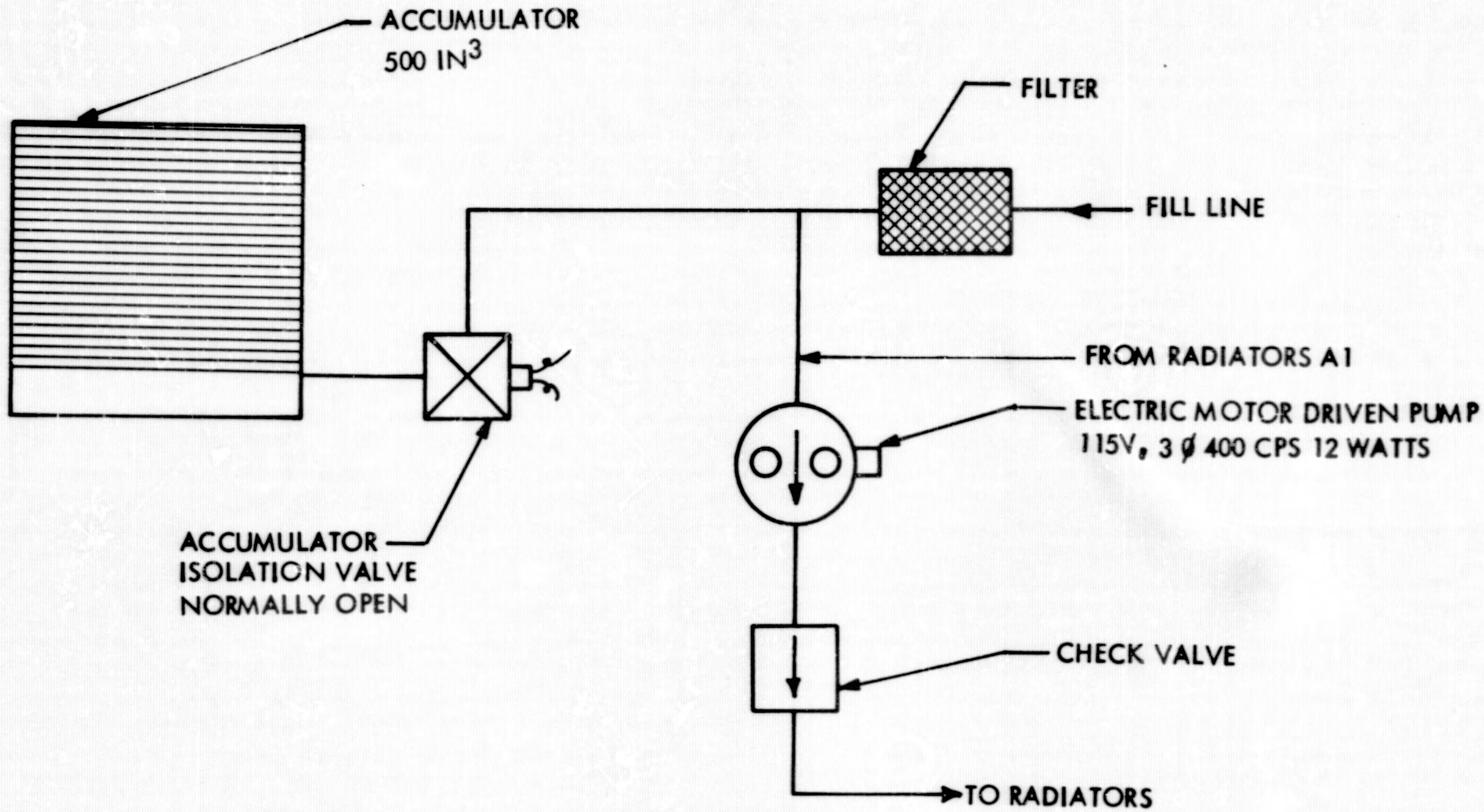

Figure 4.2-6 Typical Freon Pump and Accumulator Schematic

should not be allowed to degrade since the spacecraft is ground based which provides an opportunity for maintaining the desired surface characteristics. Each panel contains inlet and outlet manifolds and five tubes running circumferentially in parallel. The thickness of the radiator tubes are 0.080 inches as shown on Figure 4.2-2. This thickness is required to prevent meteoroid impact on the tube and possible loss of the fluid in the coolant system.

The temperature of the Freon coolant may reach +170°F when the fuel cell is operating at maximum load. Due to the relatively high vapor pressure of Freon-21, the system operating pressure must be a minimum of 150 psi. When the fuel cell is operating at minimum load, the heat rejection heat is approximately 500 Btu/HR. Added to this value is the continuous heat input from the Freon pump of 42 Btu/HR resulting in a total of 542 Btu/HR. Referring to Figure 4.2-7 which presents the worst possible cold case with the spacecraft in a geosynchronous orbit and oriented with the x axis parallel to the sun, it can be determined that the radiator temperature will be maintained at -140°F. This temperature requires a heat input of 15 Btu/HR/FT^2. Under these conditions, the temperature of the Freon will not reach the freezing point of -211°F.

In the event that the minimum heat rejection rate from the fuel cell is not maintained, it may be necessary to incorporate an electrical heater into the system.

The physical and chemical properties of Freon-21 are considered to be good for this application with the exception of water solubility. This property will require that the system be thoroughly purged and dried prior to charging with Freon.

4.2.7 Alternate Concept, Heat Pipe System

Fuel Cell Passive Thermal Control System

A passive thermal control system for the fuel cell utilizing heat pipes was investigated as an alternate to the use of a fluid loop cooling system. Heat pipes offer the advantages of inherent reliability, low power consumption, and potentially lower weight.

System Design Requirements

The design heat load for the fuel cell is 2500 Btu/Hr, corresponding to a power generation of 1.1 kilowatts. Above this power generation level, waste heat generated in excess of 2500 Btu/Hr is dumped in an open cycle cooling system which uses the internally generated water.

Under normal operating conditions, the fuel cell coolant enters the heat exchange interface with the external loop as a vapor at a temperature of 183°F and exits in a condensed state of 140 F. The external thermal control system must maintain the heat exchange interface at a temperature sufficient to cause the fuel cell coolant to undergo this temperature drop. The radiator must be sized to reject 2500 Btu/Hr under the hottest external environment. When the fuel cell is generating less than 1100 watts or when the radiators are exposed

WORST COLD CASE

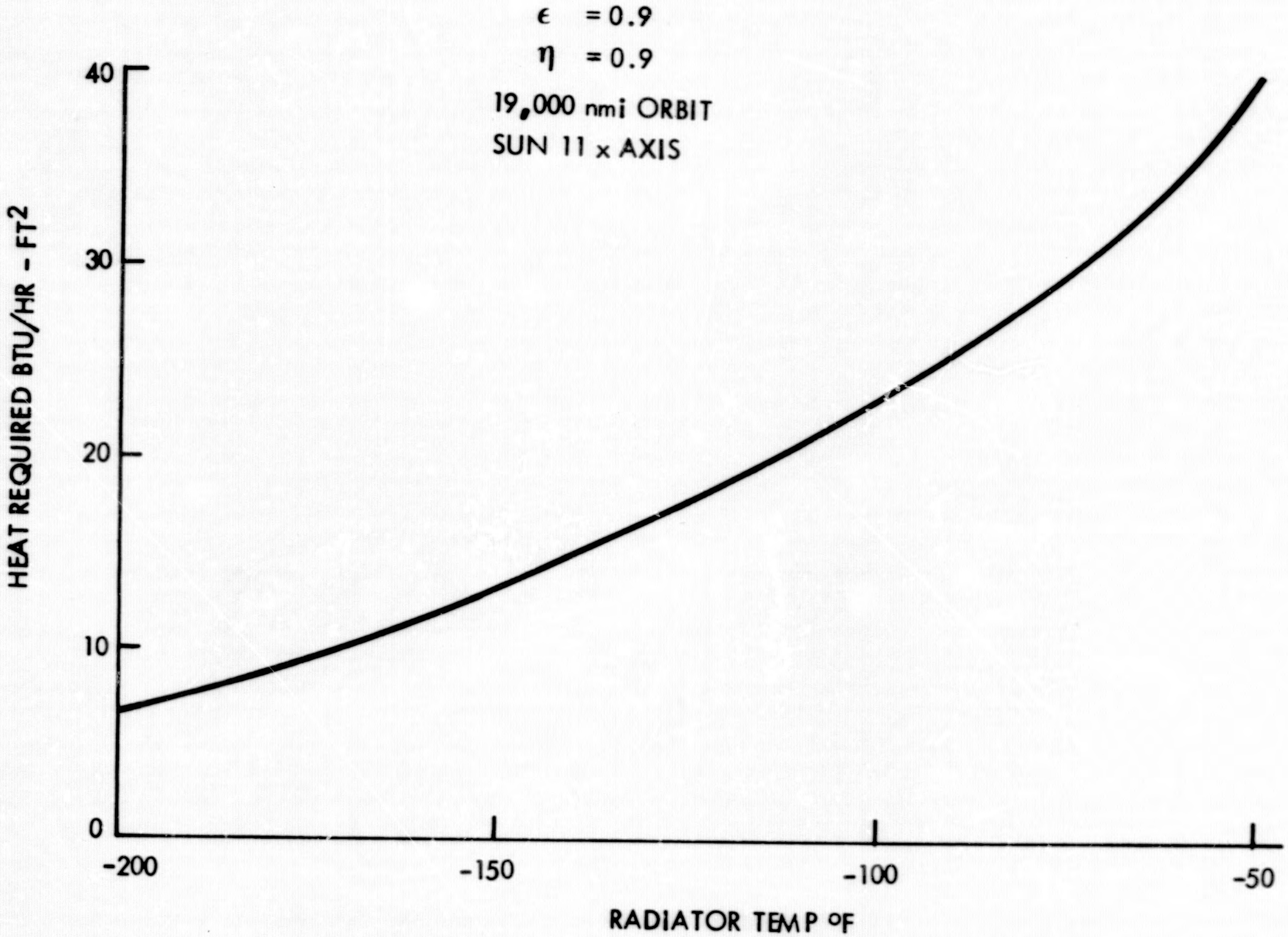

Figure 4.2-7 Heat Required to Maintain Radiator Temperature

to a cold environment, temperature control is maintained by a bypass system within the fuel cell. This bypass system is necessarily different than it would be with an active external heat transfer loop and would have to be designed to be compatible with the heat pipe system.

System Description

A schematic representation of the passive fuel cell thermal control system is shown in Figure 4.2-3. Heat pipes are used to transport heat from the heat exchange interface with the fuel cell coolant loop to the external radiator where it is rejected. The heat rejection or fin efficiency of the radiator is enhanced by the use of heat pipes which distribute the heat over the radiator surface so that the radiator can be assumed to consist of a number of fins. The radiators are mounted on the external skin of the vehicle and require thermal control coatings.

External Thermal Environment

Within the Tug mission envelope, the fuel cell radiators will be exposed to incident solar, albedo, and earth emitted heat fluxes. The thermal control system is designed such that it can provide thermal control within the range of the hottest and coldest external thermal environments. The coldest environment occurs in the high altitude orbit (19000 n.m.) with the nose of the vehicle pointed toward the sun. This results in essentially no incident environmental heating in the cylindrical side wall of the vehicle. The hot case is determined by examining orbit flight envelope and determining when the total heat flux on the vehicle is maximum, such that heat rejection from a circumferential radiator will be minimum. For the Tug, this occurs at the point directly between the sun and earth in a subsolar orbit (angle between earth-sun line and orbit plane is zero). The absorbed heat flux as a function of position circumferentially around the vehicle is shown in Figure 4.2-8 for various values of the solar absorptance to infrared emittance ratio (α_s/ϵ) of the radiator thermal coating. For a 360 degree circumferential radiator, the total absorbed heat flux is just the integral under the curve. For segmented radiators, the maximum heat flux is the integral over the circumferential angles subtended by the radiators. For this study, an α_s/ϵ of about 0.3 is assumed. This corresponds to a slightly degraded value for a standard radiator coating such as S-13G or Z-93.

Parametric Analyses and Trade Studies

Parametric analyses were conducted to determine the sensitivity of the weight and performance of the external radiator to the various design parameters. Studies included parametric radiator heat rejection, optimum heat pipe fin spacing, and external radiator location.

System Design

Results of the parametric analyses show that approximately 40 square feet of radiator are required using four panels. The resulting design consists of four panels as shown in Figure 4.2-9 each panel 2.5 ft x 4.0 ft. In order to minimize the length of the transport heat pipes, the radiators were located

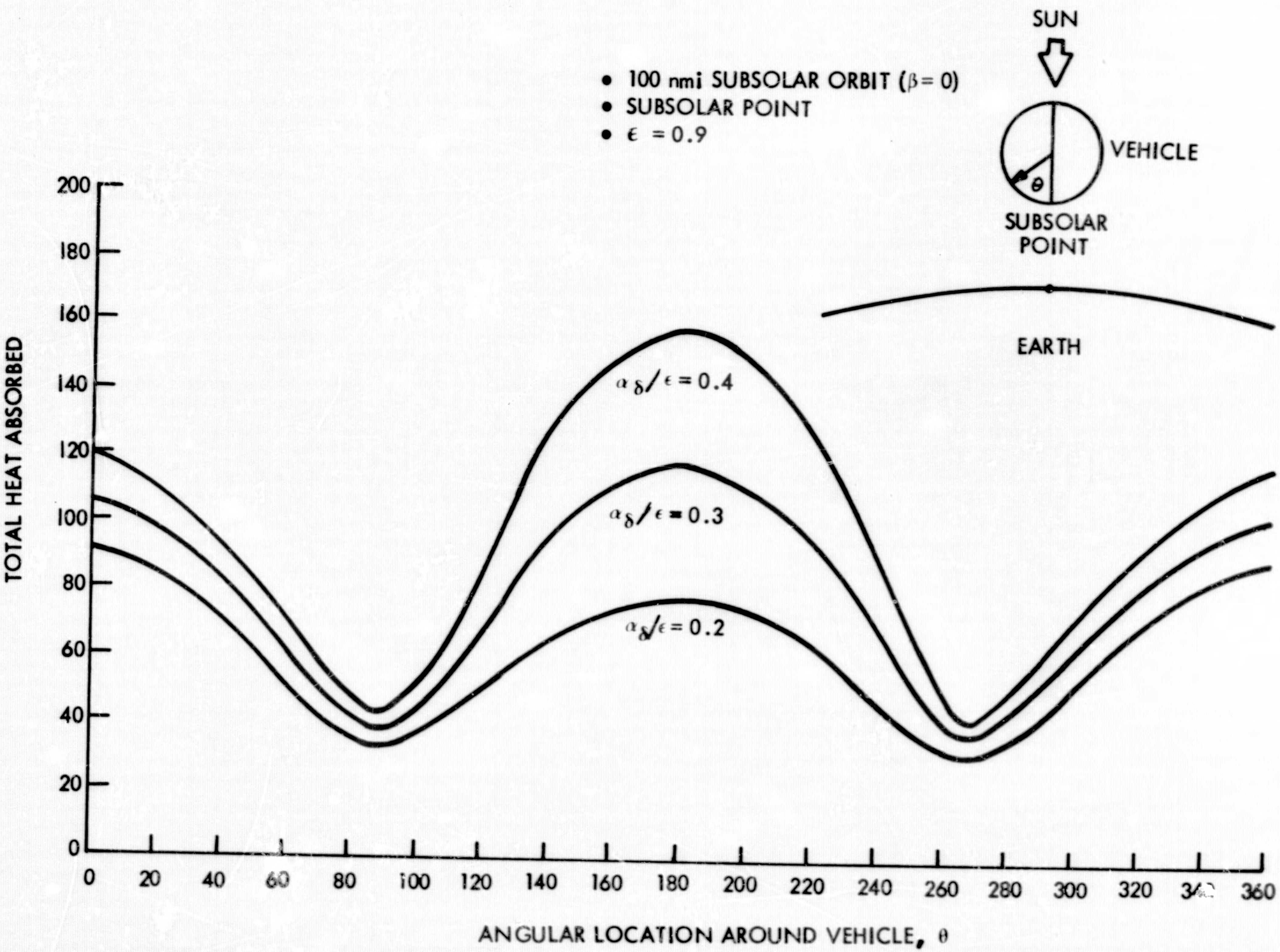

Figure 4.2-8 Total Absorbed Heat Flux vs α_s/ϵ

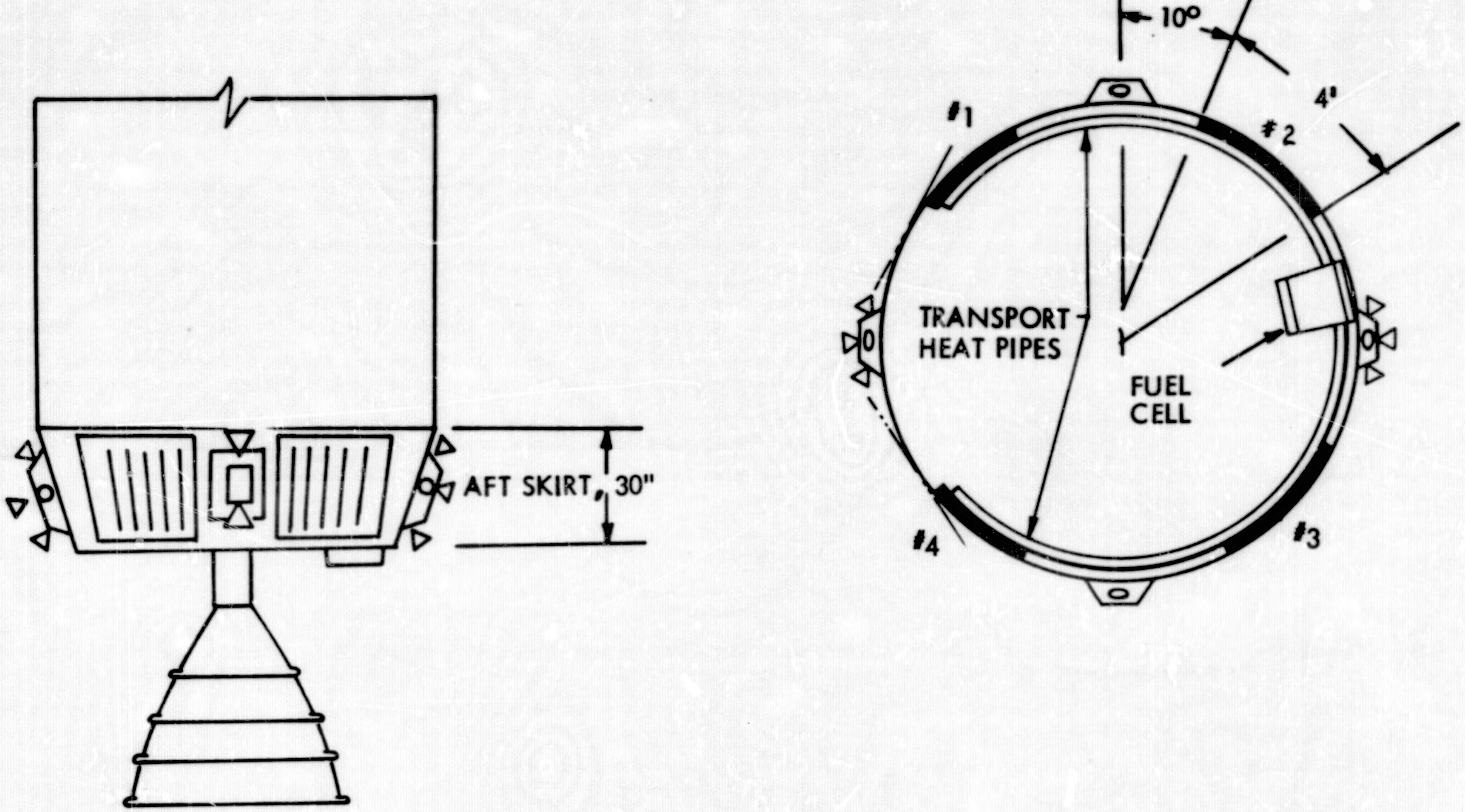

Figure 4.2-9 Fuel Cell Heat Pipe Radiator Configuration

on the aft skirt of the TUG. Also to minimize the effects of RCS plume impingement on the radiators, as shown in Figure 4.2-9, the radiators are located adjacent to the RCS housings without roll engines, and extend circumferentially around the vehicle. The radiator consists of 10 mils of aluminum bonded to the aft skirt structure. Verticle heat pipes are spaced every four inches to provide a fin efficiency of approximately 70 percent. The radiator heat pipes contain a non condensible gas reservoir at the top end of the pipes. This provides some degree of control which helps prevent the heat exchanger from getting cold enough to freeze the fuel cell water in a cold external environment. The fluid in the radiator heat pipes is Freon 21.

The transport heat pipes are designed to transfer 500 watts a distance of 20 feet in zero g. These are high capacity heat pipes and will probably use ammonia for the internal working fluid. These pipes interface with the fuel cell loop at one end, and with the external radiator at the other end. Four such pipes are used to provide redundancy. Two pipes go to each set of two radiators connected in series.

System Performance

The heat pipe system provides an efficient means for transporting waste heat from the fuel cell to the external radiator. The system is sized to reject 2500 Btu/Hr under worst hot case conditions. Under these circumstances, the temperature of the transport heat pipes varies from about 135 F at the heat exchange interface at the fuel cell to about 120 F at the external radiator where the transport heat pipes interface with the radiator heat pipes. The average fin root temperature is approximately 110 F and the fin efficiency is 0.71. Under nominal external heating conditions, the radiator and heat exchange interface temperatures are somewhat lower. Temperature control of the fuel cell coolant is provided by a bypass system within the fuel cell.

Under worst case cold conditions (no external heating), the variable conductance radiator heat pipes act to "shut off" some of the exposed radiator surface. This reduces the minimum heat required to maintain the heat exchange interface above the freezing point of the water in the fuel cell. Under these conditions, it is estimated that 200 to 500 watts of electrical power would have to be generated continuously to prevent the fuel cell water from freezing. This requirement might be reduced somewhat if the transport heat pipes were of the variable conductance type at the cost of some additional development.

4.2.9 Conclusions and Recommendations

A combination fluid coolant loop and radiator system has been selected as the baseline design for the fuel cell thermal control system. This approach provides good temperature control, uses available technology and does not impose orientation requirements on the spacecraft.

5.0 ELECTRICAL POWER SUBSYSTEMS

The Tug Electrical Power Generation Subsystem uses a single high performance fuel cell to supply the vehicle primary electrical needs during on-orbit operations. An active coolant loop provides thermal control for the fuel cell. The coolant loop interfaces with four radiators which reject the subsystem thermal waste to open space.

A single primary battery is included in the subsystem to provide a minimum of 30 minutes of power to the Tug backup stabilization system and payload deployment circuits in the event of primary power failure.

The electrical Power Conversion and Distribution Subsystem converts, through the use of a static inverter, DC power to AC power for specialized AC loads and distributes power to all using components on the vehicle. The distribution of power to the various loads is through the use of solid state switches controlled by the DMS.

5.1 FUEL CELL POWER GENERATION

The fuel cell is a high thermal efficiency (approx. 50% to 70%) device for the direct conversion of chemical energy into electricity. Two reactants (gaseous hydrogen and oxygen) supplied to the device are consumed in an electrochemical reaction producing electricity, water, and heat. Unlike electricity production by conventional heat cycles, the efficiency of energy production by fuel cells is not limited by a Carnot-type relation, thus permitting high thermal efficiencies. Unlike conventional primary cells or batteries, the fuel cell continues to supply current as long as reactants are fed in and products are removed.

5.1.1 Requirements

The requirements adhered to in the fuel cell design consisted of both specific and general groundrules and guidelines set forth in the NASA Tug Point Design Study Plan. Other operational requirements were established as a result of NR design analysis

Specific Groundrules and Guidelines

The following specific groundrules and guidelines were defined by the study plan:

1. Fuel Cells

   Primary Electrical Power 28 vdc ±5%; 1KW

O_2 & O_2 Tank

Cryo Storage for Fuel Cell O_2 99.99% IPPM CO & CO_2

H_2 & H_2 Tank

Cryo Storage for Fuel Cell H_2 99.98% IPPM CO & CO_2

2. The oxygen and hydrogen required by the fuel cell system will be stowed in common tankage with the APS system. This approach requires putting small scrubbers in the line to keep the CO and CO_2 content low for efficient fuel cell operation.

3. Tug avionics will be in an unpowered condition during Shuttle ascent and descent operations except for safety and thermal requirements.

4. Sustaining power to Tug during ascent and descent is Shuttle furnished.

General Groundrules and Guidelines

1. Advanced materials and concepts are to be used with a materials and concepts technology of 1976 and an IOC of the end of 1979.

2. The Tug will be designed as a fail-safe vehicle.

3. The Tug will be designed for a mission life of 20 missions. Refurbishment of subsystems after each mission is acceptable.

4. The Tug will be designed for an on-orbit stay time of 6 days unattached to the Shuttle.

Design Analysis Requirements

An electrical load analysis for the Tug vehicle was performed during the study. The analysis resulted in the following operational requirements:

1. Continuous Power (28V X 32A) 0.907 KW.

2. Peak Power (28V X 114A) 3.2 KW for 5 seconds.

3. Total Energy per Mission 151.5 KWH.

5.1.2 Subsystem Trades

Two types of fuel cells are currently being developed for space application. Pratt and Whitney Aircraft produces a fuel cell in which the reaction takes place in an alkaline (KOH) porous matrix. General Electric uses an acid solid polymer membrane. Two NASA Centers have a number of technology programs

underway to build prototype fuel cells applicable to the Shuttle and Space Station requirements. Table 5.1-1 summarizes the goals for these programs. Two technology levels are shown. The Pratt and Whitney Mark I and the General Electric EM-1 represent goals to be realized by 1972-73. The Pratt and Whitney Mark II performance shown represents 1976-78 fuel cell goals. The Pratt and Whitney development effort for the Mark I technology is being funded by NASA-MSC under Contract No. NAS9-11034. NASA Lewis Research Center is funding the advanced Shuttle fuel cell technology program under Contract No. NAS3-15339 to Pratt and Whitney Aircraft. The acid solid polymer fuel cell technology is being developed by General Electric for NASA-MSC under Contract No. NAS9-11033. The principle advances to be achieved by the NASA Lewis Advanced Technology Program is decreased fuel cell weight and increased lifetime.

Table 5.1-1 Fuel Cell Technology Goals

	PRATT & WHITNEY		
	MARK I	MARK II	GE EM-1
Responsible Agency	NASA MSC	NASA Lewis	NASA MSC
Technology Date	1972/3	1976/8	1972/3
Sustained Power (kw)	5	7	5
Voltage Level (Volts)	28	112	28
Voltage Regulation (0-5 kw)	±5%	±5%	±5% *
Weight (lb/kw)	40-60	20-30	40-60
Specific Reactant Consumption (lb/kw-hr)	0.7-0.8	0.7-0.8	0.7-0.8
Reactant Supply Press (Psia)	20-1000	35-1000	20-1000
Reactant Purity Grade	Propulsion	Propulsion	Propulsion
Heat Rejection Means	Coolant	Coolant & Other	Coolant
Start/Stop Cycles	500	Undefined	500
Operating Life (Hours)	2000-5000	10,000	2000-5000

NOTE: *Requires external voltage regulator

The industrial contractors for the above three development programs were requested to submit fuel cell weights and performance based on Tug requirements. The data received is documented by References 5.1-1 and 5.1-2. The solid polymer electrolyte fuel cell data supplied by General Electric (Reference 5.1-2) was based on their current Shuttle fuel cell program and represents the 1972-73 technology level. Data supplied by Pratt and Whitney (Reference 5.1-1) was based on the advanced Shuttle fuel cell program and represents a 1976-78 technology base.

The data received are summarized by Table 5.1-2. Fuel cell weight, volume, and performance based on the advanced Shuttle technology and the alkaline porous matrix is considerably better than values for the same parameters shown for the acid solid polymer based on 1972-73 Shuttle technology.

The acid polymer fuel cell requires a total parasitic power of 100 watts continuous regardless of power level. By incorporating a coolant pump run by the hydrogen gas flowing into the stack, the alkaline cell does not require parasitic power. Also heaters have not been included in the stack for fuel cell start up or operation. Accordingly, the selected fuel cell concept is based on the advanced Shuttle technology. Availability of this fuel cell depends on the continuation of NASA support of this technology.

Table 5.1-2. Tug Fuel Cell Comparison

Technology	Adv. Shuttle (1976-78)	Shuttle (1972-73)
Type Fuel Cell	Alkaline Porous Matrix	Acid Solid Polymer
Power Range, Watts	200-3000	200-1700 (1)
Voltage Regulation (Inherent	±5%	±5%
Stack Temperature, °F	180	180
Specific Reactant Consump. lbs/Kw hr	.824	.844 (2)
Weight, lbs	37	75 (1)
Volume, ft^3	0.68	2.35
Parasitic Power, watts	0	100

(1) Based upon 1 stack of 32 cells, 2 stacks of 32 cells each will allow operating over a range of 200-3000 watts within 28 ±5% V, the associated fuel cell weight is 125 lbs.

(2) Increases 5% to 8% if regulator required.

5.1.3 Subsystem Operation

A preliminary schematic of the advanced alkaline porous matrix fuel cell selected for this study is shown in Figure 5.1-1. The numbers following identification of components in this section are coded to Figure 5.1-1. The principal subsystems making up the power-plant are the fuel cell stack, the combined heat and water management subsystem, the reactant supply subsystem, and the instrumentation and control subsystem.

The fuel cell stack generates direct current electrical power on demand. The stack consists of fuel cell plaques and evaporative cooler assemblies bonded together between end plates. By-product water removal cavities adjacent to each cell assembly allow direct, passive removal of water vapor from the cells. Cell waste heat is removed by an evaporative cooler assembly located between sets of plaques. This assembly consists of two water cavitites on either side of a central steam cavity. Water separators provide barriers between each water cavity and the steam cavity. Each separator consists of a porous material which is wet proof and allows steam to pass through while retaining water in the water cavity for cell cooling.

The combined heat and water management subsystem provides thermal control, removes steam produced in cooling the cells, and removes byproduct water. A steam pressure regulator (13) maintains the steam cavity pressure at a constant value. This indirectly controls stack temperature by controlling the boiling temperature of the water in the cell coolers. This temperature is approximately 183°F resulting in a vapor pressure of approximately 8 psia.

The consumption of hydrogen gas at the cell anode and oxygen at the cathode results in the production of heat, electricity and water. The product water vapor from the stack vents directly into the low pressure vapor line running to the steam condenser. A low pressure is maintained in this line by the water ejector and condenser pressure regulator (16) located in a water bypass loop around the pump. This pressure, combined with the stack temperature maintained by the evaporative cooler assembly, controls cell electrolyte concentration.

Evaporative cooler steam joins the product water vapor stream, then enters the condenser where the latent heat of vaporization is transferred to the radiator loop. Steam enters the condenser at approximately 183°F, is condensed and then subcooled to 140°F at the condenser exit. The condenser operating pressure is approximately 3 psia. The condensing temperature is controlled by a condenser bypass valve (17) on the spacecraft coolant loop. This valve will be designed for 100 percent modulation. Water from the condenser is returned to the intercell evaporative coolers in the fuel cell stack by the hydrogen driven water pump. Coolant pressure downstream of the pumps are in the order of 15 - 17 psia. The water flows into the stack through a regulator (14) which maintains the pressure in the evaporative cooler water cavities higher than the pressure in the steam cavity. A gas separator at the pump exit removes and vents any noncondensables that may be accumulated in the loop. These are primarily unpurged nitrogen from storing the power plant, and

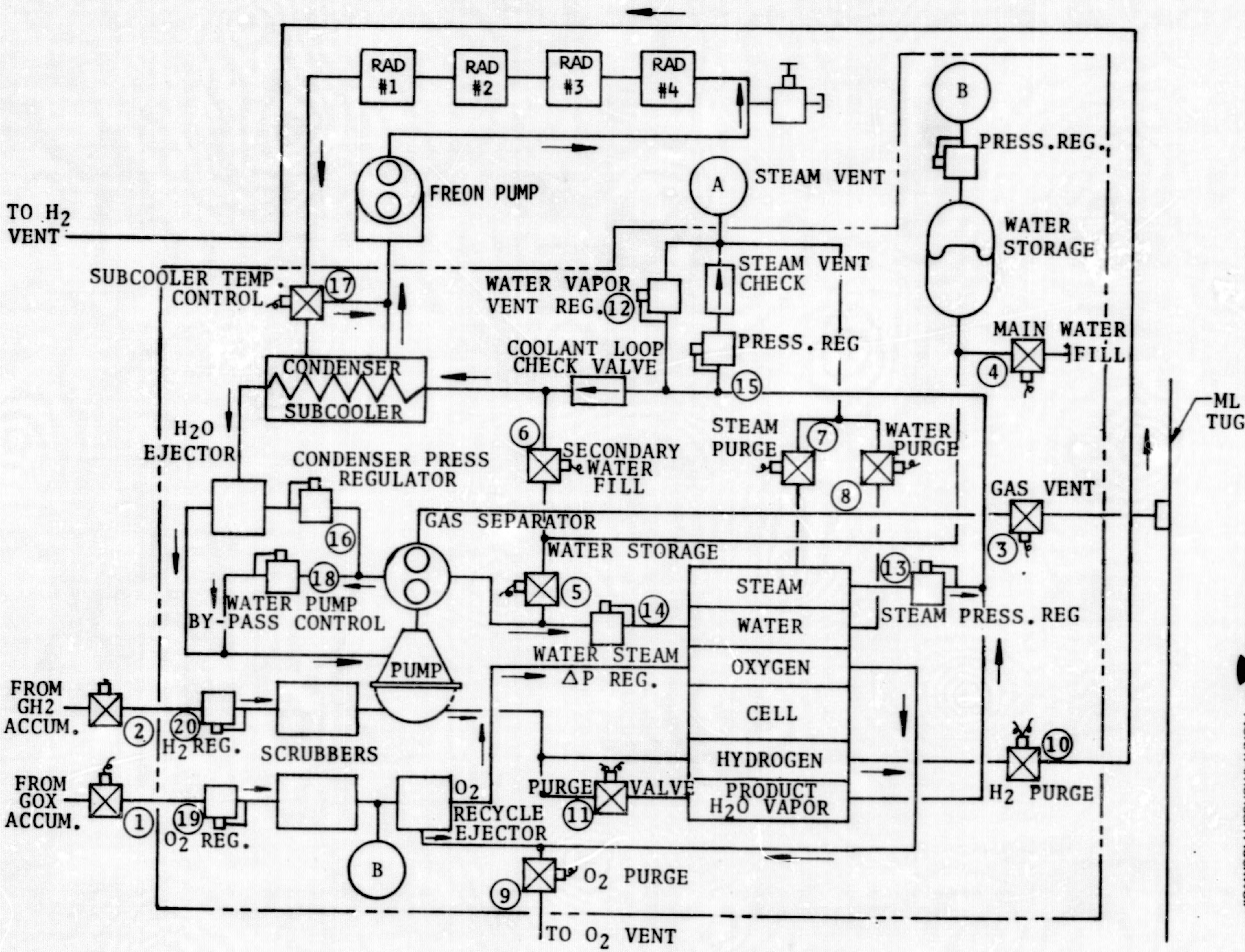

Figure 5.1-1 Fuel Cell System Schematic

air that may have gotten into the coolant loop. A water pump bypass control (18) recirculates water that is not required by either the fuel cell evaporative coolers or the water ejector.

While operating between the specified power range of 200 to 1100 watts, the power plant is cooled by a radiator. At power outputs of more than 1100 watts, radiator heat rejection is augmented by open cycle cooling. This technique is known as dual mode heat rejection. Open cycle cooling is accomplished by venting a portion of the steam from the evaporative coolers overboard together with a portion of the product water. Water supplied from the fuel cell water storage system makes up for the vented steam. Dual mode cooling will only be required during main engine burn. The longest burn is for 1300 seconds, requiring 1.75 pounds of water from the water accumulator. Total engine burn is 2520 seconds requiring a total of 3.4 pounds of water from the accumulator. The overboard vent regulator at the condenser inlet (15) provides automatic switch-over to open-cycle cooling and byproduct water removal whenever the spacecraft coolant loop cannot handle the heat rejection requirements of the powerplant. A check valve in the vapor line to the condenser prevents back flow of liquid downstream of the condenser. During open-cycle operation water for cooling is drawn from the spacecraft water storage system and flows directly in the stack.

During powerplant operation at less than 1100 watts, byproduct water is vented through the water vapor vent (12), and overboard through the steam vent system since there is no requirement for water by the Tug subsystems. Current power system design does not allow venting byproduct water vapor through the open cycle cooling mode steam vent. However, this byproduct water could be recovered at low power and used for dual mode cooling to reduce radiator area requirements. Radiator weight savings must be equated against the increase in weight required for storage of byproduct water. For the small amount of water required for augmented cooling (3.4 pounds) the water storage tank is assumed to be an integral part of the power plant assembly. Water will be displaced from the tank by a bladder pressurized by the gaseous oxygen line. Water anti-freeze capability is not included in the design described in this study. Such a capability must be included in any detailed design effort. It is anticipated that addition of anti-freeze capability will result in a minor increase in system weight. The tank will be filled on the ground, and it is assumed that only a static regulator will be required. This concept does not permit storing byproduct water during flight. The steam purge valve (7) and the water purge valve (8) shown by Figure 5.1-1 are required only during power plant start up and shutdown.

Reactants will be supplied in the gaseous state from the APS accumulators at 175 to 400 psia. Hydrogen is stored at -260°F and oxygen is stored at -60°F. Since the present regulator design specifies a minimum of 0°F operating temperatures, the reactants will need to be preheated. Neither method nor hardware definition for the reactant heating is presented in this study. However, a design solution for the problem should be achievable with only a minor system weight increase. Reactant isolation valves (1) and (2) are required for power plant startup and shutdown.

The reactants to the cell will be controlled by demand-type coupled regulators (19) and (20) to maintain 16 psia reactant pressure in the stack. Further advances in fuel cell technology beyond that projected for this application will permit increasing the pressure to about 60 psia, thereby permitting saving of about 5% in reactant consumption. Scrubbers downstream of each regulator remove any traces of carbon dioxide, a source of fuel cell performance loss, from the incoming gases. During normal operation hydrogen and oxygen flow into the stack and are completely consumed. Inert gases contained in the reactants accumulate in the cells and are periodically purged from the system during ground maintenance. Purge valves (9) and (10) are provided for the hydrogen and oxygen cavities in the cell stack. For propulsion grade reactants the total mount of purged gas will be about one percent of consumption.

Fuel cell characteristics are summarized by Table 5.1-3. The basic powerplant weight taken from Reference 5.1-1 is 33 pounds. An additional 4 pounds has been added to the weight to allow for a controller and the cooling loop water inventory. An additional weight of 6.0 pounds is estimated to be required for open cycle cooling accessories. This weight includes water, valves, lines, regulator and the water storage tank.

Table 5.1-3. Advanced Fuel Cell Characteristics

PERFORMANCE	
1. Maximum Power - Dual Mode Cooling	300 watts
Radiator Cooling	1100 watts
2. Minimum Power	200 watts
3. Minimum Voltage (at 3000 watts)	26.6 volts
4. Maximum Voltage (at 200 watts)	29.4 volts
5. Voltage vs. Power (Stack Temperature Constant at 180°F)	Figure 5.1-2
6. Heat Rejection at 1000 Watts	2237 Btu/hr
7. Heat Rejection vs. Power	Figure 5.1-3
8. Specific Reactant Consumption at 1000 Watts	0.824 lb/KWH
9. Reactant Consumption vs. Power Output	Figure 5.1-4
WEIGHT AND VOLUME	
1. Weight (WET)	Basic Unit 33 Controller 4 Open Cycle Cooling Accessories 6 Total 43 lbs.
2. Size	19.6" x 7.7" x 7.8"
3. Volume	0.68 Cubic Feet

Figure 5.1-2 shows the variation of fuel cell voltage with power output for a fixed stack temperature of 180°F. Fuel cell heat generated as a function of power output is given by Figure 5.1-3. Heat rejected by open cycle cooling is not included in Figure 5.1-3. Fuel cell reactant consumed as a function of power output in shown in Figure 5.1-4.

5.1.4 Fuel Cell Installation and Checkout

The powerplant will be stored in a container pressurized with dry nitrogen. The fluid connections at the interface panel will be capped and all voids within the fuel cell stack and liner will be filled with nitrogen at approximately atmospheric pressure.

Assuming that a bench test is not required, the fuel cell can be removed from the container and mounted in the tug. All connections to the power plant are made on a single interface panel. These connections are summarized by Table 5.1-4. Instrumentation and control signals between the power plant and the avionics are used to perform pre-start checkout, to start and stop the power plant, and to monitor operation while the power plant is on load.

After the fuel cell has been secured to the Tug mounting bracket, the fluid line connection at the interface panel will be uncapped and mated to the appropriate lines installed in the tug. Interface panel electrical connections to Tug electrical leads will then be made.

Startup and Checkout - Preliminary analysis indicates the following procedure for startup and checkout (refer to Figure 5.1-1 for component code numbers). Connect the ground supply water line to the water fill connection at the interface panel. Make gaseous hydrogen and oxygen available at the inlet to the reactant isolation valves. Fill the coolant loop with water between the check valve and the gas generator by opening the main water fill valve (4), the gas vent valve (3), and the secondary water fill valve (6). It is assumed that the gas separator will be designed not to vent water during this operation. The remainder of the coolant loop is filled by closing the secondary water fill valve (6), opening the water storage valve (5), opening the reactant isolation valves (1) and (2), opening the hydrogen (10) and oxygen (9) purge valves. After the oxygen chambers in the stack are purged of nitrogen (approximately 2 to 3 minutes), close the oxygen purge valve (9). When the condenser pressure has reached 3 psia, close the hydrogen purge valve (10). Apply a starting load (dummy load bank) and when stack temperature reaches 180°F, the fuel cell is ready for checkout. Checkout will consist of applying different level loads and monitoring current, voltage, stack temperature, condensing pressure, product water vapor pressure and comparing against specified values. Table 5.1-5 itemizes valve operation during startup of the cell.

Shutdown to Remove Power Plant - Pratt & Whitney Aircraft estimates the power plant can be left on open circuit for 30 days (charged with reactants). The following procedure is tentatively advanced to inert the fuel cell for storage or removal from the Tug. Remove load and connect gaseous nitrogen supply on tug side of reactant isolation valves (1) and (2) and to tug side

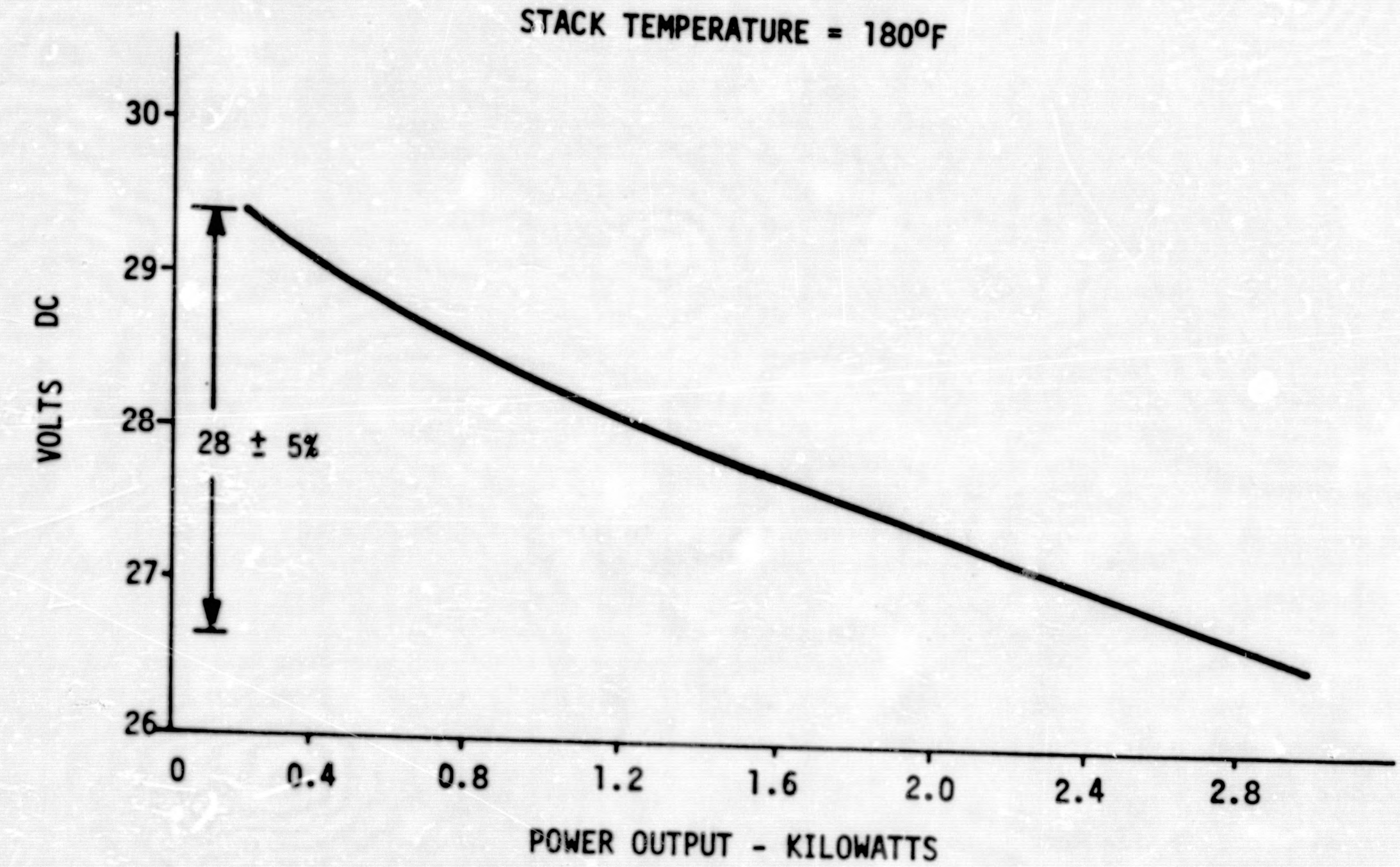

Figure 5.1-2 Fuel Cell Voltage

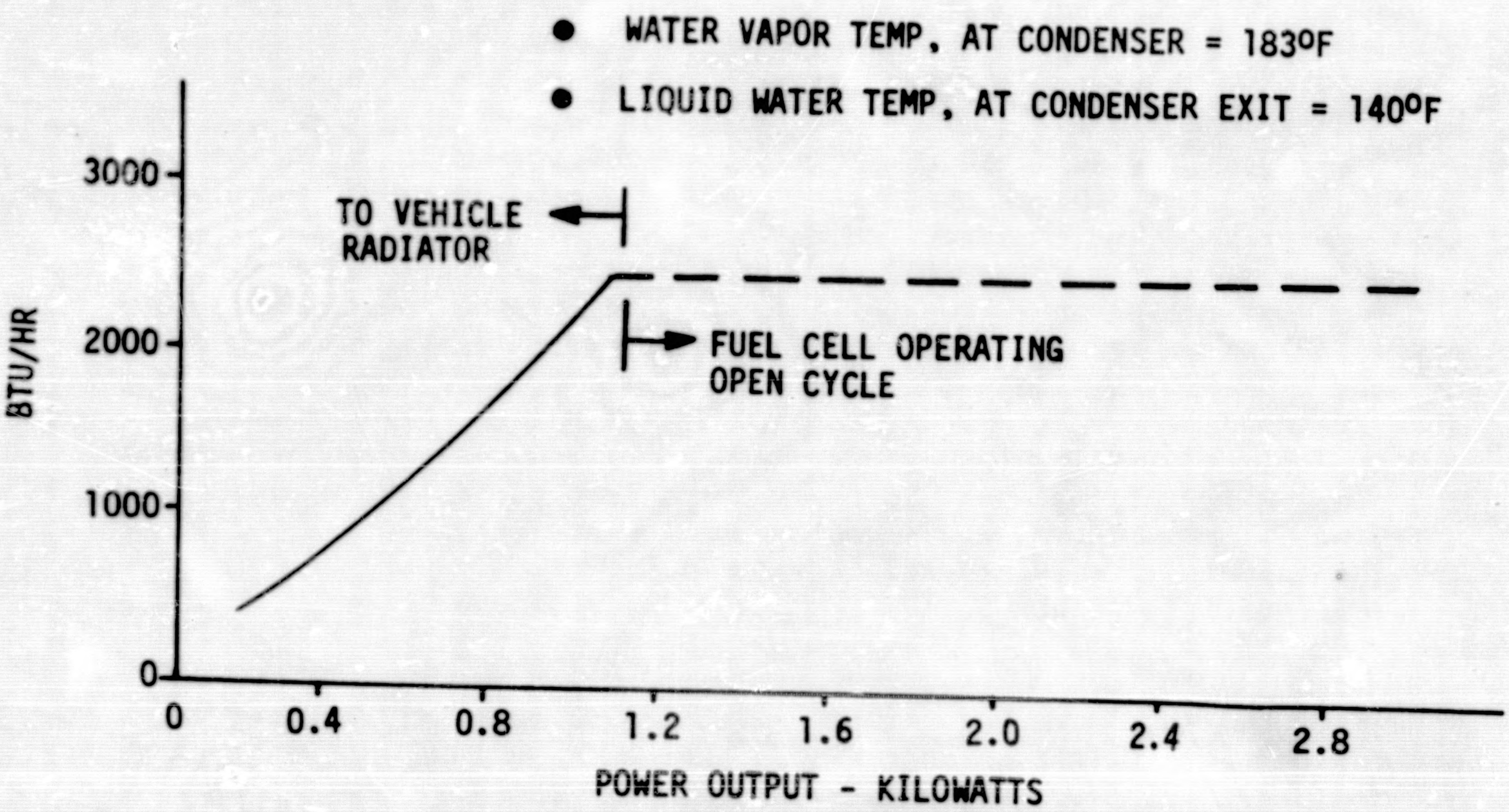

Figure 5.1-3 Fuel Cell Heat Rejection

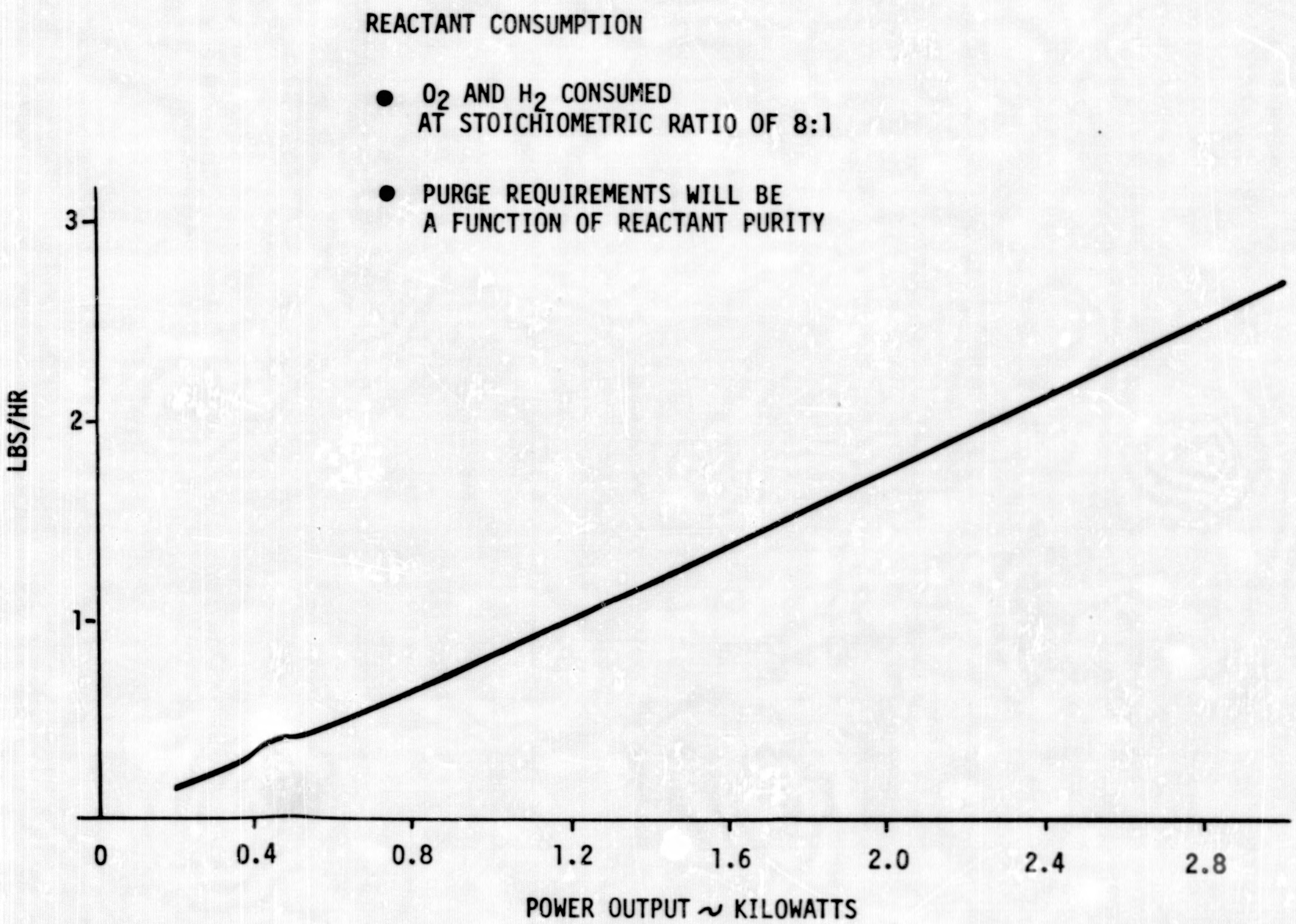

Figure 5.1-4 Fuel Cell Reactant Consumption Rates

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of main water fill valve. Open valves (1) and (2) and hydrogen and oxygen purge valves (10) and (9). Completely fill gas chambers with nitrogen and then close hydrogen and oxygen purge valves. Apply nitrogen pressure to water storage tank through valve (4), close gas vent valve (3), close water storage valve (5), open water purge valve (8), and open secondary water fill valve (6). When water quits flowing from water purge valve close secondary water fill valve (6). Open water storage valve (5) and close water purge valve when no more water comes out of the water chamber. Open steam purge valve (7) and product water vapor purge valve (11). When the steam and product water vapor chamber have been purged of water vapor close all valves and complete shutdown.

Table 5.1-4. Fuel Cell/Tug Interface

INTERFACE PANEL
1. Power Output 2. Instrumentation and Control Signals 3. Hydrogen Inlet and Outlet 4. Oxygen Inlet and Outlet 5. Liquid Water Fill 6. Steam Vent* 7. Coolant in and out 8. Water Pump Vent Gas *Steam purge, water purge, water vapor vent tied to steam vent inside fuel cell.
INSTRUMENTATION
1. Stack Current 2. Stack Voltage 3. Stack Temperature 4. Product Water Vapor Pressure 5. Condenser Pressure
ELECTRICAL POWER AND CONTROL
1. Electrical Pwr to Actuate Spacecraft Reactant Vlvs. at Startup 2. H_2 and O_2 Purge Valve Actuation Signal 3. Instrumentation Signals 4. Start/Stop Signals

5.2 ELECTRICAL POWER CONVERSION AND DISTRIBUTION

The Electrical Power Conversion and Distribution Subsystem provides the capability to convert fuel cell generated DC power to AC power and to distribute both DC and AC power to the Tug vehicle components requiring electrical power. The subsystem is composed of a 115 VAC, 3 Ø, 400 Hz, 1 KW static inverter, a 3-position power transfer switch, solid state power control

switches (PCS), resistor diode modules, and power bus modules. Although it is a power generation rather than conversion or distribution device, the backup system battery is included in this section of the report for the purpose of convenience.

As shown on the subsystem functional block diagram, Figure 5.2-1, the power control switch is used to allow fuel cell, EOS, or GSE generated power to be applied to the main power distributor (power bus module). Power to the individual subsystem distributors is supplied through power control switches operating under DMS control. Vehicle components are then powered off the appropriate subsystem distributor. Current limiting, isolation and arc suppression are provided by the use of the resistor diode modules (not shown in Figure 5.2-1). Power to the DMS distributor is taken directly off the main distributor to allow DMS operation with minimum external interfaces. Power to the backup system distributor is applied directly from the fuel cell in the normal mode. The battery is continually floated on the line to automatically supply power to the backup system in the event of a feul cell failure. The static inverter is powered off the TVC distributor to drive the TVC hydraulic pump motor and other selected loads.

5.2.1 Requirements

The design of the subsystem was developed following the specific and general groundrules and guidelines of the study plan and using NR design analysis established operational requirements.

Specific Ground Rules and Guidelines

The following specific groundrules and guidelines were identified by the study plan:

1. Battery, 28 vdc, size for 30 minutes operation to provide emergency power attitude and etc.

2. Avionics unpowered during Shuttle ascent and descent operations except for safety and thermal control.

3. Sustaining power to tug during ascent and descent provided by Shuttle (300 W Avg., 500 W Peak).

General Groundrules and Guidelines

General groundrules and guidelines set forth in the study plan are as follows:

1. Fail-safe design with no unsafe condition for Shuttle or its crew and no payload destruction.

2. 1976 materials and concepts technologyy.

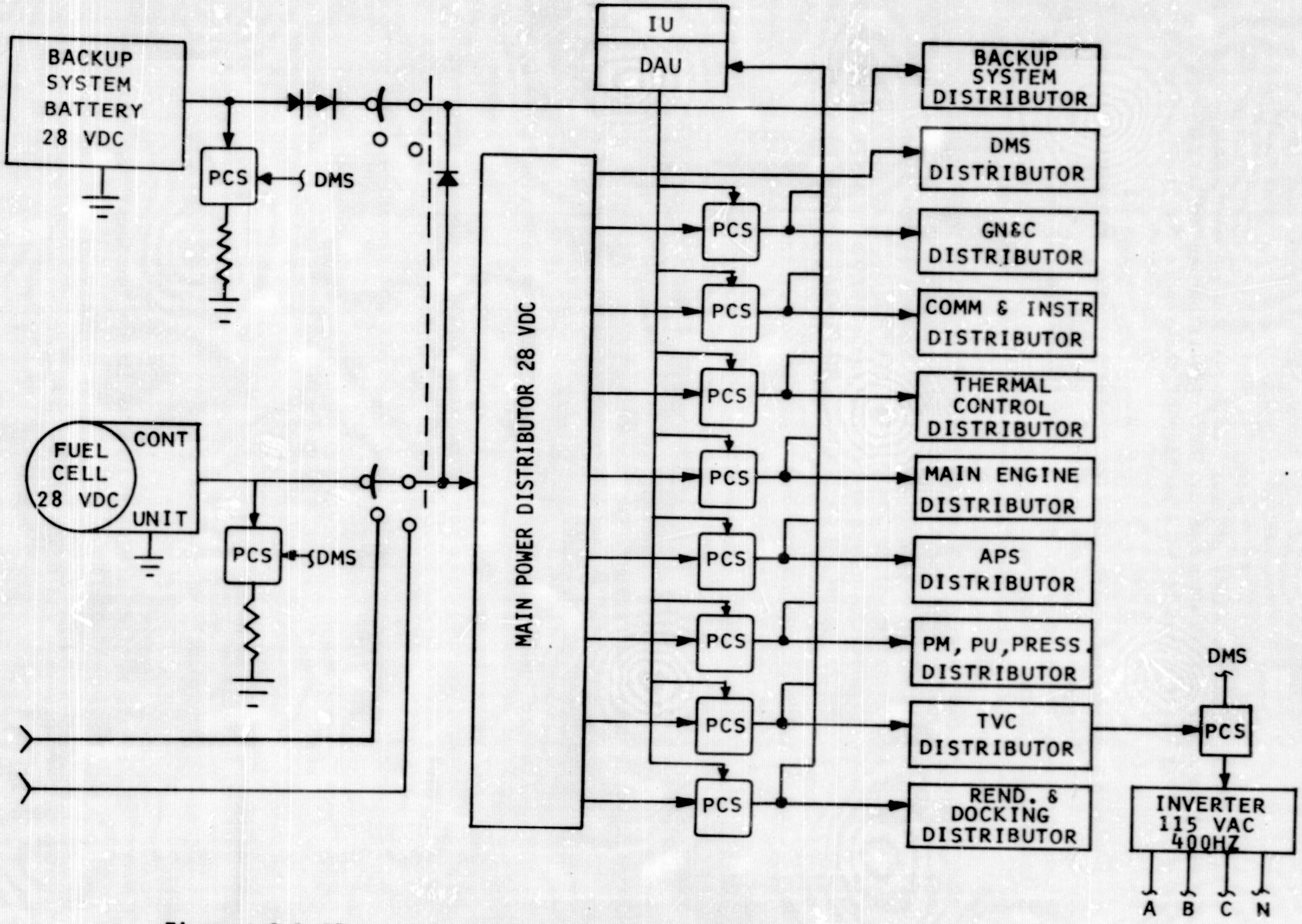

Figure 5.2-1 Electrical Power Conversion and Distribution Subsystem Functional Block Diagram

3. 6 day mission capability.

4. 20 mission life capability

5. Ground refurbishment, maintenance and repair after each mission

Design Analysis Requirements

The study design analysis resulted in the establishment of the following requirements.

1. Provide for 1 kw of power conversion from 28 vdc primary power to 115 vac, 3 ϕ, 400 Hz power.

2. Provide approximately 186 PCS's for power distribution. The quantities and ratings required for the particular vehicles are itemized in Table 5.2-1. A detailed listing may be found in Table 3.1-3.

Table 5.2-1 Tug PCS Requirements

SUBSYSTEM	PCS RATING CAmps									TOTAL
	1	2	3	5	7.5	10	15	25	35	
Propellant Feed, Fill & Drain	-	-	8	12	-	-	-	-	-	12
Pressurization	-	-	-	8	-	-	-	-	-	8
Propellant Orientation	8	-	-	-	-	-	-	-	-	8
Safing & Venting	-	-	18	-	12	-	-	-	-	30
Propellant Management	-	-	4	4	-	-	-	-	-	8
Thrust Vector Control	-	-	-	-	1	-	-	-	2	3
Auxiliary Propulsion	11	2	32	-	-	-	-	-	-	45
Data Management	2	-	-	-	-	-	-	-	-	2
Guidance, Navigation & Control	-	-	-	2	-	-	-	-	-	2
Rendezvous & Docking	2	-	-	-	-	-	-	-	-	2
Communications	2	-	4	-	-	2	-	-	-	8
Electrical Power Dist.	2	10	1	-	3	1	1	2	5	25
Fuel Cell	-	2	11	-	-	-	-	-	-	13
Purge Bag	2	-	6	-	-	-	-	-	-	8
Tug/Payload Docking	-	-	-	-	2	-	-	-	-	-
Totals	29	14	84	26	18	3	1	4	7	186

3. Sustaining power during ascent and descent is 207W for average usage and 570W for peak demand. The latter wattage is 70W over the guideline limit of 500W. Should this limit be inviolate, EOS crew control of the TVC heaters can be included in the design to allow turning the heaters off during application of power to the instrumentation for status checks.

5.2.2 Load Analysis

Three top level load analyses were performed to verify subsystem compatibility with study plan requirements and to develop the baseline subsystem design. The three analyses involve EOS provided power usage, AC power usage, and total mission power usage.

EOS Provided Power Usage

While the tug is in the EOS cargo bay during the low earth orbit ascent and descent phases of the mission, the fuel cell will be inactive and the tug will obtain power from the EOS. In order to stay within the study plan groundrule limitation of 300W average and 500W peak power usage, all tug electrical components will be powered down with the exception of those needed to maintain safety status and thermal control of avionics components. The equipment requiring power and the amount of power required is identified in Table 5.2-2.

Table 5.2-2 EOS Sustaining Power Usage

EQUIPMENT	AVG. PWR. (W)	PEAK PWR. (W)
DMS Computer	60	60
*DMS DAU	-	30
DMS Status & Control Panel	40	80
Avionics Equipment Heaters	107	107
TVC Heaters	-	200
Instrumentation	-	93
Totals	207	570

* This equipment will be turned on for periodic system status checks. No more than one DAU will be powered up at any given time.

AC Power Usage

Vehicle components requiring AC power were identified and an analysis made to determine the necessary size of the static inverter. The TVC hydraulic pump motor will require approximately 1.4 KW peak during start for 1 second, 0.9 KW maximum intermittently and 0.54 KW average during pump run condition. The fuel cell freon pump motor will require 25 watts of AC power continuously. The hydraulic pump will be used continuously

during main engine burn for engine gimballing and intermittently to circulate hydraulic fluid during coast periods. Intermittent operation of AC motor operated valves will require inverter power. The total AC loads were estimated as follows:

	AVERAGE	MAX.	CONTINUOUS
TVC Hydraulic Pump Motor	540 W	900 W	N/A
Fuel Cell Freon Pump Motor	25 W	25 W	25 W
Motor Operated Valve	75 W	75 W	N/A
	640 W	1000 W	25 W

Based on the above estimates, the inverter was sized as follows:

Assume: 75% efficiency

0.9 Power Factor

1 KW Load 0.9 KVA

Input Voltage 28 VDC

Output Voltage 115 VAC, 400 Hz, 3 ϕ, Wye

$$P_{in} = \frac{P_{out}}{Eff.} = \frac{1\ KW}{0.75} = 1.3\ KW$$

0.9 KVA = 0.9 x 1 KW

Total Mission Power Usage

Figure 5.2-2 is an estimate of the internal electrical power profile for a 6 day mission while separated from the orbiter vehicle. The first main engine burn is described by Figure 5.2-2 indicating a peak starting power of 3.2 KW with a 2.2 KW power run condition. Other systems required during main engine operation are defined by Figure 5.2-2. The total energy estimated for a 6 day mission is 151.5 KWH. Approximately 125 pounds of reactants will be required for the fuel cell (0.824 pounds of reactants for each kilowatt hour). The average continuous power of 0.907 KW is utilized by the Tug subsystems as follows:

28 VDC PRIMARY POWER FUEL CELL
TOTAL ENERGY PER MISSION 151.5 KWH
AVERAGE POWER (28V X 32A) 0.907 KW
PEAK POWER (28V X 114A) 3.2 KW

1. TUG POWER ON 0.907W CONTINUOUS.
2. LH_2 & LOX TANK HIGH RELIEF MODE ON (340W FOR 1 SEC)
3. PROP SETTLING THRUSTERS ON 38 SEC PRIOR TO ENG START
4. STAGE HYDRAULIC PUMP 1.4 KW FOR 1 SEC (540W RUN)
5. SETTLING THRUSTERS OFF
6. LH_2 & LOX PREVALVES OPEN (200W EACH FOR 1 SEC)
7. MAIN ENGINE START 1601W FOR 5 SEC, INSTR PWR 124W
8. MAIN ENGINE IDLE MODE 591W FOR 119 SEC
9. PREPRESS ON 336W FOR 115 SEC (ENG START + 7 SEC)
10. PREPRESS TANK SYSTEM OFF - MAIN ENG BURN ON 1176 SEC)
11. MAIN ENG BURN OFF - PREVALVES CLOSED - 5 SEC PURGE ON 300W
12. 5 SEC PURGE OFF - ENG INSTR PWR OFF
13. LH2 & LOX TANK HIGH RELIEF MODE OFF (340W FOR 1 SEC)
14. TUG POWER 0.907W CONTINUOUS

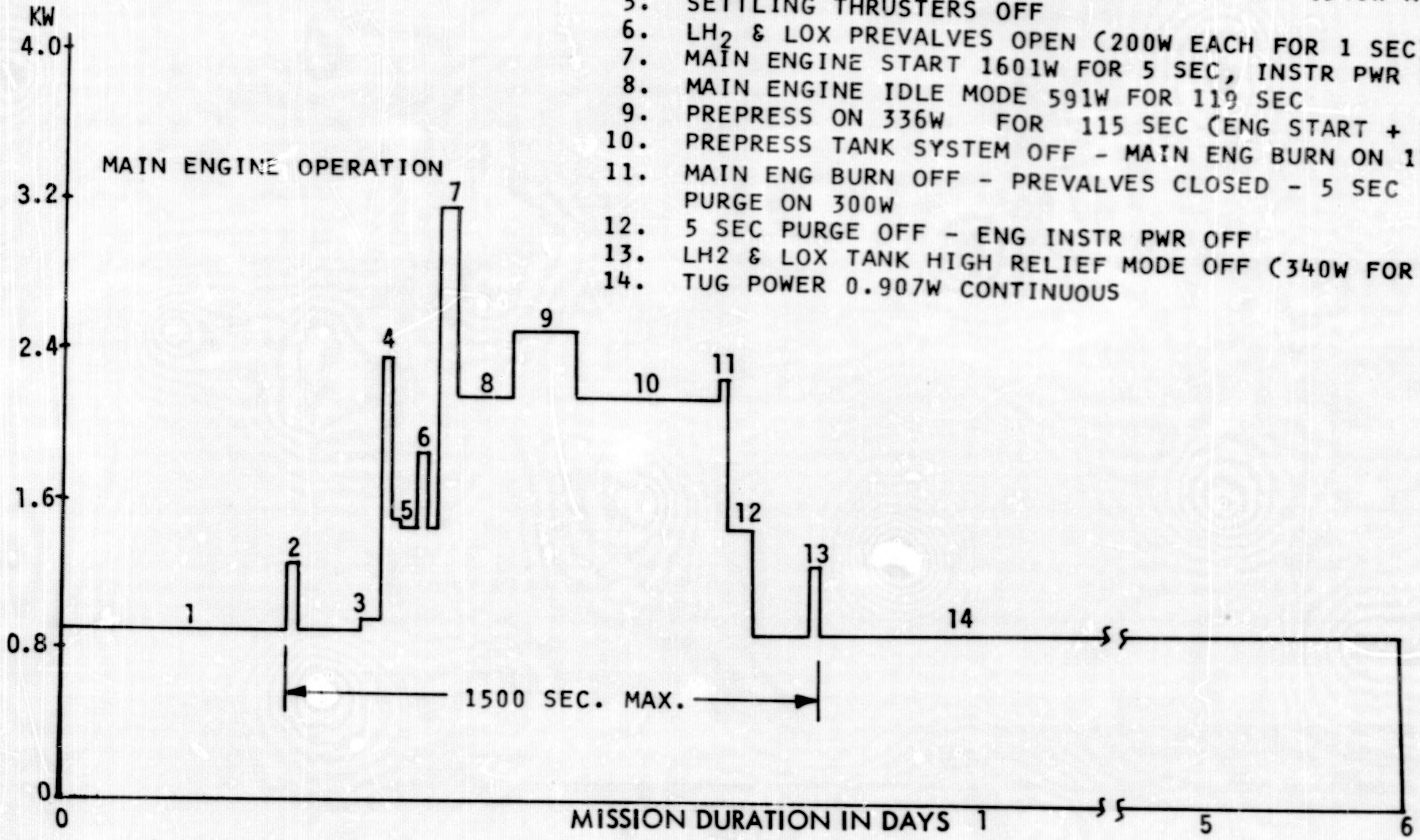

Figure 5.2-2 Electrical Power Profile

SUBSYSTEM	AVG. CONT. PWR. (W)
Communications	59.0
Instrumentation	95.3
Avionics Thermal Control	151.0
Data Management	200.0
GN&C	136.5
Main Propulsion	10.0
Thrust Vector Control	79.0
Fuel Cell Freon Pump Motor	25.0
Elec. Pwr. Gen. & Distr.	131.0
Total	906.8 W

The remaining systems consume electrical power intermittently during the mission. The total 151.5 KWH energy level for the 6 day mission was determined by estimating the quantity of power required for individual systems during intermittent operations plus the 0.907 KW continuous power for the above systems.

5.2.3 Subsystem Operation

The subsystem design has the capability to operate using GSE power connected to the main power distributor through the power transfer switch. It is anticipated that this would be the pre-launch operational mode. Shortly before liftoff of the EOS, the power transfer switch would be driven to the EOS position by either a GSE command or an EOS command initiated at the DMS status and control panel located in the EOS. The Tug would receive sustaining power from the EOS until deployment is initiated.

After the Tug has been deployed from the EOS cargo bay, but prior to Tug separation from the docking adapter, the fuel cell will be activated and checked out. Upon verification via the status and control that the fuel cell is operating properly, the EOS crew will command the power transfer switch to the fuel cell, placing the Tug loads on the fuel cell output. At that time, the DMS will commence a Tug systems power up and checkout routine at the direction of the EOS crew. Once proper status is indicated by the status and control panel, the EOS crew will transfer control to the DMS and release the Tug from the docking adapter to allow performance of the mission.

Typical on board control of components by the use of DMS activated PCS's is shown in Figure 5.2-3. The subsystem illus-traded is the LH_2 tank vent subsystem. At the initiation of a valve open or close command by the DMS, an output from the IU/DAU turns on the appropriate PCS. This results in the application of 28 VDC to the valve solenoid, thereby causing valve actuation. The output of the PCS's is monitored by the DMS computer

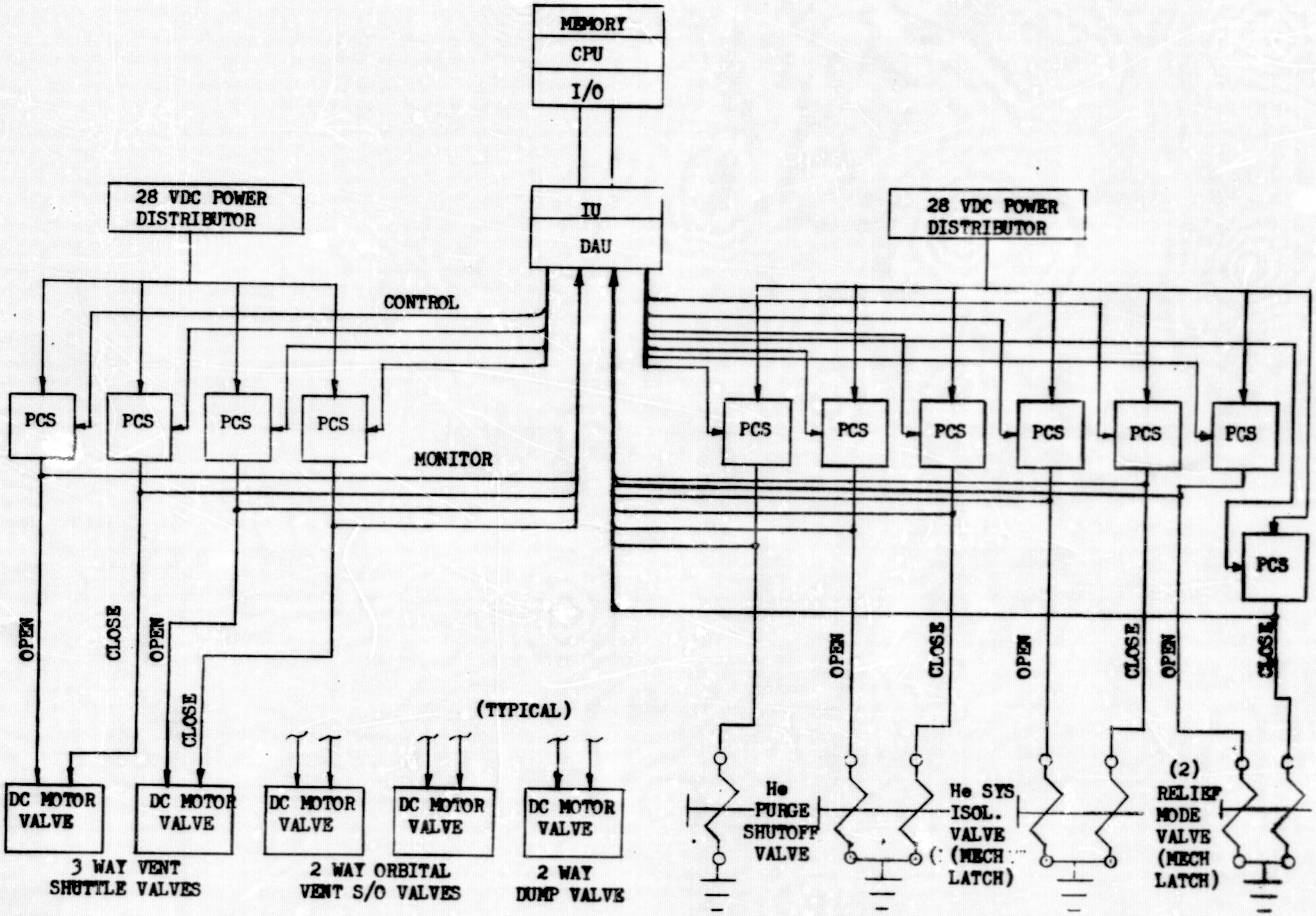

Figure 5.2-3 Tug Power Conversion and Distribution Subsystem (LH$_2$ Tank Vent System)

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to verify proper system operation. In the event of an overload on the PCS, the output will return to 0 VDC. Removal and re-application of the PCS control voltage by the DMS will cause the output to return to its normal voltage unless the overload is still present. The PCS requires the presence of an input control voltage to maintain the output voltage.

All distribution of power to valves, sensors, distributors, avionics components, heaters, etc., is accomplished in a manner similar to that described above.

5.2.4 Subsystem Checkout

Checkout of the subsystem will be performed to a detailed level prior to installation of the Tug in the EOS cargo bay. Only limited tests will be conducted on the subsystem once the Tug has been mated to the EOS.

The fuel cell will be activated for detailed checkout. After checkout is completed, and prior to installation of the Tug in the EOS, the fuel cell will be deactivated.

A battery substitute will be installed to allow checkout of the battery associated associated subsystem circuits. Battery installation and activation should not take place until immediately prior to applying GSE power to the Tug for the launch countdown. Should this prove to be operationally difficult or impossible, the subsystem must be redesigned to prevent battery loading when no fuel cell or external power is applied to the Tug. Such a redesign should impose no significant weight increase nor should it result in an unacceptable failure mode.

The detailed checkout activities will be as follows:

1. Measure the battery simulator output voltage downstream of bias diodes.
2. Measure the fuel cell output voltage.
3. Measure the main distributor voltage and current.
4. Measure the voltage and current at each subsystem distributor.
5. Verify the operation of each PCS through the use of the DMS checkout programs. Verify the capability of each PCS to drive the applicable load.
6. Verify the operation of the power transfer switch to be within rated voltage and current limits. Verify the transfer time and make-before-break capability of the switch.

The limited status checks of the subsystem after installation in the EOS will consist of measuring battery temperature, voltage and current and main power distributor voltage and current. Once the fuel cell has been activated, fuel cell stack voltage and current will be measured.

5.2.5 Component Characteristics

Backup Battery (1 Required)

The battery supplying the backup system safing loads is a nominal 28 VDC, primary silver oxide zinc battery. An integral heating element and solid state controller maintains the battery temperature within operating limits. A temperature probe is incorporated to monitor heater performance. Table 5.2-3 summarizes the physical and performance characteristics of the unit.

Power Transfer Switch (1 Required)

The motorized power transfer switch is a 2 pole triple throw device having make-before-break action. Auxiliary contacts are used to remove current from the motor windings and to indicate switch position. Table 5.2-4 summarizes the physical and performance characteristics of the unit.

Static Inverter (1 Required)

The static inverter is rated at 1 KW output to supply 115 VAC, 400 Hz loads via a 4 wire WYE system (ϕ A, ϕ B, ϕ C and neutral). Table 5.2-5 summarizes the physical and performance characteristics of the unit.

Power Control Switch (186 Required)

The PCS meets the requirements of MIL-P-81653. It is designed to include a circuit breaker type fail-safe current circuit. Removal and re-application of the input control voltage will reset the PCS. Tables 5.2-6, 5.2-7 and 5.2-8 summarize the physical and performance characteristics of the unit.

Resistor Diode Module (5 Required)

The resistor diode module provides encapsulated components for use in current limiting, isolation, and arc suppression. Table 5.2-9 summarizes the physical and performance characteristics of the unit.

Power Bus Module (2, 40 A Units ϕ 10, 20 A Units Required)

The power bus module is an encapsulated distributor for the Tug subsystem loads. Table 5.2-10 summarizes the physical and performance characteristics of the unit.

Table 5.2-3 Backup Battery Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
Weight	12 Lb.
Size	4 x 3 x 6 In.
Type	Silver Oxide Zinc
Operating Temperature Range	-65F to +125F
Construction	Sealed Case With Pressure Relief Valve
Number of Cells	20

PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE
Capacity	18 Ampere - Hour
Voltage	$28 \begin{smallmatrix} + & 4 \\ - & 2 \end{smallmatrix}$ VDC

Table 5.2-4 Power Transfer Switch Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
Weight	2 Lb.
Size	4 x 5 x 5.5 In.
Operating Temperature Range	-65F to +125F
Construction	Sealed

PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE
Motor Voltage	28 $^{+2}_{-4}$ VDC
Auxiliary Contacts	5 Amp Resistive
Switch Time	200 Millisec.
Inrush Current	15 Amp Max.
Duty Cycle	1 Cycle/Min.
Power Contacts	200 Amp Resistive
Contact Voltage Drop	0.15 VDC Under Load

Table 5.2-5 Static Inverter Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
Weight	15 Lb.
Size	3.25 x 9 x 12 In.
Operating Temperature Range	-65F to +150F
Reliability MTBF	1000 Hrs.

PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE
Input Voltage	28 ± 2 VDC
Output Voltage	115 VAC ± 10%, 3 ∅ wye
Output Frequency	400 ± 4 Hz
Output Power	1 KW Continuous 1.4 KW Peak
Efficiency	$\geq$ 75%

Table 5.2-6 Power Control Switch Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
Weight	Ref. Table 5.2-7
Size	Ref. Table 5.2-7
Operating Temperature Range	-65F to +248F

PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE
Operating Voltage	+29 VDC
Control Voltage	+5 VDC
Current Rating	Ref. Table 5.2-8
Turn-on Voltage	+3.5 VDC Min.
Turn-off Voltage	+2.5 VDC Max.
Turn-on Time	1 Millisec Max.
Turn-off Time	1 Millisec Max.
Output Rise Time	0.1 Millisec Min. 0.5 Millisec Max.
Output Fall Time	0.1 Millisec Min. 0.5 Milisec Max.
Input Resistance	500 Ohms $\pm$ 10%
Voltage Drop	0.5 VDC Max
Power Dissipation	Ref. Table 5.2-8
Fail-Safe Current	Ref. Table 5.2-8
Life Operating Cycles	1,000,000 Min.

Table 5.2-7 Power Control Switch Physical Characteristics

CURRENT RATING (AMPS)	WEIGHT (OUNCES)	SIZE
1	1.8	1" x 0.75" x 1.5"
2	2.0	1" x 1" x 1.5"
3	2.0	1" x 1" x 1.5"
5	2.0	1" x 1" x 1.5"
7.5	2.0	1" x 1" x 1.5"
10	3.0	1" x 1.6" x 2.0"
15	5.0	1.3" x 1.5" x 2
25	5.0	1.3" x 1.5" x 2.5"
35	5.0	1.3" x 1.5" x 2.5"

Table 5.2-8 Power Control Switch Performance Characteristics

CURRENT RATING (AMPS)	POWER DISSIPATION (WATTS)		LEAKAGE CURRENT (MICROAMPERES)	FAIL-SAFE CURRENT (AMPS)
	ON	OFF		
1	1.5	0.15	100	5
2	2.0	0.156	500	15
3	2.5	0.158	500	15
5	4.5	0.164	500	15
7.5	6.5	0.174	1000	30
10	8.5	0.178	1000	30
15	25	0.311	1000	45
25	40	0.311	1000	75
35	55	0.311	1000	105

Table 5.2-9 Resistor Diode Module Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
WEIGHT	0.7 Lb.
Size	2.63 In. Dia. x 3.5 In Ht.
Operating Temperature Range	-65F to +150F
Construction	Encapsulated

PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE
Resistor Rating	1/2 W
Diode Rating	1 Amp Continuous
Diode Reverse Voltage (V_R)	600 VDC
Diode Forward Voltage	1 VDC Max.
Diode Power Dissipation	0.75 W @ 77F
Diode Reverse Current	5 Microamp @ V_R

Table 5.2-10 Power Bus Module Characteristics Summary

PHYSICAL CHARACTERISTICS

ITEM	CHARACTERISTICS
Weight	3 Lb - 40 Amp Unit 0.7 Lb - 20 Amp Unit
Size	4 x 4 x 6 In - 40 Amp Unit 2.63 In. Dia. x 3 In Ht - 20 Amp Unit
Operating Tempeaatue Range	-65F to +150F

PERFORMANCE CHARACTERISTICS

FARAMETER	PERFORMANCE
Current Rating - Continuous	40 Amp & 20 Amp
Current Rating Peak	120 Amp & 60 Amp

REFERENCES

5.1-1 "Fuel Cell for Advanced Spacecraft," Letter from John W. Schmitt, Pratt & Whitney Aircraft to T. M. Littman, North American Rockwell, Space Division, dated December 3, 1971.

5.1-2 "North American Rockwell Corp., Seal Beach, California, Fuel Cell for Advanced Vehicles," Letter from L. E. Chapman, General Electric, Direct Energy Conversion Programs, to R. J. Casler, General Electric Defense Program Division, dated December 3, 1971.

VOLUME III

PART 1

APPENDIX A
GUIDANCE AND PERFORMANCE ACCURACY

APPENDIX A

GUIDANCE AND NAVIGATION PERFORMANCE ACCURACY

A1.0 INTRODUCTION

The guidance and navigation subsystem is required to autonomously transfer the Tug from one circular orbit to another with a prescribed terminal state accuracy. Since the Tug is a point design as far as G&N system parameter values are concerned, establishing Tug performance capability is resolved by answering the following two questions:

1. Using specified and state-of-the-art values for G&N system parameters, can the Tug meet specified injection accuracy requirements?

2. Is an autocollimator (or equivalent alignment) required as part of the baseline G&N system in order to meet specified injection accuracy requirements?

An error analysis investigation indicates that the answer to both questions is a "Qualified Yes". (For the present, an autocollimator is included to provide 15 $\widehat{\text{sec}}$ alignment accuracy. Either mounting both observational sensors on a common navigation base or conclusive evidence that optical misalignments induced during flight remain small would eliminate the need for an autocollimator. Furthermore, later, more refined study data may reveal that in-flight determination of the misalignment bias may be obtained from the Kulman estimation data.)

Error analysis data was generated by simulating major portions of the Tug mission timeline; for example, the sequence of orbital phasing, transfer injection burn, midcourse coast, orbit insertion into geosynchronous orbit. Position and velocity errors at the initiation of a mission timeline segment are propagated through successive coasting/navigation and thrusting/guidance segments in order to determine the resultant injection error.

A1.1 ERROR SOURCES

Due to the short duration of the study, the list of error sources was kept to a minimum, consisting of specified point design values, state-of-the-art values for non-specified parameters, and estimates of sensor mounting bias uncertainties. The error sources are grouped into two categories, observational sensing and inertial measuring. This grouping is consonant with the two mission operations of coasting/navigation and thrusting/guidance. Values for the major error sources are listed in Table A-1.

Table A-1. Error Sources

Error Designation	3σ Value	Origin
Initial State Errors		
3 Components of Position	2KM per axis	Shuttle Handoff
3 Components of velocity	2M/S per axis	Shuttle Handoff
Initial Attitude Misalignment		
3 Components of Attitude	.1 deg per axis	Mission Spec
Gyro Errors		
3 Components of random drift	.1 deg/hr	Mission Spec
Mass unbalance drift	.3 deg/hr/g	Mission Spec
Torquer scale factor uncert.	.01%	State-of-art
Accelerometer Errors		
3 Components of bias error	$.3 \times 10^{-4}$G	State-of-art
Scale factor uncertainty	.01%	State-of-art
Nonlinearity	$.1 \times 10^{-4}$G/G^2	State-of-art
Horizon Tracker Errors		
Total random errors	6 min	State-of-art
Star Tracker Errors		
Total random errors	30 sec	State-of-art
Optical Alignment Uncertainty		
Without autocollimator	30 min	Estimated
With autocollimator	15 sec	State-of-art

A1.1.1 Navigation Sensing

The optical navigation equipment consists of a double gimbal star tracker and a four head horizon edge tracker. This equipment is used for attitude reference as well as to make navigation measurements. The horizon sensor provides a crude measure of spacecraft altitude and the direction of the earth horizon. The star tracker measures azimuth and elevation angles with respect to the sensed horizon reference. These measurements are used to estimate the

spacecraft position and velocity vector. Errors in the state estimation are evidenced due to inherent errors in the measurement sensor elements and electronics, and due to measurement dynamics. The error sources, itemized in Table A-1, are grouped into two categories, random noise and bias uncertainty. Random noise represent current or specified instrument errors. Systematic bias errors can be determined and can be compensated in the G&N computations and preflight calibration. Uncertainties remain however due to calibration error and changes due to flight dynamics and environmental effects.

The major source of bias uncertainty is occasioned by the geometric factors associated with mounting the navigation sensors. The star tracker and horizon tracker are not located on a common navigation base due to different pointing and field-of-view requirements for viewing the celestial objects being sensed. Since the vehicle structure is somewhat flexible and since the horizon scanner must be deployed, the relative orientation between the navigation devices is not a predictable quantity. If this uncertainty is large, 30 min, it is difficult if not impossible to determine spacecraft position and velocity to within required accuracy. Inflight calibration, using an optical link between the navigation sensors, will reduce the bias uncertainty to acceptable levels. Available autocollimators can reduce bias uncertainty to approximately 15 sec.

A1.1.2 Inertial Measuring

During thrusting periods, navigation is performed by inertial instruments. The inertial measurement unit type analyzed is a triad configuration strapdown inertial measurement unit. Major error sources considered include initial state, initial misalignment, non "g" sensitive gyro drift, mass unbalance drift, accelerometer bias, acceleromatic scale factor, and gyro torques scale factor.

A1.2 PERFORMANCE EVALUATION

The error analyses data presented herein were generated by digital simulation techniques described below:

1. Quantitative performance data was produced in terms of position and velocity errors at points along a reference trajectory. The reference trajectory data simulates orbital coast, transfer coast, and injection burn segments of the Tug mission timeline.

2. During powered flight phases the trajectory is stepwise integrated in accordance with a nominal thrust and attitude time history, and with the perturbing influence of individual error sources. The transition matrix is computed from initiation to end of burn and is used to propagate the initial covariance matrix of state through the burn. The covariance matrix at the end of burn represents the statistical combination of the inertial measuring error effects.

3. During coast phases the orbit determination accuracy is computed as a function of the on-board navigational parameters (random noise, bias uncertainty, measurement schedule, and initial state vector uncertainty). Simulation of the orbit estimation procedure is based upon a Kalman filter for a discrete dynamical system with measurements

which are unbiased and sequentially uncorrelated. The state estimation error covariance matrix is computed as each observational measurement is made. This determines state estimation uncertainty due to the random errors in the observables. The measurement bias acts to increase the mean-squared error of the state estimate. The state error covariance is propagated between measurements using conic formulations for the state transition matrix.

A1.2.1 Autocollimator Tradeoff Data

Navigation performance data for a representative Tug mission segment are presented in Table A-2. Tug position and velocity errors at Shuttle handoff are propagated through successive mission operations of:

- 5 hours of orbital navigation in a 100 n mi phasing orbit; navigation sightings taken 2-1/2 minutes apart
- 1307 second injection burn into a 100 x 19323 n mi transfer orbit using inertial navigation
- 5 hours of orbital navigation in the 100 x 19323 n mi transfer orbit; navigation sightings taken 1 minute apart
- 564 second insertion burn into a 19323 n mi circular orbit.

The data indicates that navigational accuracy specifications at synchronous altitude of 50 KM and 5M/sec can be satisfied using an autocollimator to reduce navigation alignment uncertainty.

A1.2.2 Sighting Schedule/Measurement Error Tradeoff Data

The time required to determine the spacecraft state to an achievable accuracy is dependent upon the frequency of navigation sightings as well as the sighting accuracy. Tables A-3 and A-4 show to some extent how navigation accuracy is related to measurement frequency, mission duration, sensor random noise, and alignment bias uncertainty. These tables also serve as a key for time histories of position and velocity components for each case listed in the tables. In low altitude orbits, a measurement frequency of every 2-1/2 minutes for a duration of approximately 3 hours appears sufficient to determine spacecraft state with adequate accuracy. At the high altitude orbit, it requires at least 5 hours of tracking to attain adequate state determination. A high sighting frequency is of no advantage due to the 24 hour period; i.e., the angle of arc changes slowly. In fact, spacing navigation sightings closely together is disadvantageous due to sequential correlation between navigation data.

The time histories of component errors show the in-track error to dominate position error while altitude rate error dominates the velocity components. The schedule of star horizon measurements should be programmed to emphasize the reduction of in-orbit position and velocity errors.

Table A-2. Autocollimator Tradeoff Data for Representative Tug Mission

Coasting Navigation Parameters:

Measurement Frequency - 1 and 2-1/2 minutes
Star Tracker Random Noise - 30 sec
Horizon Tracker Random Noise - 6 min
Optical Alignment Uncertainty - 30 min
With Autocollimator - 15 sec

Thrusting Navigation Errors:
See Table A-1

Mission Timeline		With Autocollimator				Without Autocollimator			
Event	Cumulative Time Hrs	Position Error KM	Position Error FT	Velocity Error M/SEC	Velocity Error FPS	Position Error KM	Position Error FT	Velocity Error M/S	Velocity Error FPS
Orbit phasing	0	3.464	11376	3.464	11.38	3.464	11376	3.464	11.4
	5	2.805	9204	3.3	10.7	4.206	13800	4.9	16.1
100 x 19323 burn	5	2.805	9204	3.3	10.7	4.206	13800	4.9	16.1
	5.4	7.014	23011	8.2	27.0	10.159	33329	10.6	34.8
100 x 19323 coast	5.4	7.014	23011	8.2	27.0	10.159	33329	10.6	34.8
	10.4	15.066	49431	1.5	4.9	>100		>15	
Orbit insertion burn	10.4	15.066	49431	1.5	4.9	Exceeds Mission Requirements			
	10.6	15.745	51659	4.7	15.5				
		G&N Capability Satisfies Mission Req'mt of 50KM and 5M/S							

Table A-3. Autonomous Navigation Accuracy - 100 NM Circular Orbit

Fig. No.	Random Noise		Alignment Bias Uncertainty	Meas Freq Sec	Initial State Error		State Error After 3 Hrs		State Error After 5 Hrs	
	Star Tracker	Horizon Tracker			Position KM	Velocity M/S	Position KM	Velocity M/S	Position KM	Velocity M/S
A-1	30 sec	6 min	15 sec	300	3.464	3.464	4.871	5.53	—	—
A-2			9 min				6.905	8.01	—	—
A-3			30 min				6.961	8.08	—	—
A-4			30 min	150			5.291	6.12	4.206	4.91
A-5			15 sec				3.713	4.21	2.805	3.25
A-6	30 sec	6 min	15 sec	150	45.721	18.29	4.407	5.10	3.276	3.79

Note: All values are 3σ

Table A-4. Autonomous Navigation Accuracy - Geosynchronous Orbit

Fig. No.	Random Noise		Alignment Bias Uncertainty	Meas Freq Sec	Initial State Error		State Error After 5 Hrs		State Error After 10 Hrs	
	Star Tracker	Horizon Tracker			Position KM	Velocity M/S	Position KM	Velocity M/S	Position KM	Velocity M/S
A-7	30 sec	4 min	15 sec	300	4.145	5.702	44.768	4.29	—	—
A-8			30 min				50.995	4.45	—	—
A-9		6 min	15 sec	150			45.041	4.32	—	—
A-10			30 min				51.267	4.49	—	—
A-11			15 sec	300	45.721	18.29	111.80	11.88	53.691	5.07
A-12	30 sec	6 min	30 min	300	45.721	18.29	132.13	12.08	73.440	6.73

Note: All values are 3σ

A1.2.3 Errors Accrued During Thrusting

The effect of inertial measurement errors, initial attitude alignment errors, and initial state errors upon the state errors at the end of thrusting maneuvers is shown in Table A-5. Error effects are divided into three groups. The major error sources are seen to be due to initial conditions (state errors and attitude alignment error) and gyro random drift.

A1.3 CONCLUSIONS

1. An autocollimator or equivalent alignment accuracy of 15 $\widehat{\text{sec}}$ is probably required to assure that mission injection accuracy requirements can be satisfied. Although this additional device to the baseline system adds weight, it alleviates conflicting mounting co-location and pointing requirements of the optical measuring devices. It also simplifies the preflight alignment procedures.

2. Results of preliminary analysis indicate that navigational performance capability is highly dependent upon the navigational sighting schedule as well as the error sources. An in-depth parametric analysis should be made to optimize the orbit determination process; such a study would quantify individual and combined effects of measurement frequency, sequentially correlated error, and data processing hardware for specific mission profiles and spacecraft characteristics.

Table A-5. Autonomous Navigation Accuracy - Thrusting Flight Segments

Error Sources	Burn No. 1			Burn No. 2		
	Error Value 3σ	Burnout Errors		Error Value 3σ	Burnout Errors	
		Position KM	Velocity M/S		Position KM	Velocity M/S
Initial position error	2.805 KM	5.922	5.48	15.066 KM	15.701	1.56
Initial velocity error	3.25 M/S			1.50 M/S		
Initial attitude misalignment	.1 deg	3.736	6.07	.1 deg	1.172	4.45
Gyro random drift	.1 deg/hr			.1 deg/hr		
Gyro "G" sensitive drift	.1 deg/hr/g	.474	.77	.1 deg/hr/g	.092	.34
Gyro torquer scale factor uncert	.01%			.01%		
Accelerometer bias	$.3 \times 10^{-4}$ g			$.3 \times 10^{-4}$ g		
Accelerometer scale factor uncert	.01%			.01%		
Accelerometer nonlinearity	$.1 \times 10^{-4}$ g/g^2			$.1 \times 10^{-4}$ g/g^2		
Total Error		7.014	3.22		15.745	4.73

Burn No. 1

- o Injection into 100 x 19300 n mi orbit
- o Delta Velocity = 8446 fps
- o Burn Time = 1307 sec

Burn No. 2

- o Insertion into geosync orbit
- o Delta Velocity = 5893 fps
- o Burn Time = 564 sec

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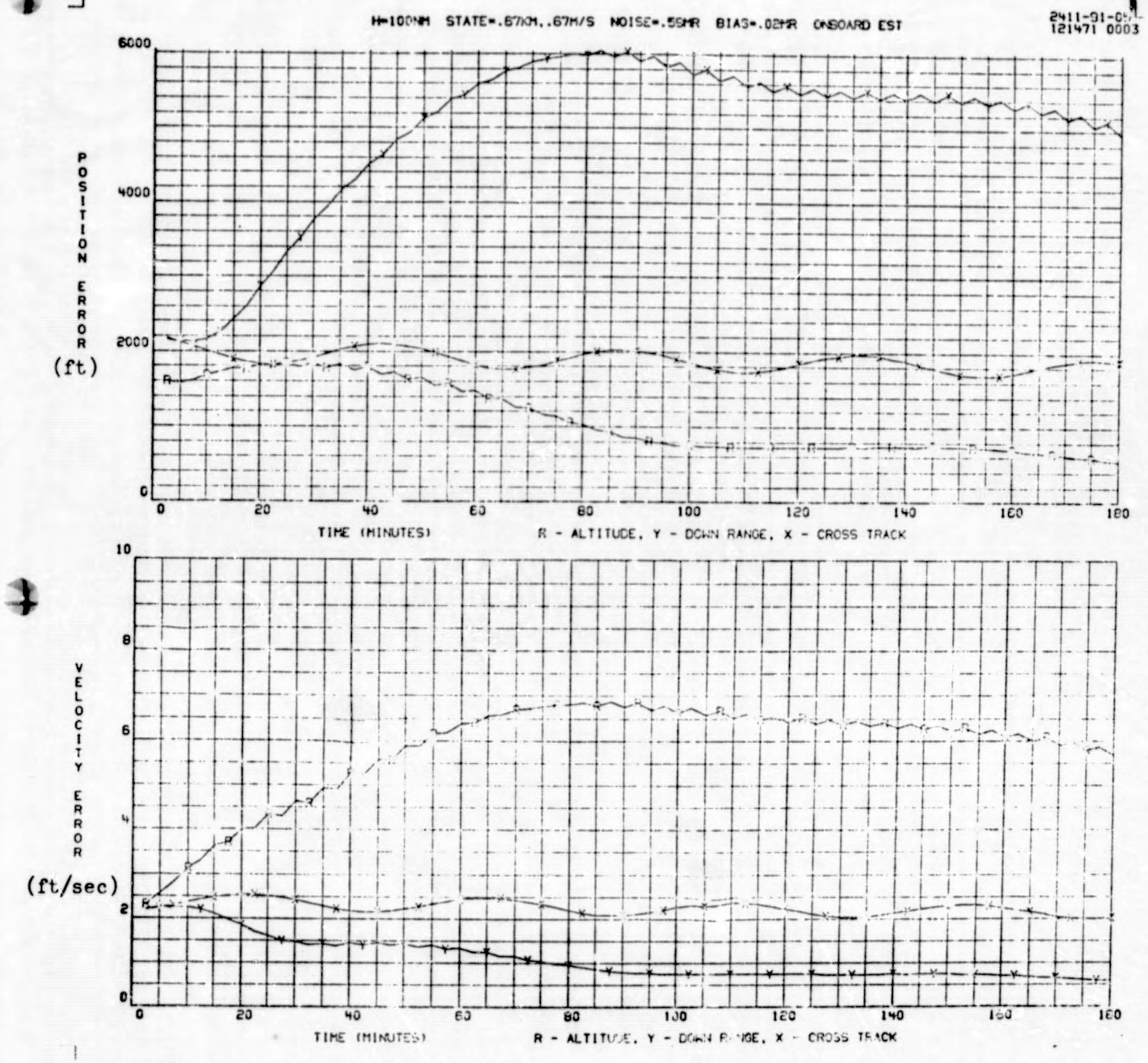
A-2. Farrar, R. J., "Survey of Autonomous Navigation Schemes for Space Shuttle Missions," Aerospace Corporation Report No. TOR-0172(2531-03)-3, 1 August 1971.

A-3. Madden, M. F., "Guidance and Navigation Techniques for Long Duration Manned Space Systems," NR Report No. TN 70-AT-4, November 1970.



NOTES: 1. WITH AUTOCOLLIMATOR

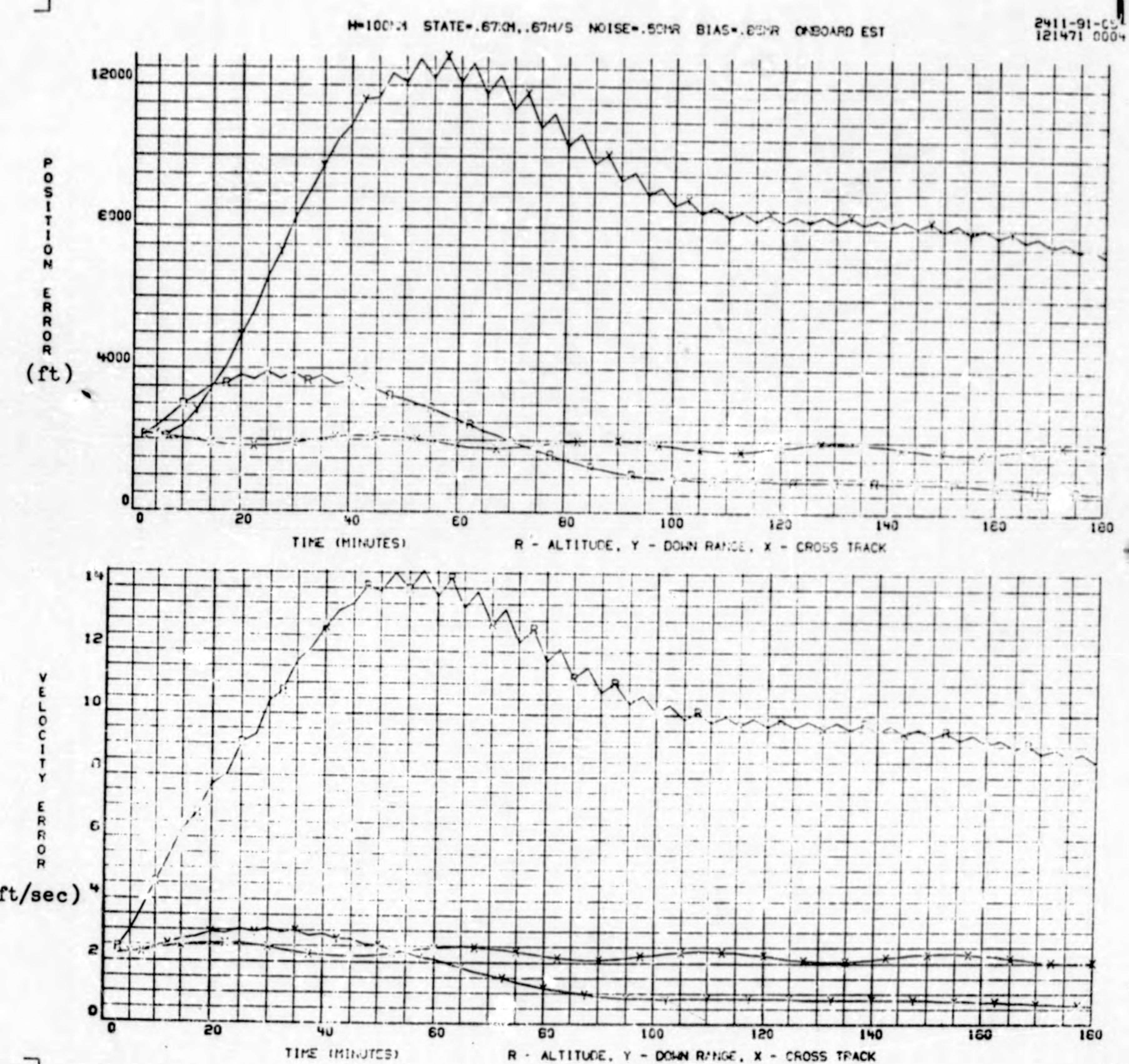
2. STAR FIXES AT 300 SEC INTERVALS

3. ALL CPT VALUES ARE 1σ

Figure A-1 Autonomous Navigation Accuracy

- 100 nmi Circular Orbit -

Space Division
North American Rockwell



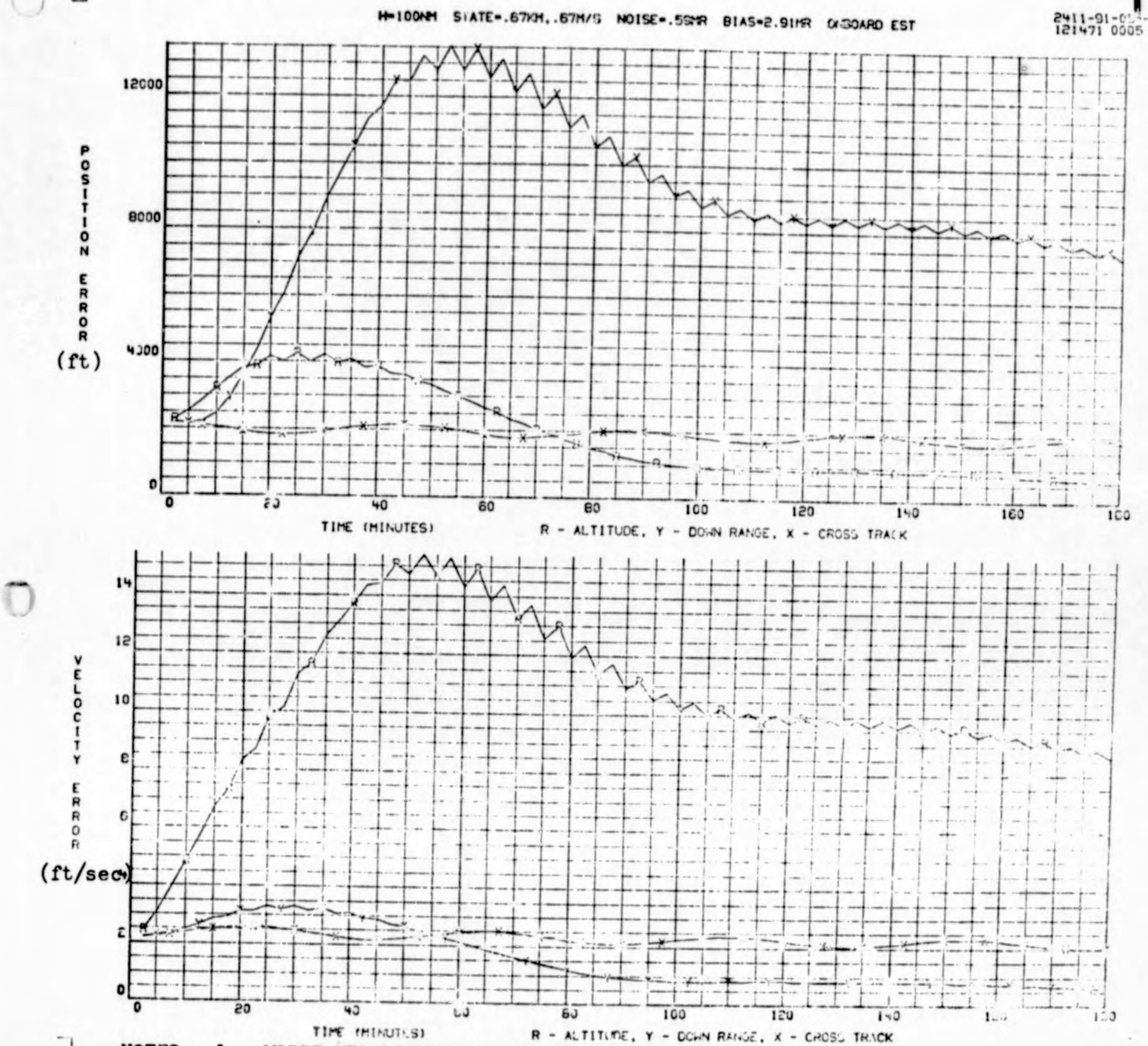
NOTES: 1. BEST ALIGNMENT WITHOUT AUTOCOLLIMATOR

2. STAR FIXES AT 300 SEC INTERVALS

3. ALL CRT VALUES ARE 1 σ

Figure A-2 Autonomous Navigation Accuracy
- 100 nmi Circular Orbit -

Space Division
North American Rockwell

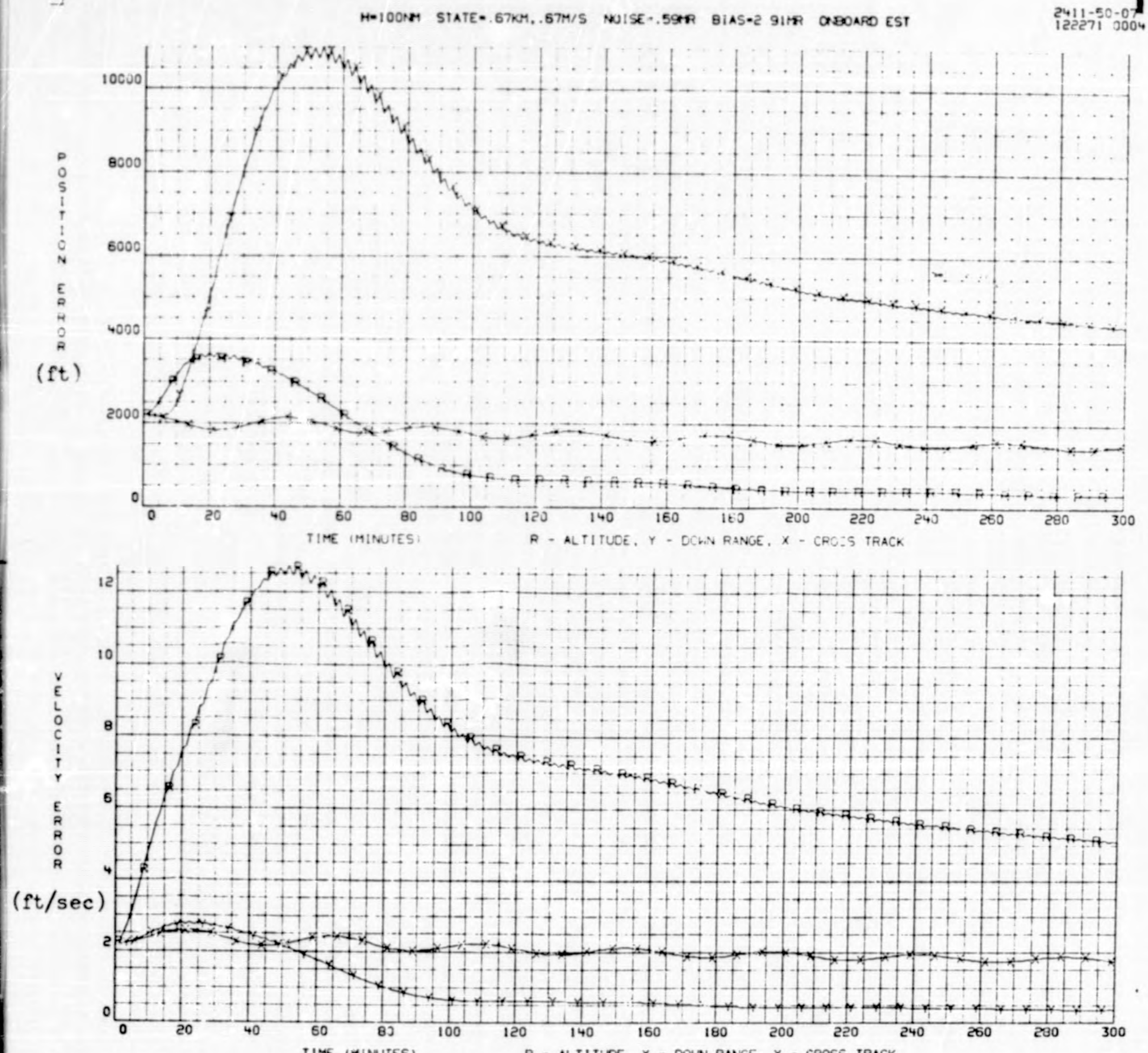


NOTES: 1. WORST MISALIGNMENT WITHOUT AUTOCOLLIMATOR

2. STAR FIXES AT 300 SEC INTERVALS

3. ALL CRT VALUES ARE 1 σ

Figure A-3 Autonomous Navigation Accuracy
- 100 nmi Circular Orbit



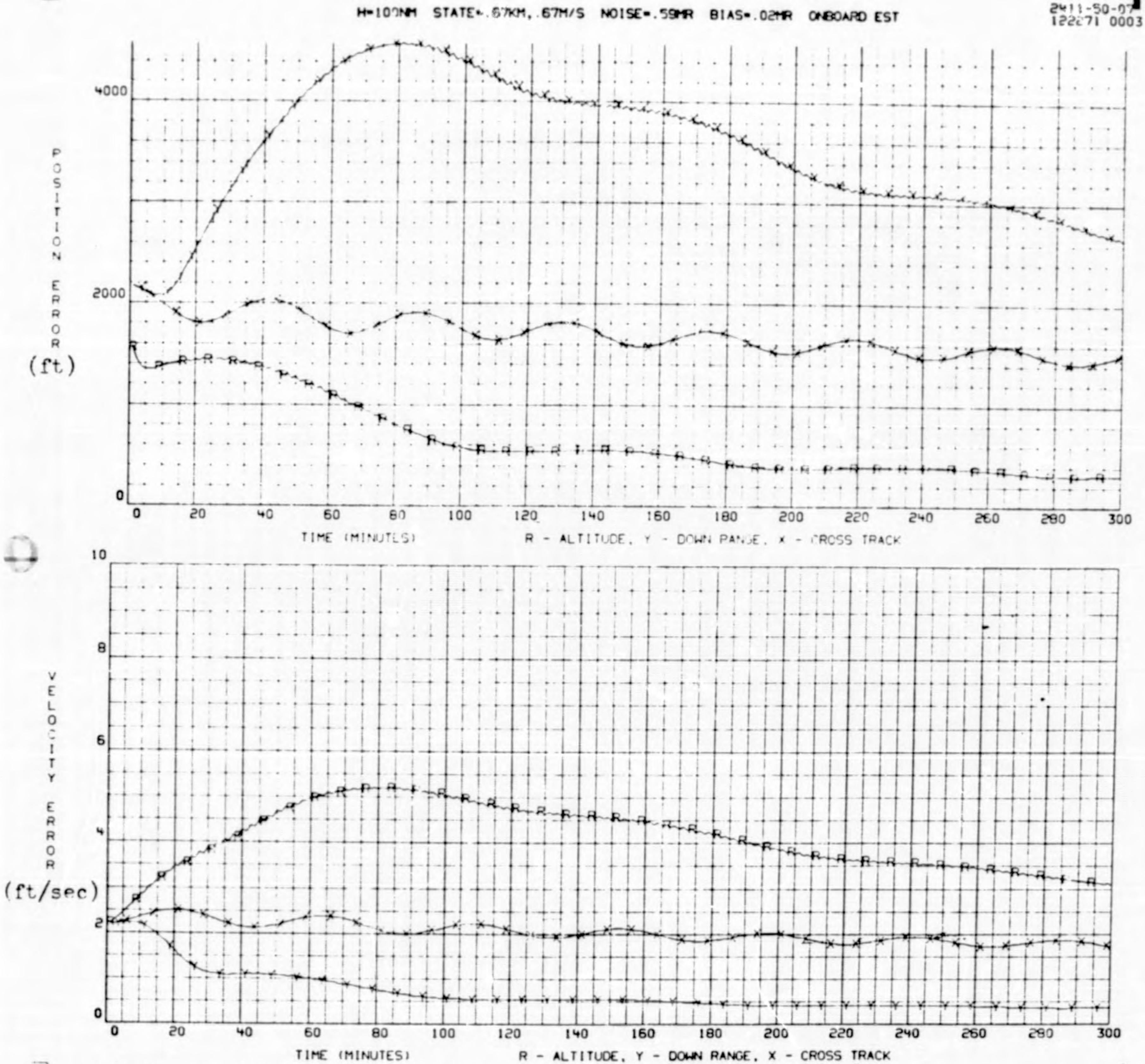
NOTES: 1. WORST MISALIGNMENT WITHOUT AUTOCOLLIMATOR

2. STAR FIXES AT 150 SEC INTERVALS

3. ALL CRT VALUES ARE 1 σ

Figure A-4 Autonomous Navigation Accuracy

\- 100 nmi Circular Orbit -

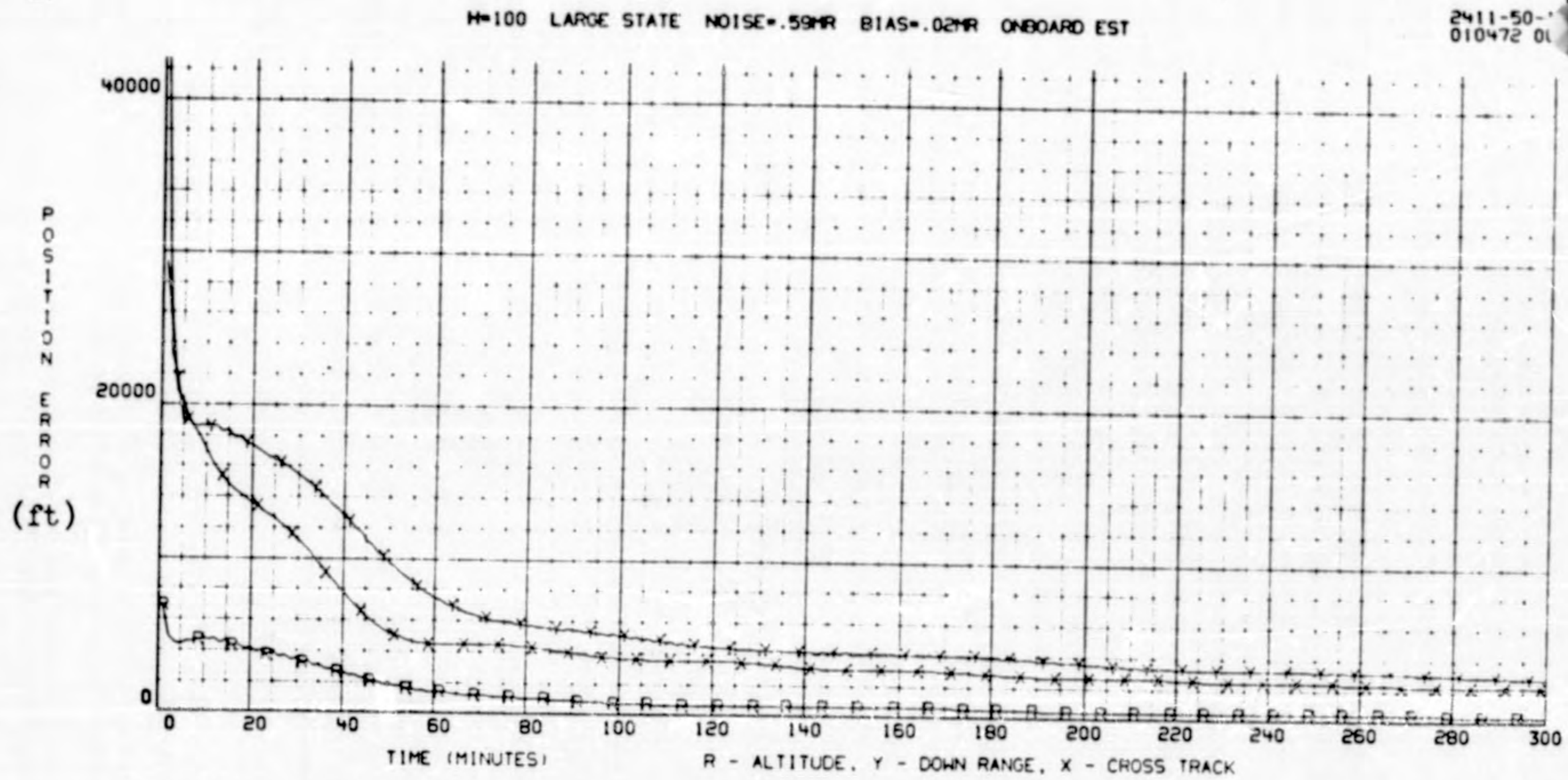


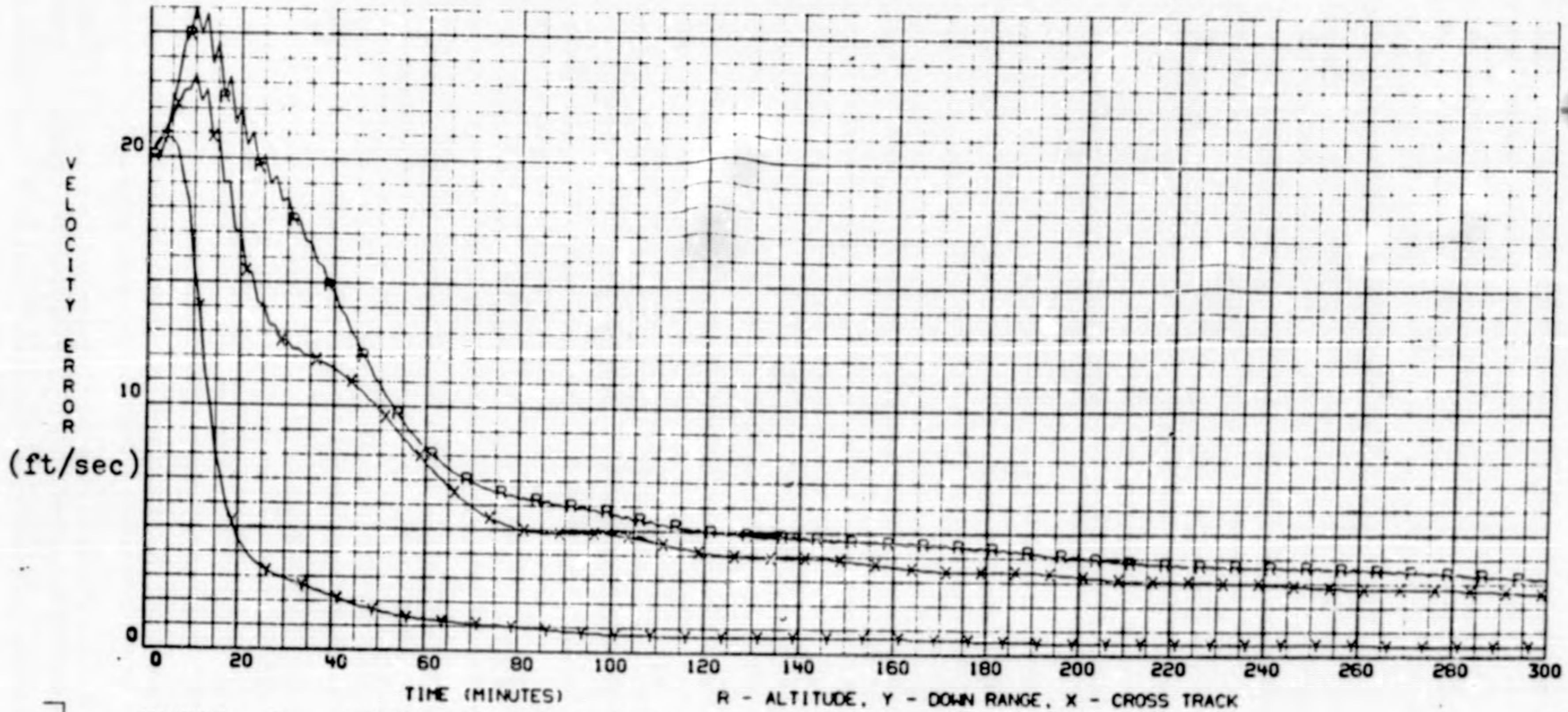
NOTES: 1. WITH AUTOCOLLIMATOR

2. STAR FIXES AT 150 SEC INTERVALS

3. ALL CRT VALUES ARE 1 σ

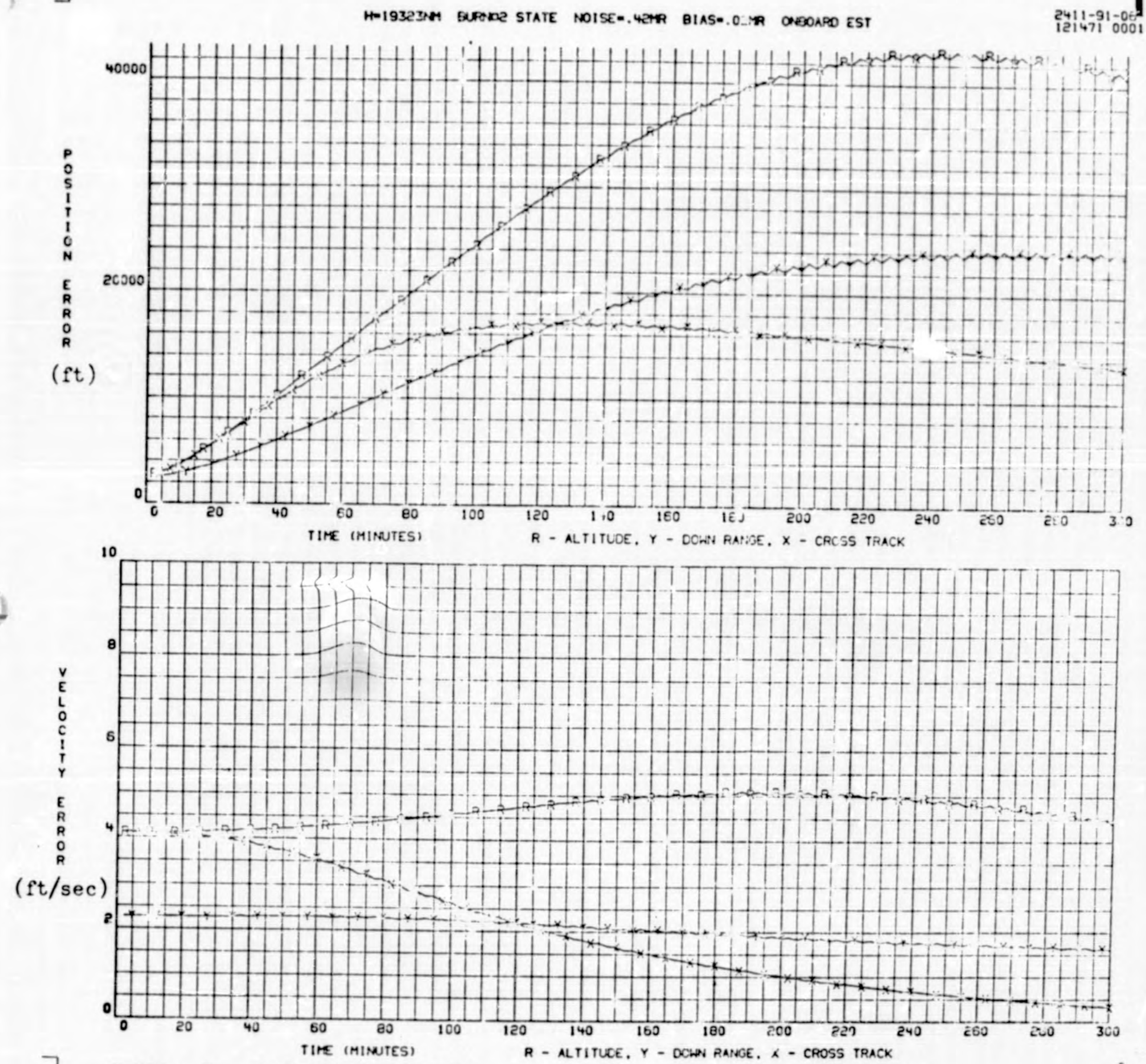
Figure A-5 Autonomous Navigation Accuracy
- 100 nmi Circular Orbit -





NOTES:
1. WITH AUTOCOLLIMATOR
2. STAR FIXES AT 150 SEC INTERVALS
3. LARGE INITIAL ERRORS
4. ALL CRT VALUES ARE 1σ

Figure A-6 Autonomous Navigation Accuracy
- 100 nmi Circular Orbit -



NOTES: 1. WITH AUTOCOLLIMATOR

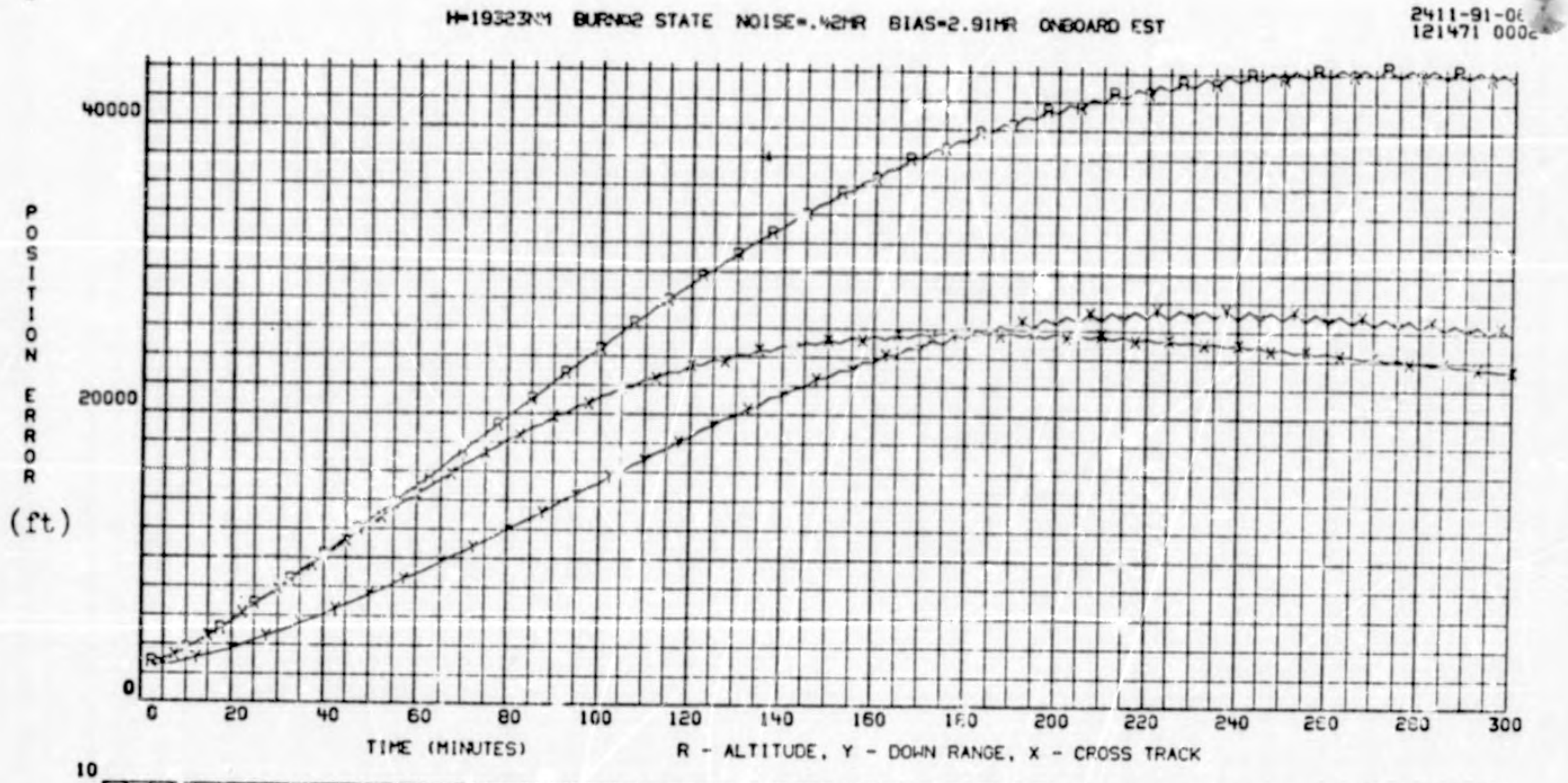
2. STAR FIXES AT 300 SEC INTERVALS

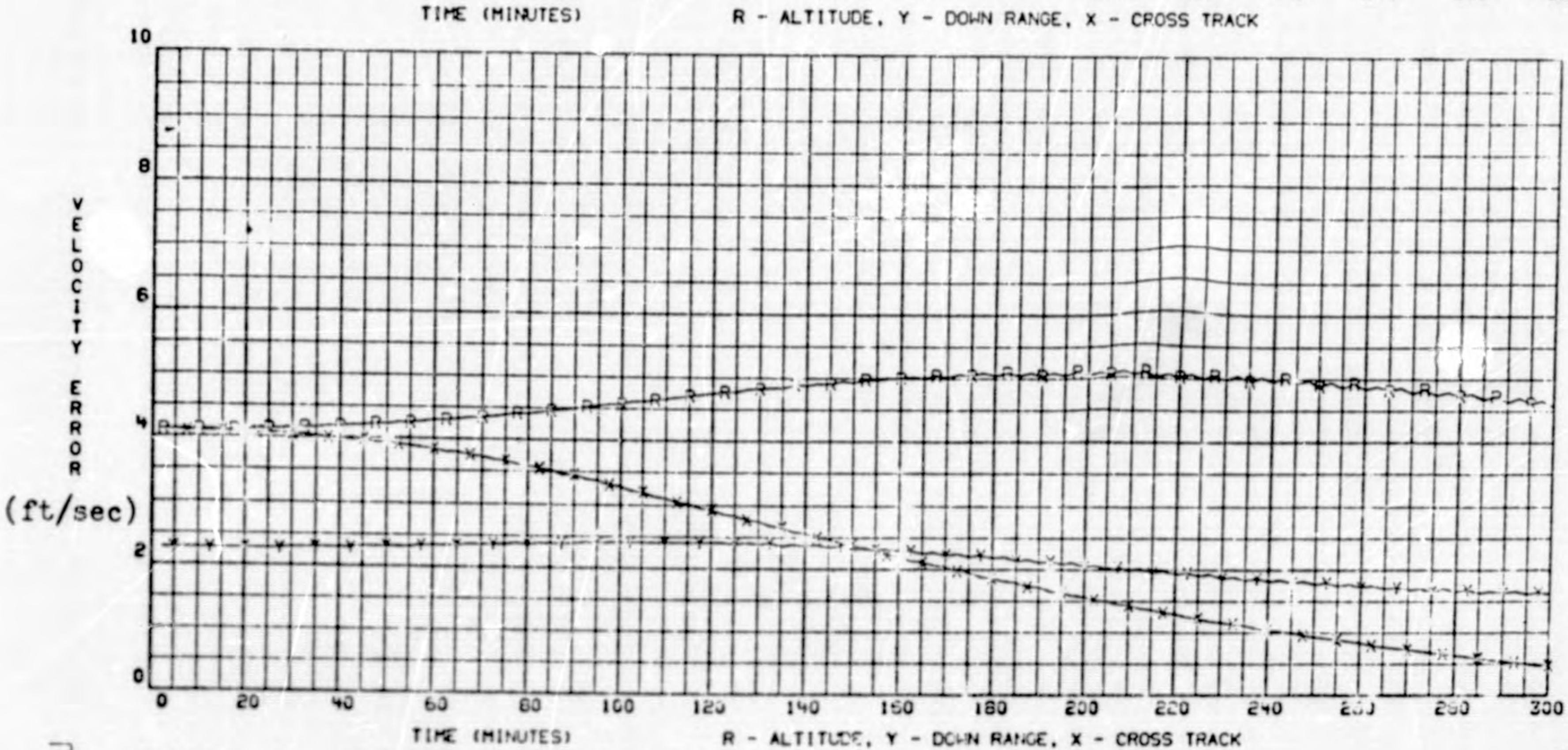
3. HORIZON TRACKER RANDOM NOISE = 4 $\widehat{\text{MIN}}$

4. ALL CRT VALUES ARE 1 σ

Figure A-7 Autonomous Navigation Accuracy

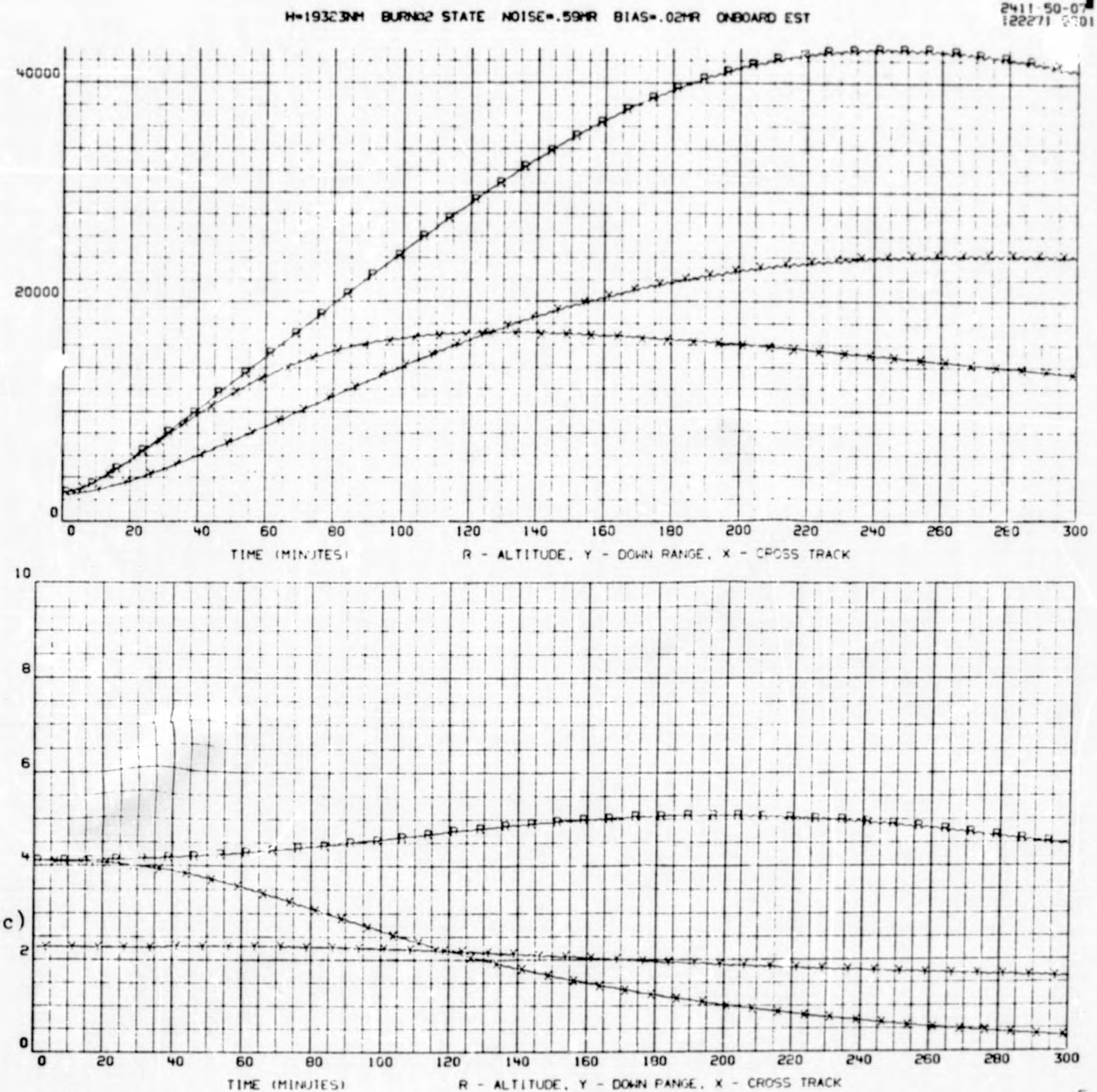
- Geosynchronous Orbit -





NOTES:
1. WITHOUT AUTOCOLLIMATOR
2. STAR FIXES AT 300 SEC INTERVALS
3. HORIZON TRACKER RANDOM NOISE = 4 $\widehat{\text{MIN}}$
4. ALL CRT VALUES ARE 1σ

Figure A-8 Autonomous Navigation Accuracy
- Geosynchronous Orbit -



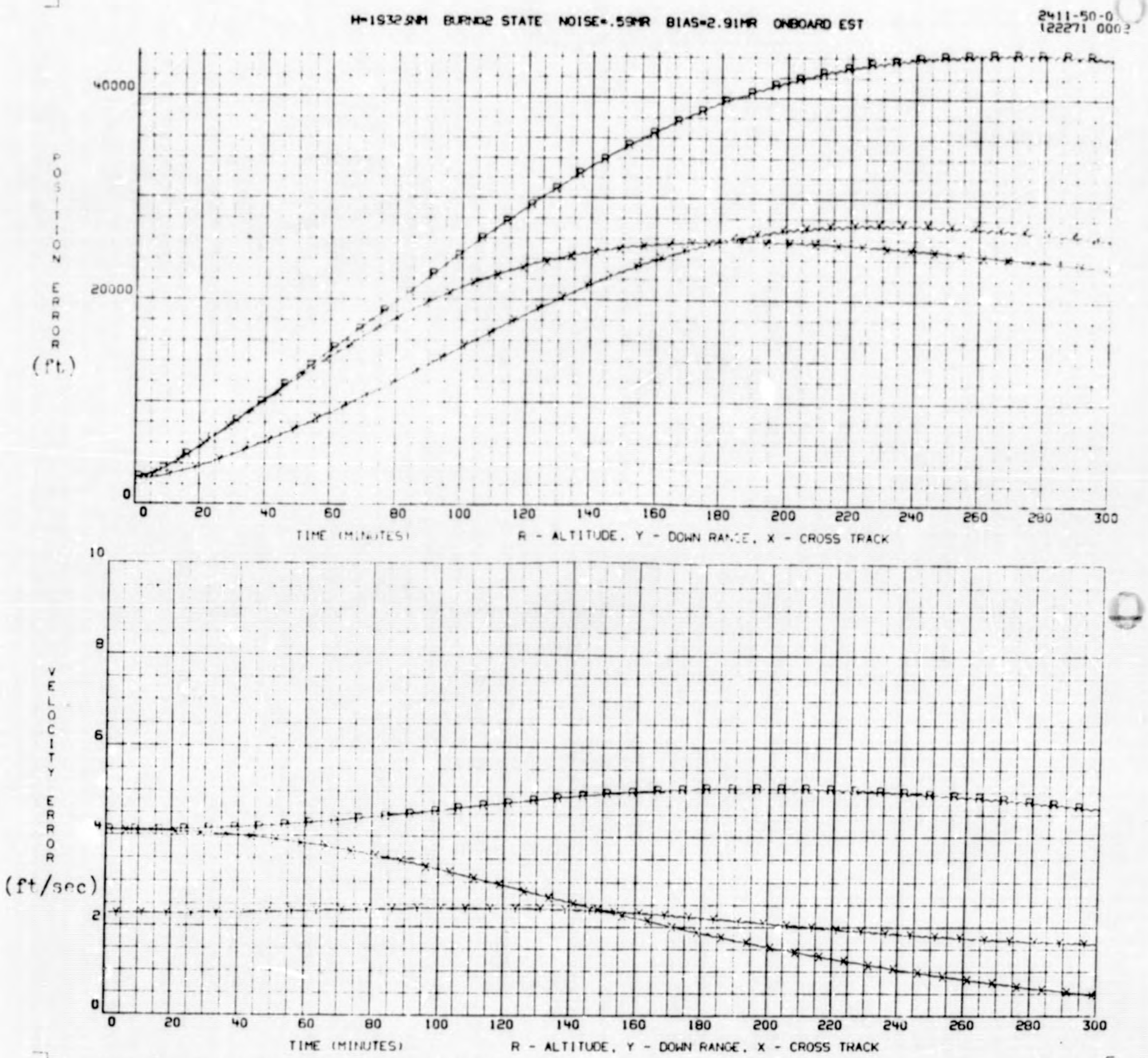
NOTES: 1. WITH AUTOCOLLIMATOR

2. STAR FIXES AT 150 SEC INTERVALS

3. HORIZON TRACKER RANDOM NOISE = 6 $\widehat{\text{MIN}}$

4. ALL CRT VALUES ARE 1 σ

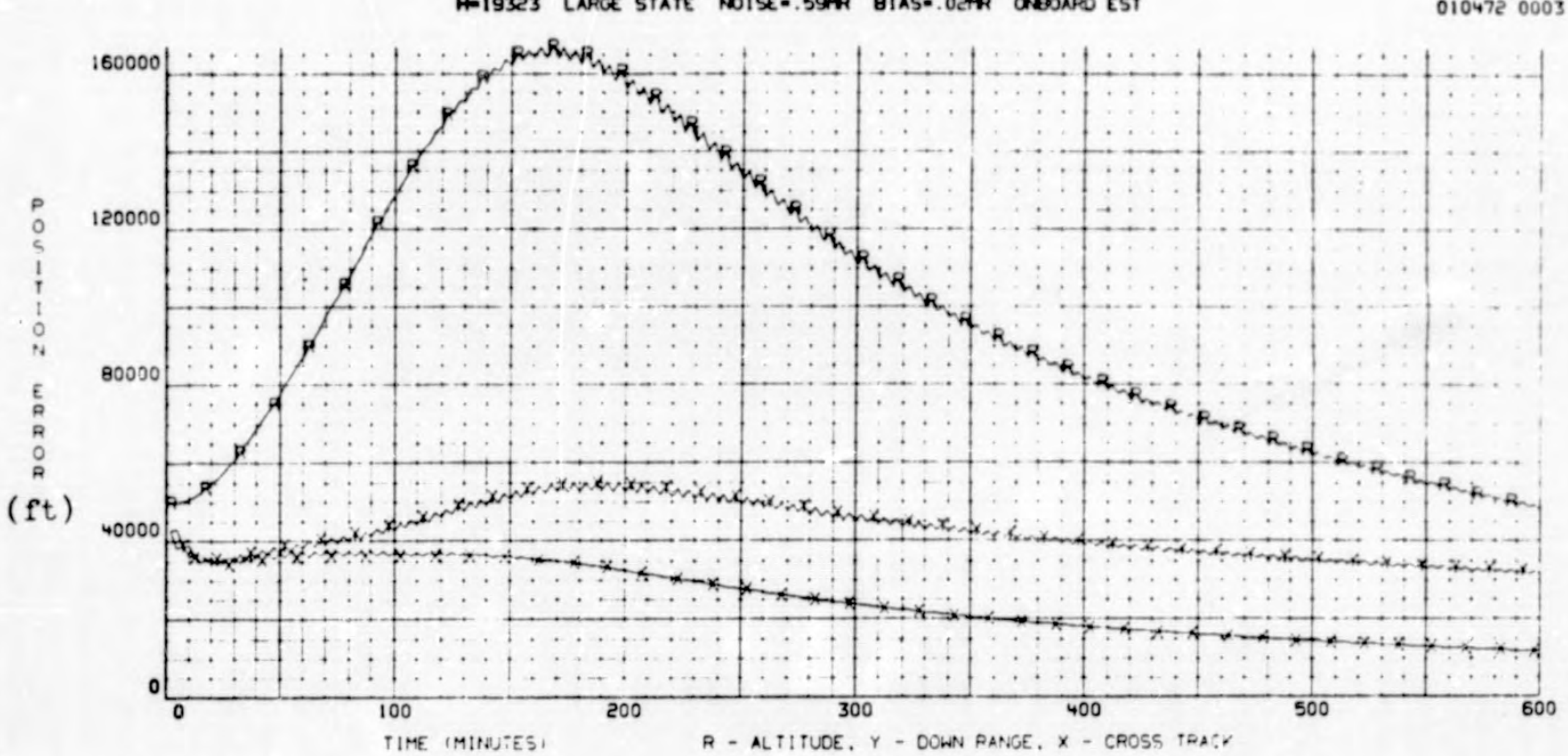
Figure A-9 Autonomous Navigation Accuracy
- Geosynchronous Orbit -

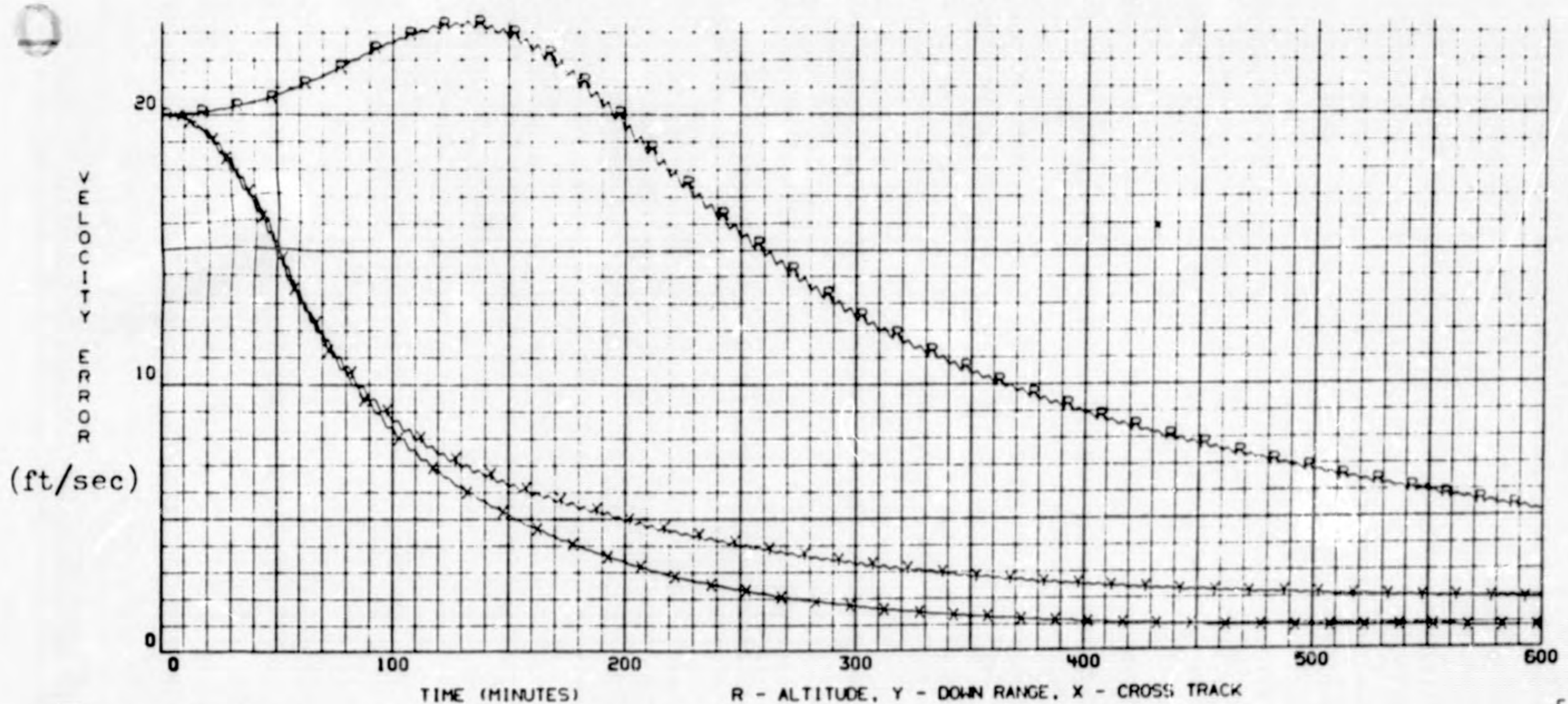


NOTES:

1. WITHOUT AUTOCOLLIMATOR
2. STAR FIXES AT 150 SEC INTERVALS
3. HORIZON TRACKER RANDOM NOISE = 6 MIN
4. ALL CRT VALUES ARE 1 σ

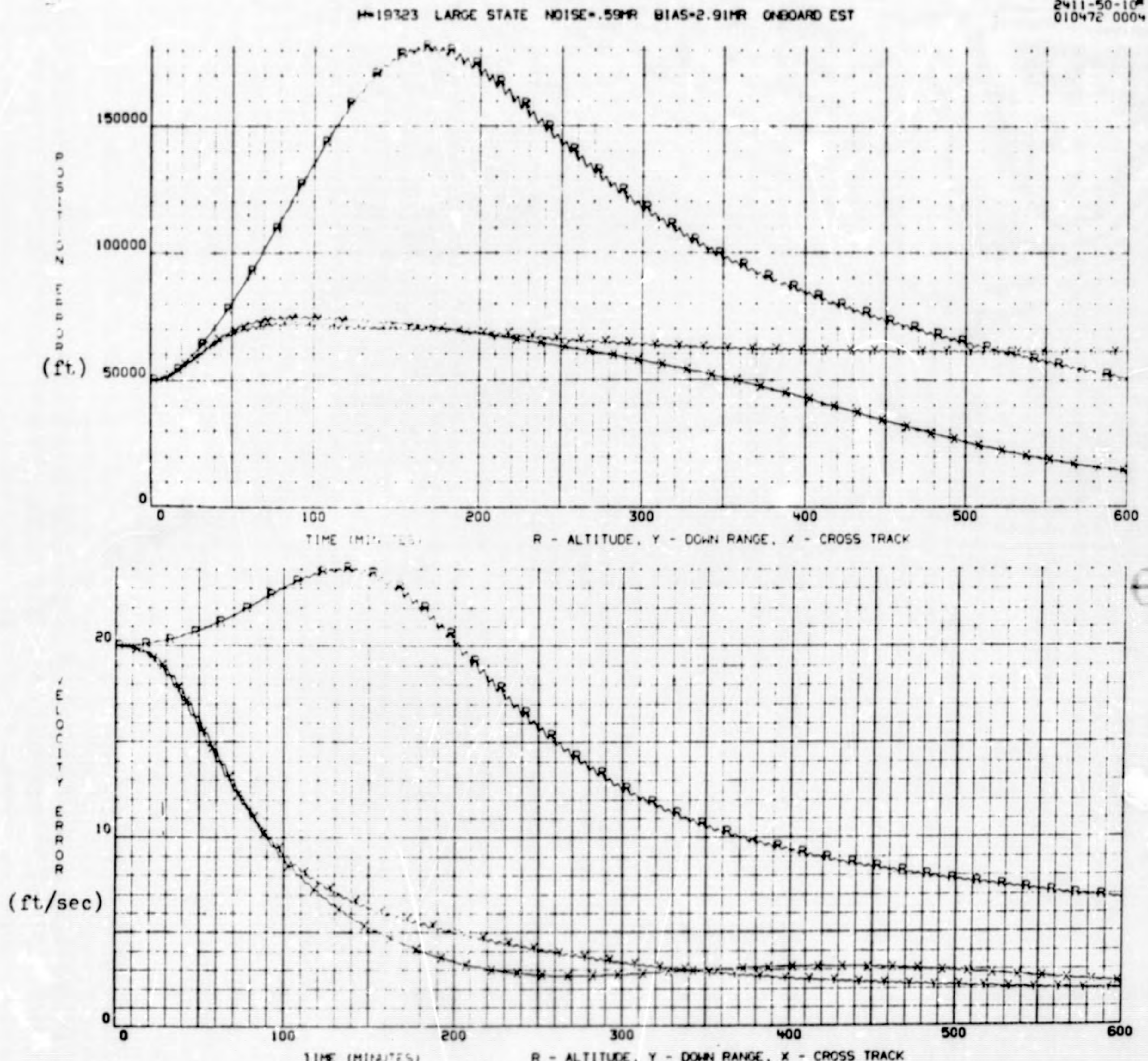
Figure A-10 Autonomous Navigation Accuracy
- Geosynchronous Orbit -





NOTES:
1. WITH AUTOCOLLIMATOR
2. STAR FIXES AT 300 SEC INTERVALS
3. LARGE INITIAL STATE ERRORS
4. HORIZON TRACKER RANDOM NOISE = 6 MIN
5. ALL CRT VALUES ARE 1 σ

Figure A-11 Autonomous Navigation Accuracy - Geosynchronous Orbit



NOTES:
1. WITHOUT AUTOCOLLIMATOR
2. STAR FIXES AT 300 SEC INTERVALS
3. LARGE INITIAL STATE ERRORS
4. HORIZON TRACKER RANDOM NOISE = 6 $\widehat{\text{MIN}}$
5. ALL CRT VALUES ARE 1 σ

Figure A-12 Autonomous Navigation Accuracy
- Geosynchronous Orbit -